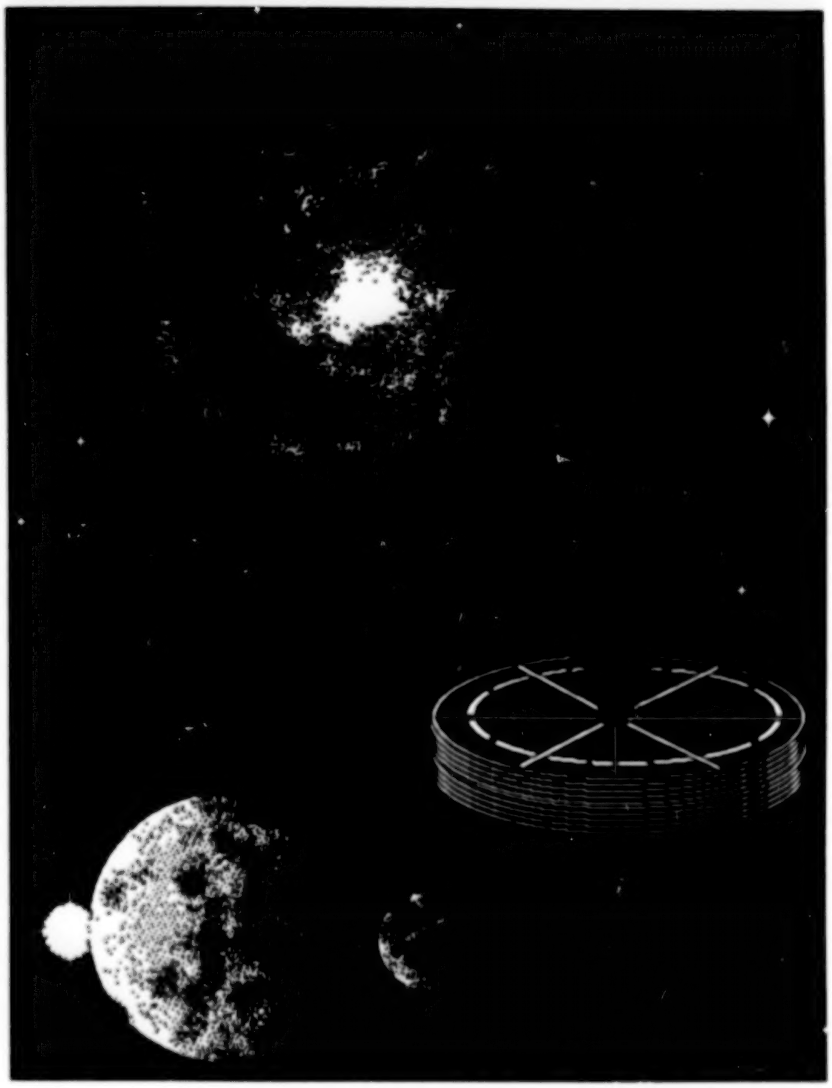
*NASA Conference Publication 10059*

# Vision-21: Space Travel for the Next Millennium



*Proceedings of a symposium held at the*
*NASA Lewis Research Center*
*Cleveland, Ohio*
*April 3-4, 1990*

NASA

(NASA-CP-10059) VISION-21: SPACE TRAVEL FOR N91-22139
THE NEXT MILLENNIUM (NASA) 600 p CSCL 22A --THRU--
N91-22181
Unclas
G3/12 0002882

NASA Conference Publication 10059

# Vision-21: Space Travel for the Next Millennium

*Geoffrey Landis, Editor*
*Sverdrup Technology, Inc.*

*Proceedings of a symposium held at the*
*NASA Lewis Research Center*
*Cleveland, Ohio*
*April 3-4, 1990*

NASA
National Aeronautics and
Space Administration

Scientific and Technical
Information Division
1991

# Space Travel for the Next Millenium

## by Jonathan V. Post

Space Travel for the Next Millenium
is no mirage, and no illusion --
we will travel to the stars
with antimatter, or with fusion

We will travel to the stars
where legendary dragons roosted
where the human heart aspires
on light-sails which are laser-boosted

Where the human heart aspires
we will master human factors
achieving high specific impulse
with low-weight nuclear reactors

Achieving high specific impulse:
patriotic, not quixotic
we'll use metallic hydrogen
or metastable fuels exotic

We'll use metallic hydrogen
or snowballs made of hydrogen ice
we'll freeze ourselves with cryogenics
the cool way into Paradise

We'll freeze ourselves with cryogenics
feed ourselves with hydroponics
trajectories, with thrust ionic,
hyperbolas or other conics

Trajectories, with thrust ionic,
puts probes where their makers sent them;
reactionlessly we can go
we can redirect momentum

Reactionlessly we can go
we'll be prestidigitators
moving asteroidal freighters
with tethers and space elevators

Moving asteroidal freighters
growing rich from space resources
coupling to space-time itself
to generate propulsive forces

Coupling to space-time itself
extracting vacuum energy
Oh, where's the loophole in the law
to let us travel over "C"?

Oh, where's the loophole in the law?
I would not die a planet-dweller.
Plasma in my blood cries out
for plasma interstellar.

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# I. Preface, Overview, and Symposium Welcome

# PREFACE

The symposium "Vision-21: Space Travel for the Next Millenium" was held at the NASA Lewis Research Center on April 3-4, 1990, sponsored by the Aerospace Technology Directorate of the NASA Office of Aeronautics, Exploration and Technology. There is, of course, an important near-term role for development of existing technologies and incremental improvements in hardware. However, there is also a role for considering more visionary possibilities beyond the next generation, to look at future technologies which might provide radical breakthroughs in performance for the twenty-first century and beyond. The object of the Vision-21 symposium was to gather people with interests in speculative concepts and advanced ideas.

Several invited speakers with international recognition in speculative thinking were recruited for the symposium. Keynote speaker for the symposium was Dr. Robert Forward, well known in the field of propulsion for studies on advanced propulsion systems and for his pioneering work on interstellar flight. Other invited speakers were Marvin Minski of the MIT Artificial Intelligence Laboratories, Joel Sercel of the Advanced Propulsion group at JPL, Theodore Taylor, former team leader for the "Orion" project, and Paul MacCready of AeroVironment.

The theme selected for the symposium was space travel for the next millenium. We had hoped that the participants would allow their focus to consider possible advances in technologies for space travel not just for currently envisioned projects, but to look at possibilities beyond the next generation and for the next thousand years. About half of the contributed papers focussed on propulsion, and half on other issues related to space travel. In a break from usual tradition, *all* of the contributed papers were presented in poster format. The intent of this was to maximize the interaction of participants with presenters and allow the presenters to answer questions and discuss concepts in as much detail as needed.

In addition to the invited and contributed papers, the symposium held a series of workshops on topics of interest. The purpose of these workshops was to gather ideas and allow interaction between all of the conference participants on an informal level, to explore and exploit synergies between the ideas of various participants.

We would like to thank all of the participants for making the symposium a success. I would also like to personally thank the symposium committee, consisting of Les Nichols, Sheila Bailey, Marc Millis, Ron Cull, George Madzar, and myself, for their time, and the Sverdrup team, led by Richard Ziegfield, John Toma, and Jack Harper, for their excellent work in making the symposium run smoothly.

Geoffrey A. Landis
Technical Chairman

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# Vision 21: Space Travel for the Next Millenium

We've sent astronauts to the Moon. We've sent sophisticated probes to every planet in our solar system and conducted scientific experiments aboard reusable space shuttles. A permanently manned space station is on the verge of reality, and technologies for manned bases on the Moon and Mars are coming into existence. But what about after that? What is needed to reach the more distant future, when humanity will use the resources in space, establish civilizations on other planets, and travel to the stars?

At NASA Lewis Research Center, a group called Vision 21 has spent the last two years developing the "spirit of inquiry" necessary to answer such questions. The Lewis scientists and engineers, who make up the group's ranks, spend the bulk of their time fine tuning the existing technologies that will allow us to take the next logical steps in space exploration and exploitation. They also believe, however, in the value of straying occasionally from the path of incremental progress to look far into the future and wonder how humanity will attain it.

Three years ago, Lewis Research Center's management also recognized the importance of such speculative thought. Lester Nichols, now Chief of LeRC's Interdisciplinary Technology Office, remembered the "old days" at Lewis when "no one was certain what could be done in space." In this atmosphere, where it was possible to explore fledgling ideas on a small scale, concepts such as the ion engine and the NERVA reactor were given a chance of survival.

Contrasting the past with the present inspired Nichols to meet with several of the Center's middle managers about the challenge of supporting new ideas. After extensive discussions, they agreed that the answer lay with LeRC's scientists and engineers. Creative scientists and engineers, encouraged by management's desire to hear their ideas, joined together to form an organization to encourage and develop ideas outside or beyond the scope of the mainstream NASA technological interests. They began meeting regularly to present their ideas and discuss them, developing and using a method called PINS in which the each listener responds to the idea by pointing out its **P**ositive, **I**nteresting, and **N**egative aspects and then making a **S**uggestion for overcoming the drawbacks. They explored the creative process itself, trying techniques such as brainstorming. They also looked at the more practical aspects of innovation, such as how to get funding. Quickly, some members began collaborating on ideas and publishing their results.

According to Marc Millis, a Lewis scientist who has facilitated the group from the beginning, Vision 21's official role at Lewis has evolved to one of "providing an environment for the open exchange of ideas and a process to explore the interesting possibilities that are too speculative and high risk to warrant official sponsorship."

As part of the process of creating an environment at Lewis hospitable to innovative thinking, the Vision-21 group organized a symposium to be specifically dedicated to exploring speculative ideas. The topic "Space Travel for the Next Millennium" was chosen as a focus. Volunteering their time and working with limited resources, Vision 21 members Geoffrey Landis, Sheila Bailey, Ronald Cull, and George Madzser invited speakers, planned an agenda, and publicized the symposium, all the time wondering just how many people would really be interested in attending such an event.

On April 3 and 4, 1990, they discovered that people from other NASA Centers, as well as from many of the nation's universities, research facilities, and businesses, share their desire to help shape our future in space. Nearly two hundred people attended the symposium held in Lewis Research Center's Development Engineering Building. Like other technical symposia and conferences, attendees submitted papers, listened to talks given by well-known experts, and participated in workshops. However, one of the most frequent remarks from attendees and speakers was that this was not a typical symposium. The emphasis was not on technical results but on achievements yet to be realized and the process which would lead

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to that realization. The symposium was not a place where experts examined painstakingly the problems and challenges of one specialized area; it was a place where imaginative people from a multitude of perspectives surveyed the vast landscape of the future.

Although the topics addressed in the talks, contributed papers, workshop sessions, and panel discussion included space power systems, computers, robotics, space resources, human factors, and more, the discussion kept returning to options for getting where we want to go: nuclear propulsion; antimatter propulsion; laser, solar, and magnetic sails; laser booster launch vehicles; and tethers. Why propulsion? Said keynote speaker Robert Forward: "In terms of our future in space, the three secrets to rapid and economical space operations are propulsion, propulsion, and propulsion."

Forward, a leading researcher on interstellar flight, usually talks about highly speculative ideas. However, for this event, he chose instead to address the question "What's THE next thing to do?", sharing his thoughts on the benefits and pitfalls of most of the propulsion options being developed, studied, or envisioned presently.

Other speakers speculated about what we will do when we reach our destinations. Joel Sercel of the Jet Propulsion Laboratory spoke about self-replicating machines for industrializing asteroids. Marvin Minsky, a pioneering researcher in artificial intelligence and robotics, proposed using teleoperated robots for building the space station. John Anderson from NASA Headquarters, who discussed a variety of speculative NASA programs, including tethers for propulsion, described how studies of sharks, dolphins, dragonflies, rats, slugs, and eagles were suggesting possibilities for technological innovations in hydronautics, aeronautics, astronautics, and computer science.

Paul MacCready and Ted Taylor chose to remind attendees of some of the earthly concerns that should never be forgotten as we look toward the stars.

MacCready, known for his work in human-powered flight, talked about some of the mental traps that humans face in trying to reach their objectives. His remarks on teamwork and motivation briefly shifted the symposium's spotlight from results to process.

Project Orion, a 1960's effort to build a spacecraft propelled by atomic bombs symbolizes for many the frustrations and triumphs of working on high risk ideas. Ted Taylor, who led the project, did not disappoint those who wanted to hear about what it feels like to lead the way toward the future of space travel. However, Taylor, whose project was canceled due to a changing political climate, did not dwell on the glorious feeling of having a vision. Instead, he concentrated on "features of a world where this vision exists" and "global concerns about how the vision may come to reality." Taylor reminded his audience that all of their visions are being threatened by the possibilities of extermination in nuclear war, destruction of the environment, and overpopulation. He advised them to strive for the abolition of nuclear weapons by the end of the century and for international cooperation in space.

The next step is to "harvest" some of the promising ideas that took root as a result of Vision 21's efforts, perhaps incorporating them into existing programs. Likewise, members of the group are forming more structured "working groups" to deal with some of the ideas that have begun to mature. However, they plan to keep working to develop the spirit at Lewis that nurtured these ideas. For example, they have started creating the "Idea Guidebook", a repository of information about "using your creativity at Lewis."

To continue perpetuating the network of imaginative people who came to Lewis this year, consideration is already being given to holding a second Vision 21 symposium, possibly in collaboration with another laboratory.

Deborah Vrabel, Editor, *Journeys in Thought*
Vision 21's newsletter

# Speculative Thought, Ideas, and Hardware at Lewis Research Center in the 1960's.

Lester D Nichols

Lewis Research Center was one of three National Advisory Committee for Aeronautics Centers which formed the original group of centers for the National Aeronautics and Space Administration in 1958. Among the research efforts which began at Lewis is that required for propulsion and power for manned space vehicles. We had research in the broad categories of nuclear electric power generation, nuclear thermal propulsion, electric propulsion, and other advanced propulsion concepts.

We began the nuclear power program with a study in 1959 of the 20 MW Rankine Cycle POwer Plant. There were many programs which followed, including the SNAP series, Brayton cycle studies, and magnetohydrodynamic power generation. The research spanned the scope of work from fundamental research to system development activities.

The mercury bombardment ion engine was invented at Lewis. In addition we carried out research in thermal ion engines, arc jets, magneto plasmadynamic arc jet, linear accelerators, colloidal ion engines, and linear induction accelerators.

The longer term research areas were not totally neglected. We had projects in atomic and metallic hydrogen, fusion propulsion, and even in the later times, LASER propulsion.

Lewis has a long history of interest in pushing the frontiers in propulsion and power. The present conference is just another activity toward that same goal.

# II. Keynote Speakers

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# EXOTIC POWER AND PROPULSION CONCEPTS

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## ABSTRACT

The status of some exotic physical phenomena and unconventional spacecraft concepts that might produce breakthroughs in power and propulsion in the 21st Century are reviewed. The subjects covered include: electric, nuclear fission, nuclear fusion, antimatter, high energy density materials, metallic hydrogen, laser thermal, solar thermal, solar sail, magnetic sail, and tether propulsion.

## INTRODUCTION

Chemical rockets have opened up space, landed humans on the Moon, put robotic landers on Venus and Mars, and sent flyby probes past all the major planets and moons in the solar system. All this despite the fact that any physicist can prove than any known chemical fuel doesn't have enough energy content to raise even itself into orbit, much less take any payload with it. Propulsion engineers proved the physicists wrong by designing multiple stage vehicles with extremely high mass ratios that reached 622:1 for the Saturn V moon rocket liftoff mass vs. the capsule mass that parachuted back to the Earth's surface.

If the United States decides it wants to construct a space defense, set up a lunar base, or explore Mars, then chemical rockets can do those jobs. But because of the relatively low specific impulse of chemical propellants, and the high overall mass ratios they imply for these difficult missions, the cost for accomplishing these tasks will be so high it is very likely that the United States will decide not to do the mission at all. If we are going to return to the Moon, explore Mars, and open up the solar system to rapid, economical travel, we need to find a method of propulsion that is an improvement over standard chemical rocket propulsion. That is the goal of that amorphous field of aerospace engineering called "advanced space propulsion".

## NUCLEAR PROPULSION

If the public can be sold on the idea of using nuclear propulsion for future space missions--fine. Proceed using that technology and ignore the rest of this paper. I suspect, however, that despite its real lack of danger and its great savings in cost and time, the nuclear rocket will not be a politically viable method of space travel. I will therefore write the rest of this paper and I encourage you to read it.

**Nuclear Thermal Propulsion**
Nuclear thermal propulsion is an advanced propulsion technology capable of producing thrust-to-weights greater than unity at high specific impulses of typically 825 seconds (nearly twice that of liquid-oxygen/liquid-hydrogen). Nuclear rockets were demonstrated to be feasible in the ROVER and NERVA solid core fission reactor test programs from 1959 to 1972, but unfortunately they were killed for political and budgetary reasons before they ever got off the ground. A summary of nuclear thermal rocket development and testing experience is covered in reference 1.

**Fusion Propulsion**
Since the nuclear fusion process typically converts three times as much mass to energy as the nuclear fission process, it has long been recognized that fusion fuels are much more energetic than fission fuels. Fusion propulsion is a wonderful idea, but its time has yet to come. Researchers still have not demonstrated a self-sustained controlled fusion reaction on the ground, and the reactor designs presently being funded by the Department of Energy are more suitable for massive power plants than lightweight rocket engines. That doesn't stop the advanced propulsion advocates from looking at new fusion propulsion concepts and designing new, lightweight fusion rockets. Whether these lightweight designs can ever be made self-sustaining is problematic, considering all the work put in on their larger power plant cousins.

Robert W. Bussard at $EMC^2$ has proposed (ref. 2) a low thrust fusion electric propulsion system that uses his Riggatron compact tokamak fusion reactor design designed operate on the difficult D-D fusion reaction. This reaction produces tritium, helium-3, and a fast neutron. The neutrons escape to space, while the hot (10-40 keV) tritium and helium-3 plasma is extracted at 30 atm pressure and mixed with a large amount of hydrogen gas diluent propellant to produce a high specific impulse exhaust. Bussard also speculates on an "electrostatic fusion propulsion" system using the reaction $p+{}^{11}B \rightarrow 3\ {}^{4}He$. In principle, the fusion product energy can be converted directly into electric power by causing the charged helium ions to expand against an electric field. This would result in a fusion-electric propulsion option with high specific impulse and high thrust.

V.E. "Bill" Haloulakas at McDonnell-Douglas and Bob Bourque at General Atomics carried out an Air Force Astronautics Laboratory sponsored study (ref. 3) of a $D-{}^{3}He$ fusion rocket using pulsed translating compact toroids that borrows from the DoE spheromak program. Again, thermal energy from the plasma heats a hydrogen propellant to obtain the optimum specific impulse.

In a combination of two technologies, Gerald Smith of Penn State has shown that antiprotons impinging on uranium atoms create fission nearly 100% of the time, releasing 180 MeV of fission fragment energy in the target. Under JPL sponsorship, Smith is now studying the use of small amounts of antimatter to trigger fission in the uranium shell of a pellet, which in turn will trigger fusion in the D-T mixture in the center of the pellet.

**Antiproton Annihilation Propulsion**

Antimatter propulsion is one of the long range, high risk, high payoff propulsion technologies. A series of studies (ref. 4 to 9) have shown that antimatter propulsion is not only physically and technologically feasible, it can be both cost effective and mission enabling. When antiprotons meet normal protons, all of the mass of both particles is released, not as gamma rays, but as elementary particles called pions. Two-thirds of those energetic particles are charged, and studies have shown that it is possible to extract a significant percentage (30-50%) of the energy as thrust (see Fig. 1). The optimized mass ratio of an antimatter rocket for any mission is typically 3:1, independent of the antimatter energy utilization efficiency or the mission △V. This low mass ratio enables missions that cannot be done using any other propulsion technique. The amount of antimatter needed for all missions within the solar system is measured in milligrams.

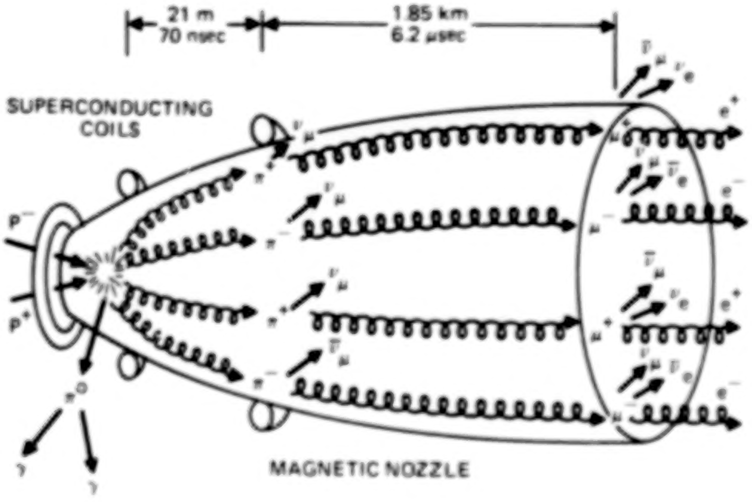


Fig. 1 - Schematic of Generic Antimatter Rocket

For example, Giovanni Vulpetti of Telespazio (ref. 10) has designed a reusable SSTO antimatter powered vehicle with a dry mass of 11.3 tons, payload of 2.2 tons, and 22.5 tons of propellant for a lift-off mass of 36 tons (mass ratio 2.7:1). This vehicle can put 2.2 tons of payload in GEO and bring back a similar 2.2 tons, while using 10 milligrams of antimatter. Moving 5 tons of payload from low Earth orbit to low Martian orbit with a 18 ton vehicle only requires 4 milligrams of antimatter.

The only source of low energy antiprotons is at CERN, in Switzerland. But Fermi National Accelerator Laboratory or Brookhaven National Laboratory could be modified to produce low energy antiprotons for less than $25 million. Ted Kalogeropolous at Syracuse has shown that present production quantities of antiprotons already are sufficient for non-destructive evaluation of rocket nozzles, as well as imaging and treatment of cancer tumors. Brian Von Herzen has formed the Antimatter Technology Corporation to commercially develop these medical applications.

The RAND Corporation has sponsored two Antiproton Science and Technology workshops (ref. 11) that found no showstoppers to antimatter propulsion, but determined that producing adequate quantities (grams per year) would require two successive generations of dedicated antiproton production facilities, a pilot plant to prove economic feasibility, followed by a large production plant. Realistically, this will take 30 years and 30 billion dollars, but it could save many hundreds of billions in the cost of future national space initiatives.

The present experimental effort in the field is concentrated on the capture and storage problem. Gerald Gabrielse of Harvard led an international team to CERN, captured 60,000 antiprotons in a cryogenic, ultrahigh vacuum electromagnetic trap no larger than a demitasse cup, and kept them trapped for 50 hours (ref. 12). The next step is adding positrons and making antihydrogen. The Air Force Astronautics Lab is looking into the growth of charged antihydrogen cluster ions as a method of condensing the antihydrogen while maintaining it in a trap. Under JPL sponsorship, George Seidel at Brown University is solving the problems of levitating milligram sized balls of antihydrogen ice. Since the electromagnetic and mechanical properties of hydrogen and antihydrogen are the same, the research is being carried out using normal hydrogen.

Antimatter rockets are a form of nuclear rocket. Although they emit insignificant amounts of neutrons, and the engines do not present a long term radiation hazard as do nuclear thermal rocket engines, they do emit gamma rays when operating, and require proper shielding and stand-off distance precautions. Unfortunately, the word "antimatter" still evokes raised eyebrows, mental images of Star Trek, and stifled giggles from upper level decision makers in the advanced propulsion branches of NASA and the Air Force. If they would read the scientific literature and be willing to consider technologies other than those that will produce results during their short time in office, they would find a new propulsion technology that could open up the solar system.

## EXOTIC CHEMICAL FUELS

High Energy Density Materials (HEDM) is the new Holy Grail of the chemical propulsion community. All the chemical elements are known, and nearly all the possible combinations of those elements into molecular compounds are known. Over the centuries since the Chinese invented gunpowder, there has been a continuing life-or-death motivation to find the most energetic of those compounds for use in propelling projectiles and rockets. To date, the most energetic fuel we have (that isn't deadly poisonous) is liquid-oxygen/liquid-hydrogen, which produces a maximum specific impulse of 500 seconds. Some would say there are no new chemical propellants to be found. The goal of the HEDM program is to somehow find a new chemical material with both a high energy density and a low molecular weight.

The major HEDM effort in the United States is at the Applied Research In Energy Storage (ARIES) Office at the Air Force Astronautics Laboratory, with another large effort in basic research being funded out of the Air Force Office of Scientific Research. A 1986 outgrowth of Project Forecast II, the Air Force HEDM program is putting more than $5 million a year into 50 R&D projects around the world. They hold annual contractors conferences where the results of the previous year are shared with the research community (ref. 13-15). There are a number of ways to increase the energy density of a fuel: Add light metals as atoms or small clusters, trap the energy of an excited electronic or vibrational state, force molecules to form highly strained bonds, and condense the material into a denser form.

Metastable helium fuel, made of electronically excited helium atoms (the easily-formed active ingredient in a helium-neon laser), has a projected specific impulse of 3100 seconds. The practical lifetime of metastable helium is less than one second, although theory projects an ideal lifetime of eight years. Early in the HEDM effort, Jonas Zmuidzinas of JPL investigated variations involving metastable helium molecules and solid metastable helium metal, with no positive results. No active research in this area is known of at this time.

Tetrahydrogen (an excited state molecule with four hydrogen atoms in a tetrahedron-shaped molecule) initially also looked promising, but detailed calculations on large computers showed it had a rapid-acting decay channel, verifying why it is not found in nature. The study of this system has led to other candidates, such as a-$N_2O_2$ (asymmetric nitrous peroxide), $Li_3H$, $FN_3$, and $B_2Be_2$. Theoretically, $B_2Be_2$ has a heat of formation of 238 kcal/mole and when unimolecularly decomposed gives a specific impulse of over 600 seconds. $FN_3$ has been prepared, is stable at low temperatures, and in addition to being an interesting monopropellant, also seems to have applications as a short wavelength chemical laser fuel and a high explosive!

Spin-polarized atomic hydrogen with a potential specific impulse of 2100 seconds has been produced in the laboratory by Daniel Kleppner of MIT in quantities large enough to cause damaging explosions in cryogenic glassware, but the lifetime of the atoms decreases drastically with density, due to an increase in three-body recombination collisions. Unless a way is found around this problem, it will not produce a usable fuel.

Unconventional molecules with "strained" bonds, such as variations on cubane (a cube made of carbon atoms with 90 degree bonds instead of the normal 180 degree linear carbon bonds) are being studied, both by supercomputers and test tube. Something may come of this research, but the increase in specific impulse over that of LOX-hydrogen will not be great.

Metallic hydrogen, a dense form of atomic hydrogen with a specific impulse of 1700 seconds and a density of 1.15 g/cc (compared to 0.07 g/cc for liquid molecular hydrogen), looks promising. H.K. Mao and R.J. Hemley at Carnegie Institution have used diamond anvil presses to apply pressures up to 300 GPa (3 Mbar) to micrometer sized samples of molecular hydrogen, trying to turn it into a superconducting metal. They have reported darkening of the sample (ref. 16), indicating absorption of light, but further work is needed to determine if it is a partially conducting form of molecular hydrogen or the desired metallic atomic hydrogen. Their darkened sample returned to its normal state when the pressure was released. Similar work, sponsored by the AF Astronautics Laboratory, is also underway by Isaac Silvera at Harvard.

Even if metallic hydrogen can be formed at high pressure, no one knows what will happen when the pressure is released. Some theorists predict it will be metastable, and stay in the dense metallic form. (Diamond is a metastable form of graphite formed at high pressure but stable at low pressure.) Other theorists predict it will rapidly revert to the molecular form of hydrogen. If it remains stable at some pressure substantially less than that necessary for form it, there is a lot of engineering to be done to move from micrometer sized batches to continuous flow production of tons per day, but with a specific impulse of 1700 seconds, metallic hydrogen will do everything for space travel that beamed laser power, nuclear thermal, and antimatter rockets could do, and be a lot cheaper and safer.

**Magnetic Engines and Nozzles**

Nearly all advanced high thrust, high specific impulse rocket concepts that use high power electromagnetic thrustors, metallic hydrogen or metastable helium fuels, beamed laser or microwave power, fission, fusion, or antimatter energy; in fact, any rocket technique that produces thermalized propellant exhausts with specific impulses above 1500 seconds, all have the same problem. The high energy exhaust from any of these processes will produce a blazing plasma that will melt the reaction chamber and nozzle if they are made out of ordinary refractory materials. One solution is to make the engine and nozzle out of magnetic fields. There are two ongoing experimental research efforts on magnetic nozzles to handle these high density, high temperature plasmas. One by Joel Sercel at Caltech sponsored by JPL, and one by Ted Yang and astronaut Franklin Chang-Diaz at MIT sponsored by JPL and AFOSR. Some recent studies (ref. 17), however, have uncovered a potential problem. Plasma constrained to an axially symmetric flow by an axially symmetric magnetic field will experience a resistive drag as it tries to axially detach itself from the radially diverging magnetic field lines. This drag will be transmitted to the vehicle through the magnetic field coils. Research in the area of magnetic field assisted reaction chambers and exhaust nozzles needs to be continued and expanded. Otherwise, we may find that we have developed a new propulsion energy concept without having the means to convert that propulsion energy into propulsive thrust.

## LIVING WITHOUT ROCKETS AND LIKING IT

If in the next few years space nuclear propulsion proves to be politically unpalatable, and the HEDM programs do not produce a new chemical fuel with a significant increase in specific impulse over liquid-oxygen/liquid-hydrogen, then those in charge of the future of this nation's space programs are going to face a harsh reality. Our future in space can only be assured if we give up our dependence on self-contained rockets. A rocket not only has to carry its payload, but it must also carry its engine, its energy source, and its reaction mass. If we want rapid, economical space travel within the solar system, we must develop and demonstrate new technologies that are not rockets and are not limited by the exponential mass growth of the rocket equation. Fortunately, there are plenty of candidates. Some examples are: beamed power laser propulsion, solar thermal propulsion, solar sails, magnetic sails, and tethers. Some do not carry their engine, some do not carry their energy source, some do not carry their reaction mass, and some do without all three.

## LASER THERMAL PROPULSION

Beamed power laser propulsion received a big boost in the past few years. Since 1987, SDIO has sponsored a two million dollar per year research program on laser propulsion managed through Lawrence Livermore National Laboratory (LLNL). Most of the effort has focussed on the nozzle-less planar thruster originally suggested by Arthur Kantrowitz (ref. 18). The payload sits on a solid block of ablative propellant such as plastic or water ice (see Fig. 2). An "evaporation" laser pulse ablates a few-micrometer-thick layer of propellant, forming a thin layer of gas. A second laser pulse then "explodes" this gas layer, producing thrust on the plate of propellant. The process takes a few microseconds and is repeated at 100-1000 Hz rates. An important feature is that the explosive expansion takes place so close to the plate of propellant that no nozzle is needed. The resultant thrust is normal to the plate and independent of the direction of the incident laser light, allowing the vehicle to fly at an angle to the laser beam. The vehicle can therefore transition into a near-circular orbit without requiring an apogee kick motor. The vehicle is steered from the ground by moving the laser beam off the center of the base plate, and so does not need an onboard guidance system. The payload size depends on the laser power; 20 MW can launch a 150 kg vehicle carrying a 20 kg payload. Higher powers can launch proportionately larger payloads. Peak accelerations are comparable to those of chemical rockets.

Jordin Kare of LLNL (ref. 19) reports that experiments have now been conducted at several industry and government laboratories. The double-pulse thruster concept works, producing high thrust efficiency and specific impulses up to 800 seconds. The actual thrust efficiencies obtained to date are only about 10%. The LLNL-SDIO program had hoped to use the induction linac fee electron laser (FEL) proposed by LLNL for SDIO tests at White Sands to do high-power suborbital (and possibly orbital) launch tests. SDIO has now decided to build a lower power RF-linac FEL, which puts out a poor pulse format for pulsed laser propulsion.

Another approach to laser propulsion is to absorb laser light in a plasma "flame" sustained by laser light focused in the center of a flowing stream of propellant gas. Thrust levels as high as 10,000 N with specific impulses of 1000 seconds appear achievable using hydrogen as the propellant gas. Dennis Keefer and others of University of Tennessee, working with a 1 kW CW $CO_2$ laser, have reported an absorption efficiency of 86% and a thermal efficiency of 38% in an argon plasma at 2.5 atm. They repeated the experiments using an RF-linac free-electron-laser that produces a 0.1 ms burst of 10 ps pulses separated by 46 ns. These short pulses ignited the argon gas and formed a plasma that absorbed 92% of the laser power (ref. 20). Hydrogen and nitrogen gas did not ignite at the pulsed laser powers available.

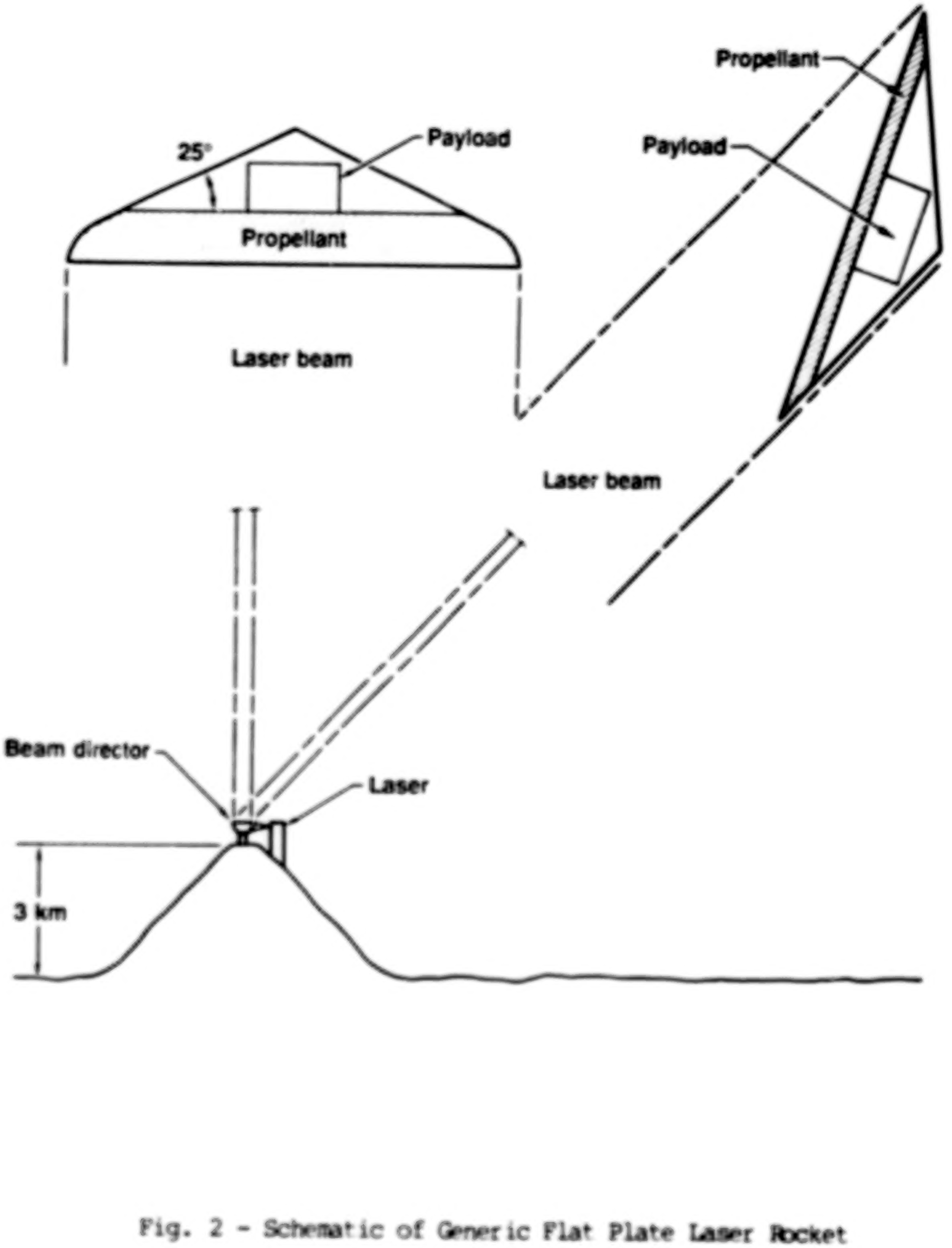

25°
Payload
Propellant
Laser beam
Propellant
Payload
Laser beam
Beam director
Laser
3 km

Fig. 2 - Schematic of Generic Flat Plate Laser Rocket

Herman Krier and Jyotirmoy Mazumder of University of Illinois at Urbana-Champaign have recently achieved a very promising 81% absorption efficiency and a 72% thermal efficiency with 7 kW of $CO_2$ laser power into a 2.5 atm hydrogen plasma flowing at 10.3 m/s. Leik Myrabo of Rensselaer Polytech (ref. 21) is investigating a 300 kg launch mass Lightcraft Technology Demonstrator that rides up a laser beam from a 100 MW-class ground-based free-electron-laser. The laser power heats scooped air in the atmosphere and onboard propellant in space, pushing the 124 kg spacecraft into orbit.

Laser powers as low as 1 MW would useful for LEO-GEO orbit raising without relay optics. 10-100 MW lasers can launch small payloads from the ground. With up to 100 launches a day, a 20 MW, 20 kg payload launcher could place several hundred tons in orbit per year. Low gigawatt lasers could launch multiton spacecraft with the same ease that present multigigawatt chemical rockets do. Laser rockets will have better payload fractions since the heavy power plant is left on the ground and the higher specific impulse results in lower propellant fractions. Although gigawatt lasers are not off-the-shelf items, there is no doubt they could be built if the need was strong enough.

## SOLAR THERMAL PROPULSION

One method of obtaining power and propulsion in space is to use large inflated concentrating mirrors to gather and focus solar energy onto a light-absorber which converts the solar energy into thermal energy. The thermal energy can be used to operate a heat engine to produce electricity, or it can be used to heat propellant (typically hydrogen) which can then exhausted to produce thrust. The major effort in this area is being sponsored by the Air Force Astronautics Laboratory (AFAL). Their program (ref.22) has proceeded through the research phase and is now directed toward flight tests in the late 1990s. In the mid 1980s, Rocketdyne built a small thrustor for AFAL consisting of a cylindrical cavity lined with rhenium tubing through which flowed hydrogen gas. Sunlight focused into the cavity from a 25 kW solar facility at AFAL produced 4.45 N of thrust at a measured specific impulse of 820 seconds.

AFAL is now investigating two advanced forms of thrustors. One uses a porous disk heat exchanger with a series of stacked discs of varying optical absorptance but constant hydrogen flow rate. The other is a directly heated gas concept where, similar to the CW laser propulsion experiments, the solar energy is absorbed in a solar sustained plasma "flame" in a flowing gas.

In the mid-1980s, L'Garde constructed a three meter on-axis diameter demonstration model inflatable concentrator for AFAL. The measured concentration ratio was a very respectable 12,000:1. More recently, a 4 by 6 m off-axis inflatable reflector was manufactured with a concentration ratio of 10,000:1. New fabrication approaches seem to indicate that full-sized, off-axis 30 m diameter concentrators are in hand. The design goals for the Astronautics Laboratory orbital transfer solar thermal propulsion system are: two mirrors of 30 m projected diameter delivering 1.5 MW of thermal power at a concentration ratio of 10,000:1 and two thrustors operating at a specific impulse of 900 seconds and 222.5 N thrust (445 N total).

## SOLAR SAILS

Solar sails are large, lightweight reflectors attached to a spacecraft that use light pressure from solar photons to obtain thrust. By tilting the sail to change the force direction, the light pressure can be used to increase the orbital speed of the spacecraft, sending it outward from the Sun, or decrease its orbital speed, allowing it to fall inward toward the Sun. Although the thrust available from sunlight is small (9 N/km$^2$), the solar sail never runs out of fuel. Over a long enough time, the small thrust can build up into extremely high $\triangle$Vs, allowing solar sails to take on missions that cannot be done by vehicles limited by the exponential growth of the rocket equation. A solar sail is ideal for shuttling of interplanetary cargo, since no refueling is required. Because the acceleration levels increase dramatically as the sail gets closer to the Sun, the solar sail exhibits tremendous performance for Mercury or Solar Probes, and many missions to the outer planets often benefit from an initial inward trajectory. (This was particularly true for the rendezvous mission to Halley's comet in its retrograde orbit.) Another ideal mission for a solar sail is a multiple small body rendezvous mission to the asteroid belt. Solar sail "tugs" can then be sent out to drag the more promising asteroids into an Earth or Mars orbit. Once the "pipeline" from the asteroid belt is full, the long transit time of solar sails hauling large cargos becomes academic.

In 1976-77, JPL carried out detailed engineering studies (ref. 23) on a square sail and a 12 blade "heliogyro" sail designed to rendezvous with Halley's Comet, not just fly by at high relative speed. The solar sail lost to solar electric propulsion, which in turn lost to the budget cutters. Solar sail studies were kept alive in the 1980s by Robert Staehle and a volunteer group of Los Angeles area engineers. They formed the World Space Foundation, which built the first solar sail in 1981. This "brassboard" model was deployed on the ground in order to confirm the packaging and deployment configuration. They presently have an engineering development model in design prototype form and are looking for a piggyback launch in order to verify the deployment procedure and fly a test mission to the Moon.

In 1989 the Columbus 500 Space Sail Cup race was announced. The purpose is to launch three or more solar sails into high earth orbit where they will undertake to travel to the Moon, and perhaps to Mars. The three lead vessels, named after the Nina, Pinta and Santa Maria, will come from three continents. One from Europe--the origin of Columbus's voyage, one from the Americas--the land Columbus discovered, and one from Asia--the land Columbus tried to reach and thought to have found. The lead ship selected for the Americas entry is the Johns Hopkins University Applied Physics Laboratory "Sunflower", a circular solar sail held in a flat circle by a large hoop supported by guy wires from a central mast. The sail has a diameter of 170 m and total mass of 180 kg. The sail is composed of 480 triangular pieces of reflective foil arranged like the petals of a flower. Some petals twist about their long axis to provide roll torque. No long seams are used, making it easy to manufacture, and each petal is individually unrolled by small deployment springs. Although the Columbus 500 Space Sail Cup Committee still has not obtained the funding required, the project is continuing ahead.

**Solar Photon Thrustor**

In 1988, a new type of solar sail called a "solar photon thrustor" (ref. 24) was invented (it was later found to have been first described by A.P. Skoptsiv of the USSR in 1971). The new sail concept is based on the realization that a space vehicle that uses a solar sail for propulsion can be significantly improved in performance by separating the function of collecting the solar photons from the function of reflecting the solar photons (see Fig. 3).

In the Solar Photon Thrustor concept, the collector is a large reflecting surface similar in size and mass per unit area to that of a standard flat solar sail. The collector faces the Sun so as to always present the maximum area for collection of sunlight. The collector is modified in structure so it is a light concentrator. The concentrated sunlight is directed to a reflecting surface of much smaller mass, which redirects the light to provide net solar photon thrust in the desired direction. Note that by tilting the reflecting mirror, the sunlight can be reflected in any desired angle off the axis formed by the Sun-spacecraft line, while rotation of the whole spacecraft around the Sun-spacecraft line allows direction of the reflected sunlight in azimuth around the Sun-spacecraft line. To minimize undesired torques, the collecting and reflecting portions of the system can be arranged so that the net force passes through the center of mass of the total system including payload.

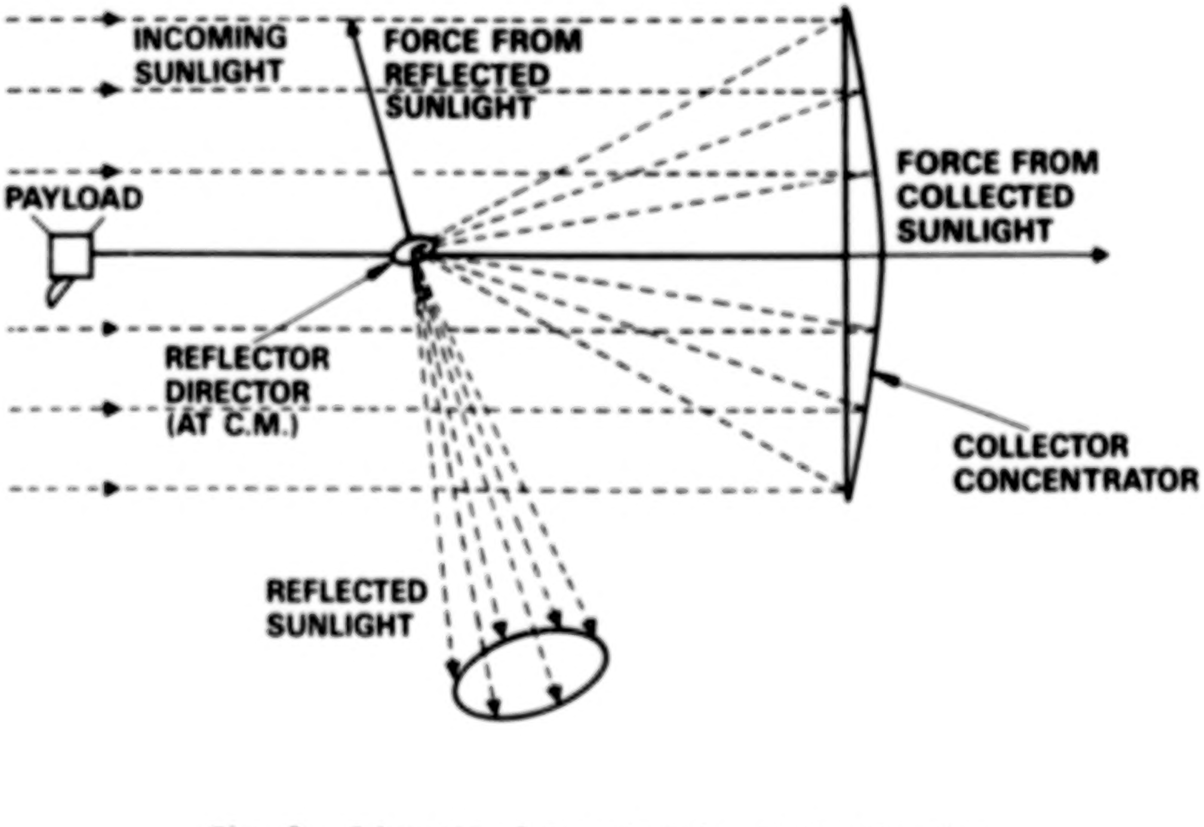

INCOMING SUNLIGHT
FORCE FROM REFLECTED SUNLIGHT
PAYLOAD
FORCE FROM COLLECTED SUNLIGHT
REFLECTOR DIRECTOR (AT C.M.)
COLLECTOR CONCENTRATOR
REFLECTED SUNLIGHT

Fig. 3 - Schematic of Generic Solar Photon Thrustor

Since the collector of the sunlight in the Solar Photon Thrustor is always facing the Sun no matter what the desired direction of thrust, the Solar Photon Thrustor always operates in a maximum solar light power collection mode. This is in contrast to a flat solar sail propulsion system where the collector and reflector are the same sheet of reflecting material. In a flat solar sail propulsion system, if the desired direction of thrust is not directly away from the Sun, the sail must be tilted at some angle $\theta$ with respect to the Sun-sail line. Since the sail is tilted toward the Sun, the effective collecting area of the flat solar sail propulsion system is decreased by an amount proportional to $\sin\theta$. This means that the Solar Photon Thrustor always collects more solar light power and therefore provides higher total solar photon radiation pressure force for the same area of collector. Since the mass of any optimized light pressure propulsion system is dominated by the mass of the light collecting area, that means that a Solar Photon Thrustor system will have better performance in terms of maximum payload capability, maximum propulsive thrust, and minimum mission time than flat solar sail propulsion systems. High solar power concentration numbers are not needed for the Solar Photon Thrustor. A concentration ratio of only 100:1 means that the area (and therefore the mass) of the reflecting optics will be 1% of the area (and mass) of the collecting optics and therefore a negligible portion of the total spacecraft mass.

The electromagnetic radiation does not have to stay in its original form. For example, the collector could collect sunlight and concentrate it on a solar cell or thermal boiler electrical generation system. The electricity generated could be used to make microwave, laser, or other useful coherent radiation, which would be beamed down to Earth. The waste heat from the process would be radiated away into space. Both the beamed coherent power and the radiated waste heat would produce propulsive force of comparable magnitude to the collected light. With proper system design, the beamed power and waste heat, along with the collected light, could provide all the propulsion needed.

Richard Moss, M.D. of Plymouth, Massachusetts has found (ref. 25) that a solar photon thrustor can be launched at shuttle altitudes. (Standard sails can only operate above 1000 km altitude, where the light pressure force exceeds the atmospheric drag.) If the solar photon thrustor is launched into a Sun-synchronous orbit over the terminator, the large collector sail facing the Sun will have minimum drag since it is flying edge-on to the residual atmosphere. It only takes four days to go from Shuttle altitudes to a safe 1000 km drag-free altitude.

## Solar Sails for Manned Missions to Mars

John Garvey of McDonnell-Douglas has been reexamining the use of solar sails for the manned exploration of Mars initiative. Prior studies by Carl Sauer of JPL resulted in optimized trip times of 824.5 days (2.25 years) for an Earth to Mars transfer. Garvey realized that most of that time was spent spiraling up from LEO to escape, matching velocities with Mars with a sail tilt angle that was almost edge-on to the Sun, and spiraling down from escape to LMO. By using a mixture of chemical boost on departure, solar sail propulsion during transfer, and aerobraking upon arrival, Garvey has found non-optimized mission profiles of 150 days one way, with even shorter return trip times for the empty sail. These short mission times cut the crew consumable weight drastically and eliminate the need for artificial gravity.

Garvey has also found a way to deploy a solar sail at Space Station altitudes, where astronauts can help solve deployment problems. The deployment is carried out at the end of a 100 km long upward-going tether, with the sail kept edge-on to the orbital motion to minimize drag. When the sail is released, it will rise upward in an elliptical orbit to where it can turn to the Sun and fly into space on its own power.

**Exotic Orbits With Solar Sails**

If a solar sail is made light enough, it can "hover" without orbiting--the light pressure from the solar photons balancing the gravity attraction of the Sun (and/or Earth). Colin McInnes of the University of Glasgow recently found (ref. 26) a large family of solar sail orbits around the Sun that produce nearly any desired orbital period (for example: zero--hovering anywhere over the Sun, moving heliosynchronously with features on the solar surface, or matching the orbital period of a planet) at nearly any desired orbital distance, in or out of the ecliptic plane. The light pressure from the Sun modifies the orbital equations so much that the orbital period is nearly independent of the orbital radius. For another example, James Early of Lawrence Livermore National Labs describes in reference 27 a large solar sail maintaining station between the Sun and the L2 point of the Earth. If the sail were 2000 km in diameter (made of lunar material), it would block enough sunlight (2%) to provide a technological solution to the greenhouse warming problem.

Robert Forward of Forward Unlimited has discovered (ref. 28) light-levitated geosynchronous orbits around the Earth that are at equilibrium positions north or south of the presently crowded equatorial geostationary orbit. The orbital radii of these light-levitated orbits are slightly less than the geostationary orbit radius, the center of the orbit is north (or south) of the center of the Earth, but the orbital rotation rate of the spacecraft matches that of the Earth's surface. Forward has also invented (ref. 29) a new kind of spacecraft that uses solar sails to assume non-orbiting equilibrium "polesitter" positions that allow communication, broadcast, or weather spacecraft to continuously hover over the polar regions of the Earth (or any other planet in the solar system). Since these spacecraft do not orbit, and therefore are not "satellites" of the Earth, the generic term of "statite" has been coined for them. Forward Unlimited has filed worldwide patents on the statite concept and is presently gathering private funding in order to fly a demonstration model.

To properly appreciate the statite concept, it is important to realize that all of the thousands of space objects presently in orbit around the Earth use the centrifugal force generated by their orbital motion to balance the Earth's gravitational force. By contrast, the statite is a space object that does not use centrifugal force from orbital motion about the Earth to counteract any significant portion of the Earth's gravitational force. Instead, the statite uses a solar sail propulsion system to maintain the statite and its payload in a desired fixed position adjacent to the Earth by balancing light pressure force against the Earth's gravitation force.

As shown in Figure 4, a space vehicle containing a Earth-services payload (broadcast, communications, weather, navigation, etc.) is attached to a solar light pressure propulsion system to form a space services station. After launch to an altitude where the light pressure propulsion system can function, the light pressure propulsion system is used to transfer the station to a point above the north or south hemisphere of the Earth where the gravitational pull of the Earth is counterbalanced by the light pressure force from the Sun.

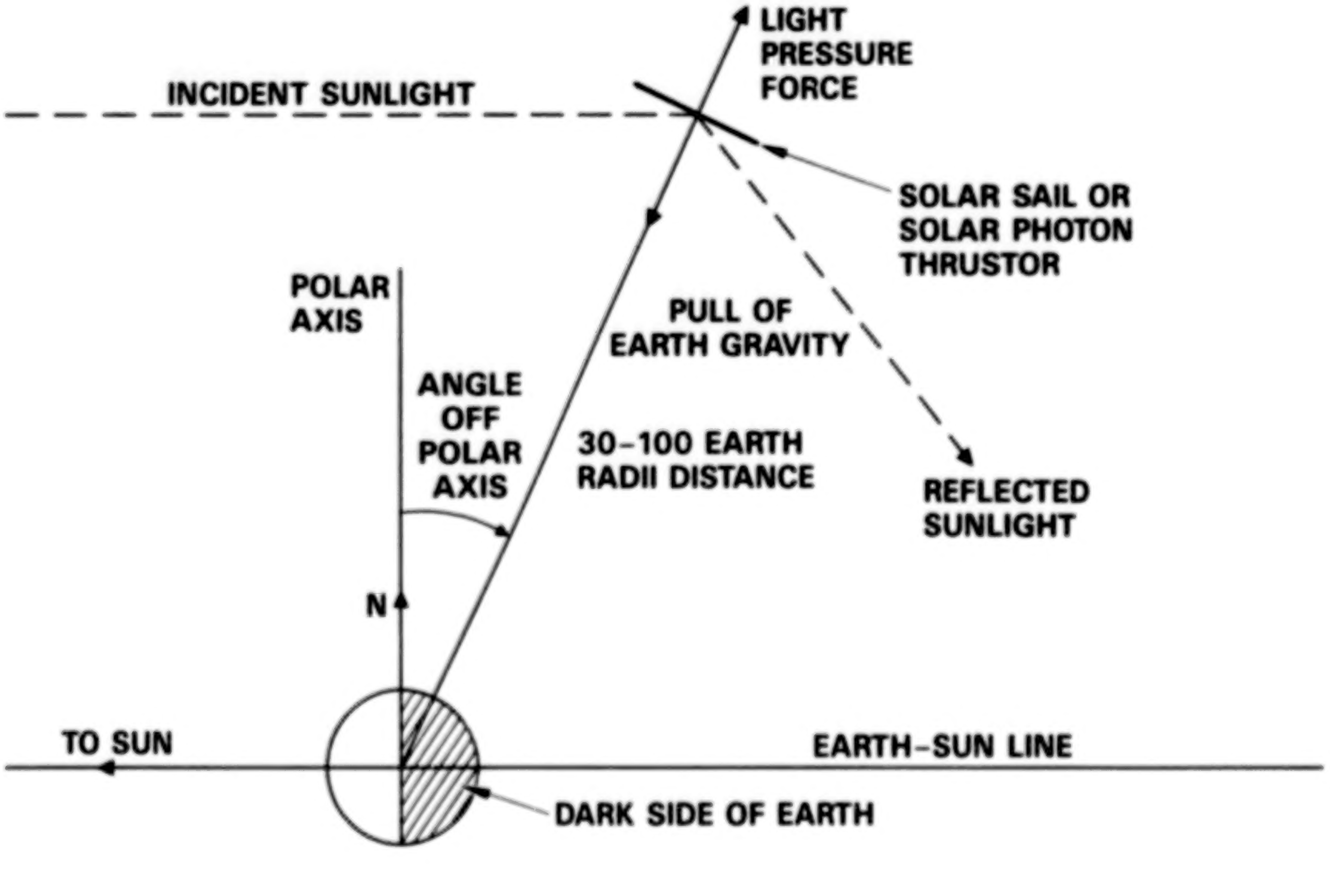

LIGHT PRESSURE FORCE
INCIDENT SUNLIGHT
SOLAR SAIL OR SOLAR PHOTON THRUSTOR
POLAR AXIS
PULL OF EARTH GRAVITY
ANGLE OFF POLAR AXIS
30-100 EARTH RADII DISTANCE
REFLECTED SUNLIGHT
N
TO SUN
EARTH-SUN LINE
DARK SIDE OF EARTH

Fig. 4 - Schematic of Generic Statite Concept

In most versions of the system, the statite is offset from the polar axis. The statite stays fixed at a point above the dark side of the Earth, while the Earth spins beneath it. The statite does not have to be positioned directly opposite from the Sun. It can be placed anywhere over a large area on the dark side of the Earth. This is in contrast to the single linear arc of the equatorial geostationary orbit. From the viewpoint of an observer on the rotating Earth, this version of the statite rotates around the pole once every 24 hours (a solar day). Thus, ground stations for communication with these statites must have their antennas on a polar mount with a 24 hour clock drive. Since the distance between the ground station and the statite does not change significantly and the doppler shifts are very low, the electronics needed for these versions of the system are nearly as simple as those at the fixed position ground stations. There is an alternate version of the statite system where the statite is kept directly over the North or South Pole of the spinning Earth. To an observer on the Earth, the statite stays fixed above the pole while the stars rotate around it. In these versions, the ground stations can used fixed mounted antennas and simple fixed gain, fixed frequency electronics.

A typical distance of a statite from the center of the Earth is 30 to 100 Earth radii. The better the performance of the sail, the closer the balance point. (For reference, geostationary orbit is at 6.6 Earth radii and the Moon is at 63 Earth radii.) The round-trip delay time for 100 Earth radii is 4 seconds, making the statite more suitable for direct broadcast, fax, data, and weather services than two-way telephone conversations. The advantages of the statite concept are: it provides continuous service to a region using a single spacecraft without requiring a slot on the already crowded equatorial geostationary orbit, and it provides continuous coverage to regions of the Earth that are too close to the poles to use equatorial geostationary orbit satellites. The disadvantages of the statite concept are: constant control is required to maintain station, the round-trip link time is in seconds, and in most versions the ground station antenna must rotate once a day.

## MAGNETIC SAILS

Magnetic sails or "magsails" are a derivative form of solar sails that use a completely different type of physical interaction with the Sun than solar light pressure sails (ref. 30). Invented by Dana Andrews of Boeing Aerospace and Robert Zubrin of Martin-Marietta Denver, a magsail is a simple loop of high-temperature superconducting wire carrying a persistent current. The charged particles in the solar wind are deflected by the magnetic field, producing thrust. Although the thrust density in the solar ion wind flux is five thousand times less than the thrust density in the solar photon flux, the mass of a solar sail goes directly as the area, while the mass of the magsail goes as the perimeter of the area enclosed. In addition, the effective cross-sectional area of the magnetic fields around the magsail is about a hundred times the physical area of the loop. As a result, preliminary calculations show the thrust-to-weight of a magsail can be an order of magnitude better than a solar sail. Recent analyses indicate that a properly sun-shielded cable can be passively maintained at a temperature of 65 K in space, well below the superconducting transition point for many of the new high-temperature superconductors.

## TETHERS

Tether propulsion a technology that will fly soon. NASA is funding Martin Marietta to build the tether (2.5 mm diameter and 100 km long) and deployment mechanism, while Italy is building the spacecraft that will fly at the end of the tether. The first experiment, scheduled for 1991, will deploy the spacecraft upward from the Shuttle on a conducting tether cable to demonstrate power generation from the motion of the conducting cable through the Earth's magnetic field. By pumping current through the cable, thrust would be generated by the "push" of the cable against the Earth's magnetic field. The second flight will deploy an atmospheric research spacecraft downward, where it will fly through the upper atmosphere, too low for spacecraft and too high for aircraft. The tether connection to the Shuttle spacecraft provides the propulsion needed to overcome the drag. Ivan Bekey, formerly at NASA Headquarters and now on the National Space Council, has been championing the use of tethers for many space applications (ref. 31 and 32), including throwing payloads from LEO to GEO, electromagnetic propulsion using a conductive tether, and momentum transfer through the Space Station. In the latter application, an Orbital Transfer Vehicle is launched from an upward going tether at the same time as the Space Shuttle deorbits from a down-going tether, all without using any fuel. The Space Station is unaffected—it merely transfers energy and momentum between the two vehicles. Paul Penzo of JPL has shown (ref. 33) it is possible to use tethers to move payloads from planetary body to planetary body (see Fig. 5), such as low Martian orbit to low Earth orbit.

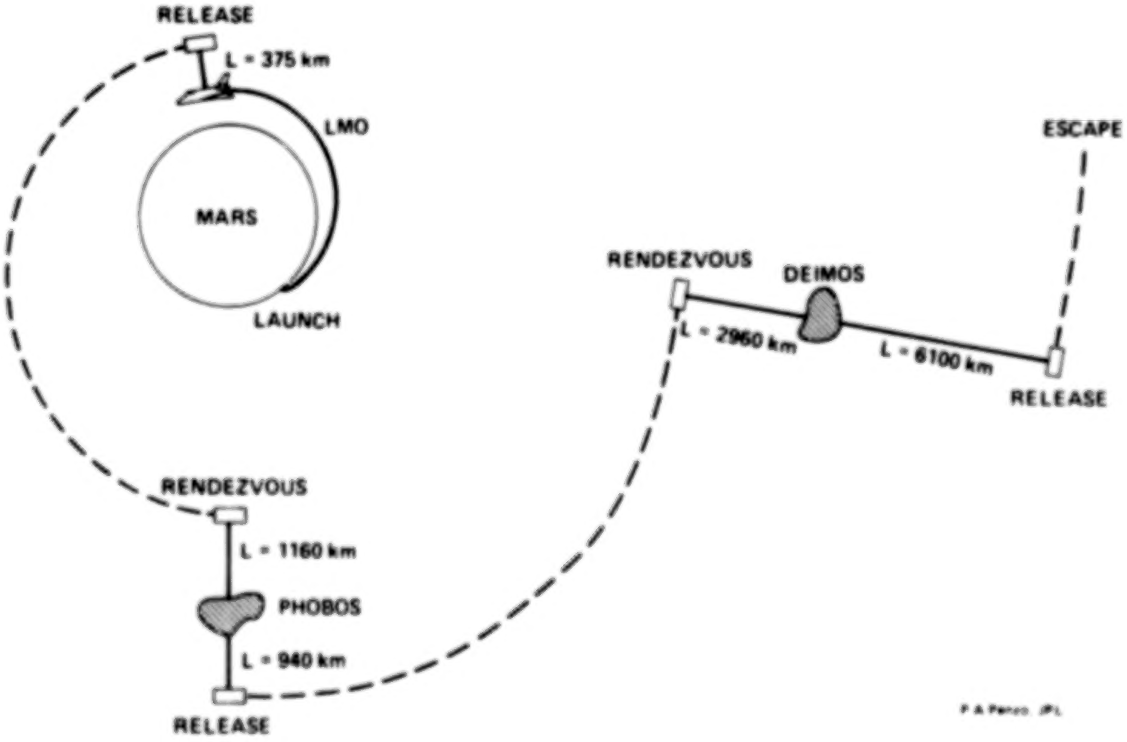


Fig. 5 - Schematic of Generic Mars Tether Transport System

Hans Moravec of CMU has shown in reference 34 that a long rotating Kevlar tether in orbit around the moon or small airless planet (like the Moon or Mars) can touch down to the surface six times an orbit, simultaneously dropping off and lifting up payloads weighing a reasonable fraction of the tether mass. This concept is being reevaluated by Joseph Carroll of Tether Applications for its potential impact on the Lunar Base initiative. Using the tether material Spectra, which has improved properties over the more familiar Kevlar material, Carroll has done a preliminary design on an ambitious tether transport node facility designed to provide a 1 km/s △V to 10 ton payloads. To stay in orbit, a typical facility mass should be at least 300 tons for 10 ton payloads, but the 300 km long tether itself would only mass 7 tons. One tether facility would be placed in a circular 400 km orbit and another in a highly elliptical orbit with a 4:1 period resonance. As shown in Figure 6, payloads would be picked up from a 150 km or lower earth orbit by the lower facility and tossed into an intermediate elliptical orbit with an orbital period twice the lower facility and half the upper facility. There the payloads would be picked up by the higher facility and tossed to the Moon. At the Moon, the payloads would be retrieved by a 200 ton, 1160 km diameter rotating tether and deposited on the surface of the Moon. By arranging things so an equal amount of mass flows in both directions, this system is self-powered. Bags of lunar dirt flowing down the tether system into the deep Earth gravity well will be the "fuel" needed to move payloads from LEO to the surface of the Moon.

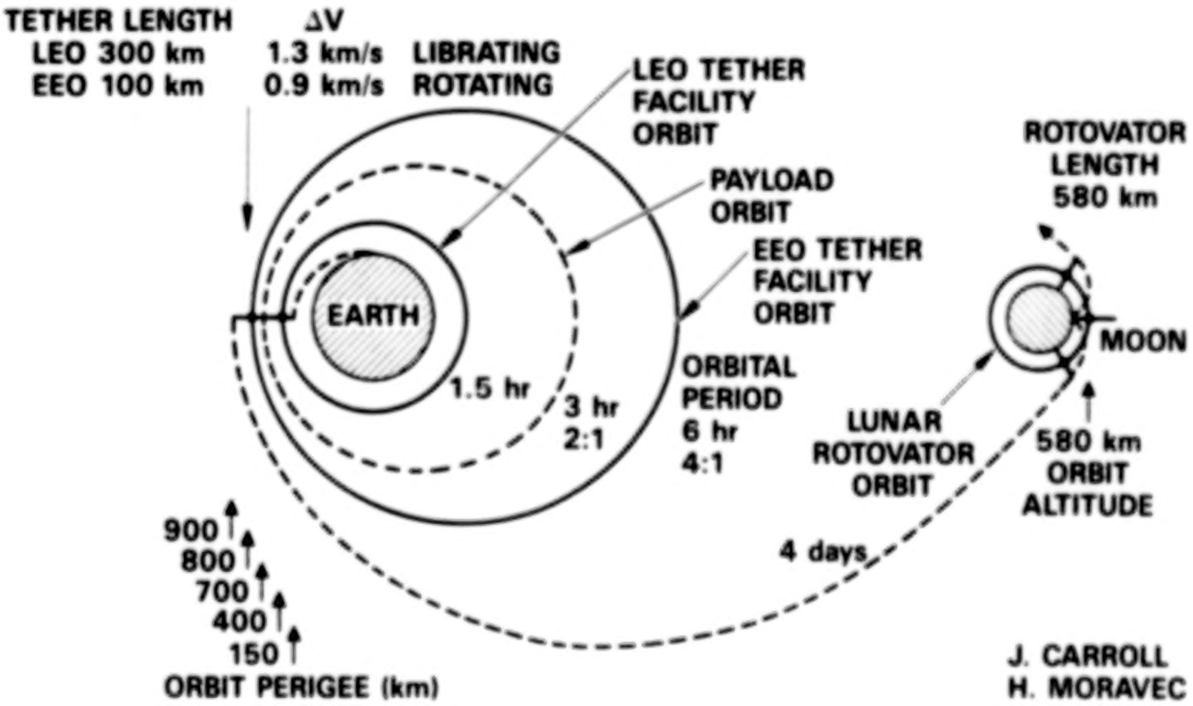
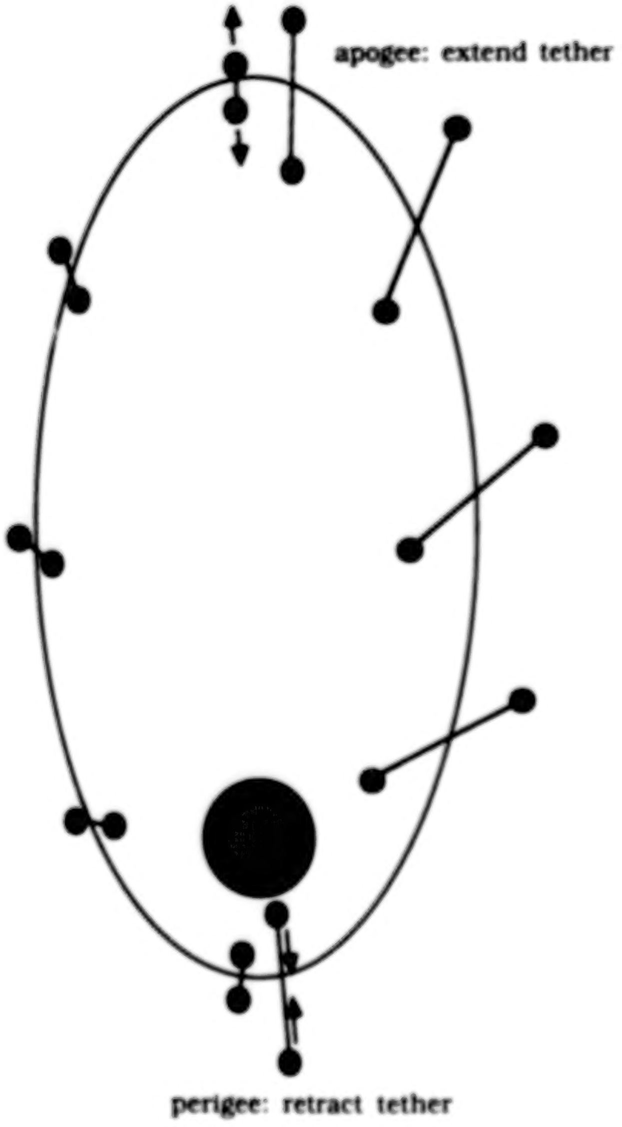

TETHER LENGTH
LEO 300 km
EEO 100 km
ΔV
1.3 km/s
0.9 km/s
LIBRATING
ROTATING
LEO TETHER FACILITY ORBIT
PAYLOAD ORBIT
EEO TETHER FACILITY ORBIT
ROTOVATOR LENGTH 580 km
EARTH
MOON
1.5 hr
3 hr 2:1
ORBITAL PERIOD 6 hr 4:1
LUNAR ROTOVATOR ORBIT
580 km ORBIT ALTITUDE
4 days
900
800
700
400
150
ORBIT PERIGEE (km)
J. CARROLL
H. MORAVEC

Fig. 6 - Schematic of Generic LEO-Lunar Surface Tether Transport System

**Tether "Bootstrap" Propulsion**
Geoffrey Landis at NASA/Lewis has shown in reference 35, how a spacecraft starting in a low circular orbit about Earth can use a power supply and a long tether "pushing" against the Earth's gravity gradient field to "bootstrap" itself (and the tether) up the gravity well nearly to escape in less than a month without using propellant. The basic concept is based on the fact that if two halves of a spacecraft (or a spacecraft and its expended booster) are extended on a long tether, the center-of-mass of the extended system shifts slightly downward from the original center-of-mass and the orbital period decreases. This shift in the center-of-mass occurs because the Earth's gravity force causes an acceleration on the masses that varies as $1/r^2$, while the counteracting centrifugal force due to orbital motion causes an acceleration that varies as r. For very long tethers, the two forces no longer exactly cancel at the two ends and there is a residual, second order, force which must be balanced by a shift in the center of mass. When the tether is pulled in again, the center-of-mass of the combined system raises upward.

As shown in Figure 7, by alternately extending and contracting the tether at proper points in the orbit, the tether can be used to "pump" an initially circular orbit into a highly elliptical orbit. Theoretically, if the initial orbit is circular and at an altitude of greater than one earth radii (orbital radius of greater than two earth radii or greater than 13,000 km), then the final orbit can be an escape parabola. Note that the angular momentum of the initial and final orbits are the same, so no angular momentum needs to be supplied. The energy of the escape parabola is much greater than the energy of the initial circular orbit, so energy needs to be supplied, either from an onboard power supply or by collecting externally supplied power. The final configuration has the payload, tether, and counterweight flying off away from the Earth at some residual velocity, so it has some linear momentum. To conserve linear momentum, the tether has transferred linear momentum to the Earth by coupling to the gravity tidal fields of the Earth through its extended length. Although it looks like the system is "pulling itself up by its bootstraps", it is not. In effect, the tether is "climbing" out of the Earth's gravity well by coupling to the nonlinearities in the gravitational gradient fields or gravity tides.

Unlike other tether propulsion concepts in the literature, where one mass (the payload) is raised in orbit while another mass (the counterweight) is lowered in orbit, the technique developed by Landis allows the center-of-mass of the entire system to be raised from a low circular orbit into a high elliptical orbit--conceptually into an escape orbit from Earth--without the use of rockets or reaction mass. Energy is required, which can be supplied from an onboard power supply, but no reaction mass is needed, and if the Earth-to-LEO booster is used as a counterweight for the payload mass, the only weight penalty is the mass of the tether (compared with the weight penalty of a LEO-GEO booster rocket).



apogee: extend tether
perigee: retract tether

Fig. 7 - Schematic of Generic Tether "Bootstrap" Propulsion Concept

**Cable Catapult**
A cable catapult is a new type of propulsion system proposed by Forward that uses a long tether as a launch rail (ref. 36). As shown in Figure 8, the tether cable is pointed in the desired direction of travel. A payload is attached to a linear motor capable of traveling along the cable. The linear motor accelerates along the cable until the payload reaches the desired launch velocity, at which point the payload is released. The linear motor then decelerates to a halt to await the arrival of an incoming payload.

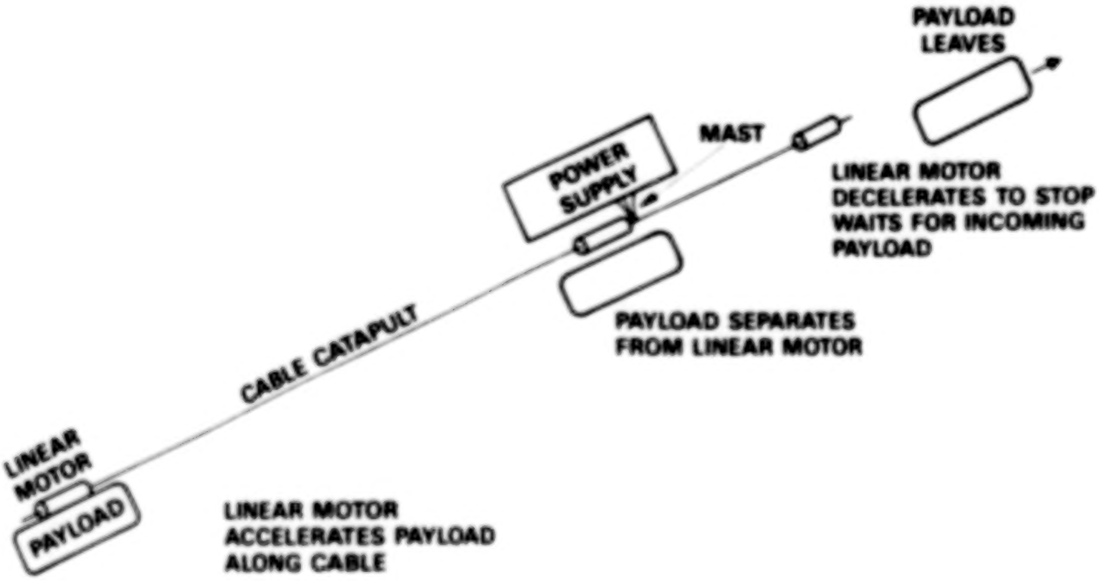

PAYLOAD
LEAVES
MAST
POWER
SUPPLY
LINEAR MOTOR
DECELERATES TO STOP
WAITS FOR INCOMING
PAYLOAD
PAYLOAD SEPARATES
FROM LINEAR MOTOR
CABLE CATAPULT
LINEAR
MOTOR
PAYLOAD
LINEAR MOTOR
ACCELERATES PAYLOAD
ALONG CABLE

Fig. 8 - Schematic of Generic Cable Catapult Concept

In the past, tethers have been considered for transporting payloads to and from the Moon, Mars, and other bodies in the solar system. These tether propulsion systems usually involved swinging or rotating tethers. Moravec has shown that the maximum tip speed V of a rotating tether (and therefore the maximum speed at which a rotating tether can launch a payload) is a function of the "characteristic velocity" of the cable given by the square root of the ratio of the design tensile strength T to the density d of the material in the tether or $v=(T/d)^{1/2}$ and the ratio of the mass M of the tether to the mass m of the payload. The exact expression is:

$$\frac{M}{m} = (2\pi)^{1/2} \frac{V}{v} e^{V^2/2v^2} \mathrm{erf}(V/2v)$$

where erf is the error function (typically of order unity or less). This can be compared to the ratio of the rocket mass to the payload mass of a rocket where v is the "exhaust velocity" of the fuel.

$$\frac{M}{m} = e^{V/v} - 1$$

Which is exponential in V/v, while the rotating cable mass ratio is exponential in the square of V/v. In contrast, Forward has shown in reference 36, that the ratio of the tether mass to the payload mass used in the cable catapult mode varies as:

$$\frac{M}{m} = V^2/2v^2 \quad .$$

Because of the squared exponential growth of the mass of the tether in a rotating tether system, the maximum launch velocity attainable for practical launcher to payload mass ratios is three times the characteristic velocity of the cable material or 3 km/s for a 1 km/s Kevlar cable. A cable catapult using the same amount of cable material could give the payload a launch velocity of 30 times the cable characteristic velocity or 30 km/s. Improved cable materials having higher characteristic velocities will allow interplanetary travel at 30-100 km/s. This could shorten trip times to Mars from years to months.

## FAR FUTURE PROPULSION

Even more exotic propulsion concepts abound in the literature. Many advanced nuclear propulsion concepts have been proposed that depend upon some exotic physical process being found practical. For one example, George Chapline of Lawrence Livermore National Lab has proposed a fission fragment rocket using thousands of kilometers of americium coated fibers suspended on dozens of rotating 100-meter-sized wheels as a combination fuel source and heat radiator. Others have examined the propulsion applications of various potential techniques for catalyzed cold fusion, using palladium, muons, fractional charges, magnetic monopoles, and strange matter. None of these fusion techniques look promising for propulsion, primarily since in most cases the energy output is in the form of high energy neutrons, which are difficult to turn into thrust except through an indirect thermalization process.

We do not lack new ideas to explore: some examples are studies on laser and microwave pushed sails to the planets and stars (ref. 37 and 38), and extracting laser power from the mesospheres of Mars, Venus, and maybe Earth (ref. 4). Even further out are recent papers on negative matter propulsion (ref. 39), space warps (ref. 40), and serious-but-skeptical studies of Biefield-Brown field propulsion and electrogravity induction field theories (ref. 22).

## SUMMARY POLEMIC

In this review I have discussed a number of exotic power and propulsion techniques, ranging from eminently feasible to the wildly impossible. But it is important for you, the reader, to realize that my main message is that we don't need to wait for truly exotic technologies like metallic hydrogen, antimatter, or space warps to improve the nation's space propulsion capabilities by orders of magnitude increase in performance and orders of magnitude decrease in cost. Chemical rocket propulsion is fine when the $\Delta V$ is small, but for the more ambitious missions, this nation needs to put substantial development funds into making real those advanced space propulsion technologies that have already shown their potential value in decade after decade of paper studies.

Solar and nuclear electric propulsion should come first, not small systems for secondary tasks like North-South station keeping or Space Station drag makeup, but large megawatt and multimegawatt primary propulsion systems for OTV tugs, Earth-Lunar shuttles, and manned missions to Mars. Then solar sails, first for communication, broadcast, and especially weather satellites that are not limited to the equatorial geostationary orbital arc, second for scientific monitoring stations hovering over the Sun, planets, and moons of the solar system, and third for hauling cargo to and from Earth, the planets, and the asteroid belt--without the expenditure of fuel.

Next should come rotovators made of long rotating Kevlar tethers that will allow transport of massive quantities of material to and from low orbit to the surface of planetary bodies such as the Moon, Mars, Mercury, and most of the moons in the solar system--again without the use of fuel. Rotating tethers around the Earth could also move massive amounts of material from LEO to GEO or escape--using no fuel in the process as long as the amount of material being brought down the gravity well of Earth exceeds the amount being hauled up.

To get off the Earth and into LEO, we must either bite the political bullet and push high-thrust hot hydrogen exhaust nuclear thermal rockets with their radiation hazard, or stick with chemical rockets and their greenhouse hazard. High thrust laser propulsion, either pulsed or CW, is an alternate choice with its own set of operational and environmental problems that need engineering demonstration, not another mile-high stack of paper studies.

Mission planners must use what they know works in order to plan a mission. If future missions, such as a return to the Moon, or the manned exploration of Mars, are to be made economically feasible, NASA needs to stop the interminable paper studies and move into the development and demonstration of advanced forms of space propulsion such as nuclear, electric, lightsail, tether, and laser propulsion. That way, those mission planners will have some viable alternatives to work with. Otherwise, this nation is going nowhere in space.

## ACKNOWLEDGMENTS

This research was supported in part by NASA/Lewis Research Center and in part by the Air Force Astronautics Laboratory through Contract F04611-87-C-0029 with Forward Unlimited.

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N91-22141

# INDUSTRIALIZING THE NEAR-EARTH ASTEROIDS: Speculations on Human Activities in Space in the Latter Half of the 21st-Century

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Presented at the Vision 21 Symposium
April 3-4, 1990
NASA Lewis Research Center, Cleveland, Ohio

# ABSTRACT

The use of solar system resources for human industry can be viewed as a natural extension of the continual growth of our species' habitat. Motivations for human activities in space can be discussed in terms of five distinct areas: i) information processing and collection, ii) materials processing, iii) energy production to meet terrestrial power needs, iv) the use of extraterrestrial materials, and v) disaster avoidance. When considering 21st-century activities in space, each of these basic motivations must be treated in the light of issues likely to be relevant to the 21st-century Earth. Many of the problems facing 21st-century Earth may stem from the need to maintain the world population of 8 to 10 billion people as is projected from expected growth rates. These problems are likely to include managing the impact of industrial processes on the terrestrial biosphere while providing adequate energy production and material goods for the growing population.

The most important human activities in space in the latter half of the 21st-century may be associated with harnessing the resources of the near-Earth asteroids for industrial processes. The near-Earth asteroids are estimated to contain approximately $10^{17}$ kg of material, enough to have a profound effect on terrestrial industrial processes. However, if this material is to be of use, challenges associated with the high cost of access to space, the hazards of the space environment, as well as other difficulties associated with any new enterprise, such as obtaining required capital investments, legal issues, and national policy issues must be overcome. The functions a space industry must accomplish include raw material collection, processing raw materials into useful feedstocks, parts fabrication, product assembly, and transport of products to Earth. Dramatic advancements in the technologies of power systems, sensors, and command and control systems will obviously be needed.

If the needed technological advancements are made, an important option to consider for harnessing space resources is that of replicating manufacturing systems. A key aspect of these systems is closure. Three types of closure have been identified as important: energy closure, information closure, and matter closure. If perfect closure in each of these areas could be achieved, the result would be a von Neumann* machine. Although the prospects of producing a true von Neumann machine are not good, replicating systems with imperfect closure can be made to effectively support space industrialization. The benefit of the exponential growth of replicating systems can be significant. Calculations suggest that within a few decades of the initial deployment of a replicating space factory, the total capacity of a space industry could exceed that of the Earth. Initially, products manufactured in space will be used in space, but eventually it will be cost effective to import goods to the Earth. Given the magnitude of the potential benefit of such a full-scale space industry, development of advanced automation technology and extraterrestrial materials processing systems should be vigorously pursued.

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* The term *von Neumann machine* is used to describe a class of self-replicating automata which exhibit complete closure. John von Neumann is generally credited with having conducted the first rigorous mathematical treatment of self-replicating machines in the 1950s.

# THE HUMAN DIASPORA AND MOTIVATIONS FOR ACTIVITIES IN SPACE

The stated purpose of the of the Vision 21 Symposium is to foster innovative thinking about space activities for the next millennium. To this end, I will talk about possibilities that will come after the Space Exploration Initiative, which NASA is now studying as the focus of our space program for the first few decades of the 21st-century. I will address space activities in which our species makes the transition from exploration to industrialization of the solar system. In particular, I will focus on how people of the 21st-century may use the near-Earth asteroids.

Before beginning this discussion of solar system industrialization, it will be useful to first describe what I refer to as the "Human Diaspora." The term is used here to refer to the process whereby human beings extend their habitat from some limited region into another, larger region. The Human Diaspora extends along a continuum in which one can identify four stages. The first stage is Exploring, whereby human beings identify and visit new habitats. The second stage is Prospecting, in which we survey new habitats for sources of wealth or other benefits. After the Prospecting stage comes Pioneering, during which initial attempts to harvest benefits from the new habitats are made. The final stage of the Human Diaspora is Settling and/or Industrializing, in which the potential benefits of new habitats are fully exploited. The word *diaspora* has the same root origins as the word *dispersion*, but it can mean a dispersion of any originally homogeneous people. To my knowledge the first person to apply the term diaspora to the human settlement of space was Robert Heinlein in his now classic-science fiction stories about the exploits of Lazarus Long.[1]

The Human Diaspora has been an on-going process which started prehistorically and continues today as human beings move into ever expanding habitats. In recent history, however, one observes an interesting new feature of the process: With technology, human beings are no longer required to physically travel into new habitats to accomplish the four stages of the Human Diaspora. For example, robotic spacecraft sent to the outer solar system are presently engaging in completely robotic exploration. I suggest that this new trend - in which we let our technology do the traveling for us - is an important new wrinkle, and one we should consider carefully in planning future human activities in space.

The concept of the Human Diaspora is presented here to provide a backdrop against which we can discuss human development of the solar system. Also useful is a classification of the basic motivations for human activities in space. For the purpose of this paper, I have classified five basic motivations: Information Processing and Collection, Materials Processing, Energy Production, Use of Extraterrestrial Materials, and Disaster Avoidance. Table 1 gives examples of space projects - either proposed, in progress, or completed - associated with each of these motivations.

Table 1 - Motivations for Human Activities in Space
and The Stage of Each in the Human Diaspora

| MOTIVATION | EXAMPLES | PRESENT STAGE |
|---|---|---|
| Information Processing and Collection | Communications, Science, and Earth Observations | Settling and Industrializing |
| Material Processing | Pharmaceuticals and Electronics | Pioneering |
| Energy Production for Terrestrial Use | Solar Power Satellites, Space Disposal of Nuclear Waste, Lunar He$^3$ | Prospecting |
| Use of Extraterrestrial Materials | Mining Common Materials and Rare Metals, Manufacturing Goods | Prospecting |
| Disaster Avoidance | Asteroid Deflection, Weather Modification, Climate Control | Exploring and Prospecting |

Table 1, under the motivation "Information Processing," lists communications, Earth observations, and science. In these areas, we are presently well into the industrialization stage. Under "Materials Processing" are included the experiments that are frequently conducted on the space shuttle or on the Soviet Mir space station. In these experiments, the unique features of the orbital environment are used to develop new pharmaceuticals and semiconductors. While the products of these experiments are valuable, it is likely that much more benefit is yet to be realized in the area of space-materials processing. As such, the level of development associated with present work in space materials processing is still in what I refer to as the Prospecting stage of the Human Diaspora.

Activities which fall under the motivation "Energy Production" for net energy benefit here on the Earth are still in the early Exploration stage. An example of the exploratory thinking that has been done in space energy production is the work of Peter Glaser who first proposed and then studied the concept of the space solar-power satellite in the 1960s and the 1970s.[2] Gerard O'Neill, who proposed the human colonization of space habitats, suggested space solar-power satellites as a justification for human colonies in space, with the space colonists working to manufacture power satellites.[3]

Another concept that may be worth considering more seriously in the near future is that of space disposal of nuclear waste.[4] If nuclear power continues to be used on the Earth, safe places to store nuclear waste products will continue to be required. Many argue that the surface of the Earth is not a good place for such storage. If we can develop a sufficiently reliable space launch system and find ways to ensure that nuclear waste can be safely launched into space, the possibility of nuclear waste disposal in space will have merit.

Mining lunar $He^3$ is an idea first proposed by a group from the University of Wisconsin. Mining lunar $He^3$ may be of interest in the 21st-century because $He^3$ produces less neutron radiation (when reacted with deuterium) then other proposed fuels for nuclear fusion. The problem with $He^3$ is that it is very difficult to obtain on the surface of the Earth. The Moon, however, possesses an abundant source of $He^3$, which is there because solar wind continually deposits it into the top layers of lunar regolith. The University of Wisconsin group and others have suggested there may be benefits associated with extracting $He^3$ from lunar regolith for transport to Earth, where it may be useful for electric power production.

Space activities motivated by "Use of Extraterrestrial Materials," include mining of common materials from extraterrestrial sources, mining rare metals for return to Earth, and manufacturing goods in space from in-situ materials. Examples of common materials which might be justifiably mined from extraterrestrial surfaces include oxygen for astronauts to breath or hydrogen and oxygen for rocket fuel. The motivation for mining common materials from extraterrestrial sources would be to save the cost of launching these materials up from the surface of the Earth.

Due to the large economic value of the rare metals that can be expected to exist in near-Earth asteroids, it may be cost-effective to mine asteroids for the purpose of returning rare metals to the Earth.[6] There may also be economic advantages to manufacturing goods in space using extraterrestrial materials. Such space-manufactured goods could include satellites used in orbit or high-value products transported to the surface of the Earth. Peter Glaser's space solar-power satellite concept is an example of a product that could be manufactured from extraterrestrial materials and used in space.

The final motivation I have identified for human activities in space is "Disaster Avoidance." I've listed asteroid deflection and weather modification as two types of space activities which might fall under the heading of disaster avoidance. Both asteroid deflection and weather modification are still in the early exploration phase.

The concept of asteroid deflection is to detect asteroids with trajectories intersecting that of the Earth and to then change the course of the asteroid enough to prevent a collision with the Earth. It would be possible to change to course an asteroid's course with any of a variety of propulsion devices now under development. Interest in asteroid deflection is motivated by the fact that significant asteroid bombardment of the Earth is known to take place on a continual basis.[7] On an average of about once a year, an object with kinetic energy equivalent to a small nuclear weapon strikes the Earth. If such an impact were to take place near a city, the results would be catastrophic. Larger impacts, such as the 1908 Tunguska impact in Siberia, which leveled hundreds of square miles of forest, have about a ten percent chance of occurring each century. The most devastating type of impact, such as the one which might have caused the extinction of the dinosaurs, is thought to occur very infrequently, at a rate of once every few hundred million years.

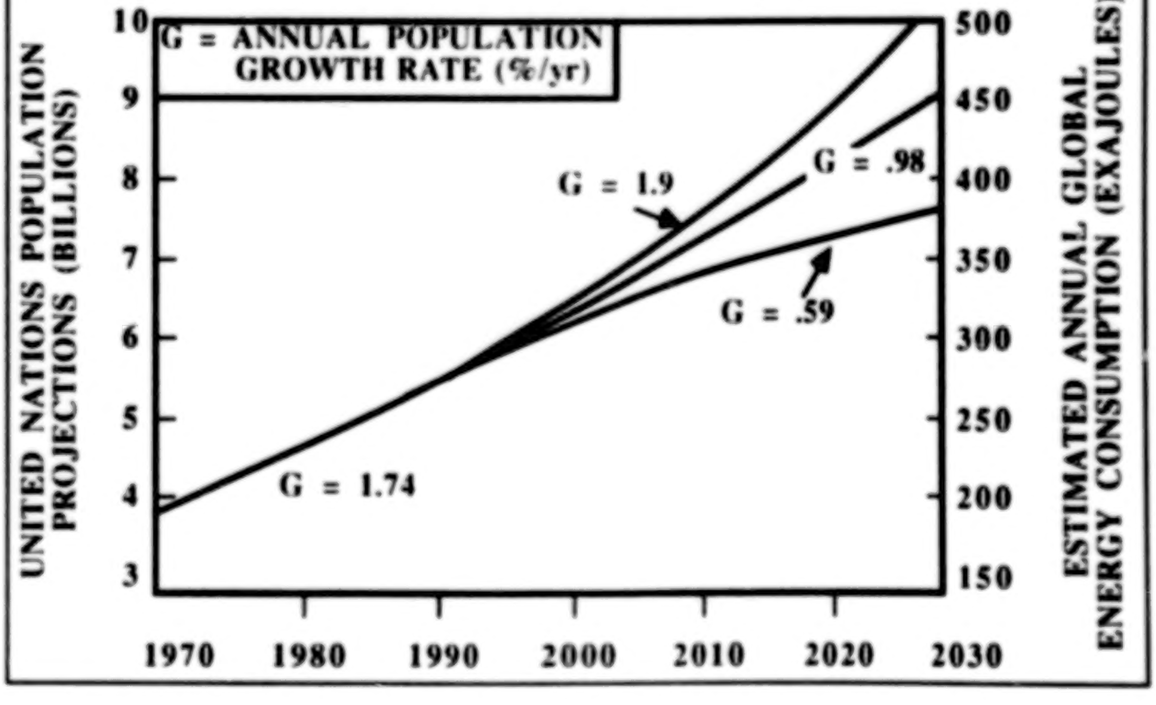
Weather modification is the next example of a space activity which might be motivated by a wish to avoid disasters here on the Earth. It may be worth considering the use of many multi-kilometer sized, thin-film reflectors located in orbit to direct sunlight or cast shadows on parts of the Earth. Potential applications include nighttime illumination of cities, heating agricultural regions in times of unusual cold conditions, and shading forming hurricanes to reduce their intensity. Little has been done with this concept, but it may merit further investigation as a space activity for the 21-Century.

The next natural step beyond weather control is climate control. Climatologists are beginning to show that human activities are affecting the climate of the Earth. One example is the production of greenhouse gases that may raise the temperature of the atmosphere. A natural question to ask is, "If we can deleteriously affect the climate by inadvertent action, why not affect the Earth's climate in a controlled fashion?" Space might be the best place from which to do controlled (as opposed to uncontrolled) climate modification. I would speculate that it may be worth considering using orbital mirrors or shades to deflect sunlight to affect changes in the total flux of solar energy reaching the Earth, and hence the Earth's climate.

## THE LIKELY CONDITION OF 21st-CENTURY EARTH

We presently inhabit a globe with a population of a more than 5 billion people. Our motivations for activities in space in the 21st-century depend very significantly on the conditions of the 21st-century Earth. Figure 1 gives United Nations population projections in billions of people as a function of year for the years 1970 to 2030.[8] This projection is based on the assumption that the population growth rate will remain near the current 1.74 percent per year until about the year 2000, and will fall thereafter to slightly less than 1 percent in the year 2025 due to a variety of demographic effects. Based on this projection, the world population will reach almost 8.5 billion by the year 2025. If the growth rate declines at a faster rate and reaches .59 percent by the year 2025, the population in 2025 will be about 7.6 billion. On the other hand, if the growth rate climbs to 1.9 percent at the end of the century before beginning a decline, the population in the year 2025 would be more than 9.4 billion people.

Figure 1- Projected Population Growth and Estimated Annual Energy Consumption



Also included in Figure 1 is a simplistic projection of global annual energy consumption based on the assumption that the globally averaged per capita energy consumption will remain constant. The units used are exojoules. One exojoule is $10^{18}$ joules. The consumption of one exojoule per year corresponds to a power level of about 32 gigawatts. The rate of world energy consumption in 1990 is about 300 exojoules per year, or about 10,000 gigawatts per year, most of which is consumed by the industrialized countries. According to John Givens, Peter Blair, and Holly Glen of the Congressional Office of Technology Assessment, less-developed countries consume 4 to 7 times less energy per person than do industrialized countries.[9] As less-developed countries industrialize, it is expected that they will start to use energy at a per-capita rate comparable to the rate at which we use energy in the industrialized world. The industrialized world has a net population growth rate of only about .4 percent compared to the 1.74 percent global average growth rate. Based on these trends, globally averaged per-capita energy consumption can be expected to increase in the next thirty to fifty years. Figure 1 is therefore optimistic, and the projected increase in annual global energy consumption to about 450 exojoules by the year 2030 may be quite low.

Many of the problems facing 21st-century Earth may stem from the need to maintain a world population of 8 to 10 billion people. These problems are likely to include managing the impact of industrial processes on the terrestrial biosphere, while providing adequate energy production and material goods for the growing population. An important resource for meeting these needs can be the near-Earth asteroids.

## NEAR-EARTH ASTEROID RESOURCES

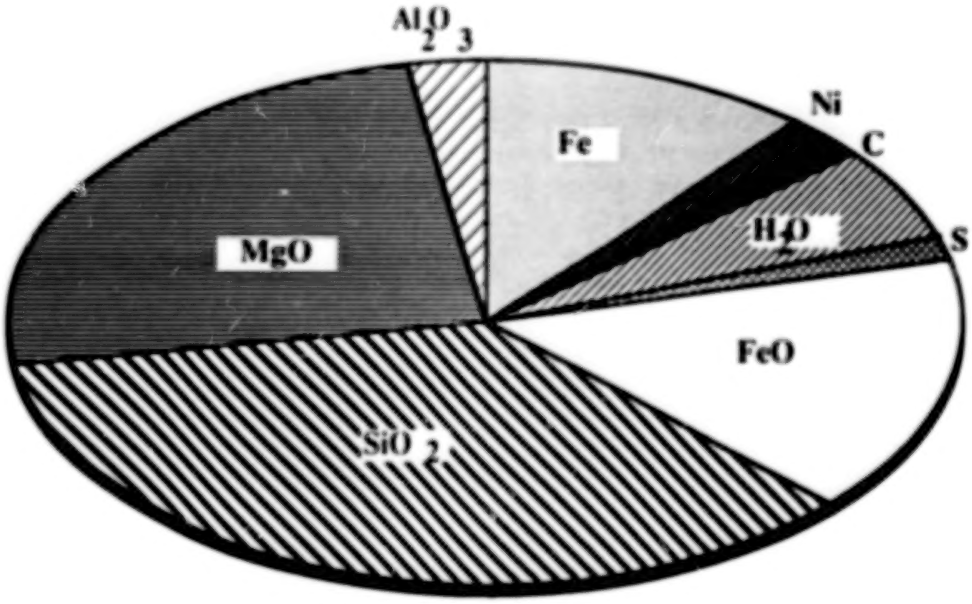
The near-Earth asteroids comprise about one part in ten thousand of all the mass of the asteroid belt, or about $10^{17}$ kilograms of material. The near-Earth asteroids are those asteroids with average orbital radii close to one astronomical unit, or whose orbits cross the orbit of the Earth. These are of a special interest to potential space industrialists because the energy required to move material from the near-Earth asteroids to Earth orbit, or to aerobrake in the Earth's atmosphere, is relatively low, corresponding to only a few kilometers per second.

The quantity of resources represented by the near-Earth asteroids is enormous.[10] For example, if all of the near-Earth asteroid material could be used to build O'Neill-type space colonies, it would be enough material to build habitats with a total living space roughly equal to that of North America. The near-Earth asteroids contain enough iron to produce the equivalent of a new car for every person in the solar system each year for a 1000 years based on an assumed total population of 10 billion people. Finally, the near-Earth asteroids contain almost a million times the raw material needed to build space solar power for satellites to meet all of the terrestrial power needs projected in Figure 1.

Another interesting observation is that if one were to separate out the precious metals in the near-Earth asteroids, one could extract a few million dollars worth of platinum group metals for each person in the solar system (at today's prices). Of course, this is just an illustrative example. It is difficult to imagine a need for such large quantity of precious metals Just by making such a large quantity of precious metal available would render it no-longer precious.

Figure 2 is a pie chart depicting some of the expected major constituents of one type of asteroid likely to be present in the near-Earth asteroids. Some important trace materials such as the platinum group metals are not shown in this figure because they represent only a small fraction of the total mass. The type of asteroid addressed in Figure 2 is a metal-rich carbonaceous asteroid. Many other types of asteroids can be expected to be found in the near-Earth groups. As the Figure shows, aluminum, iron, carbon, water, and other useful substances are expected to be present.

Figure 2 - Likely Composition of a Typical Metal-Rich Carbonaceous Asteroid



## CHALLENGES OF SPACE INDUSTRIALIZATION

Based on the quantity and composition of the materials in the near-Earth asteroids, it is clear that even in a 21st-Century world of 10 billion people, the potential industrial value of the near-Earth asteroids is enough to dramatically effect the global economy and the environment. If this material is to be of use, however, challenges associated with space industrialization must be overcome. These challenges include the high cost of access to space, the hazards of the space environment, and other difficulties associated with any new enterprise, such as obtaining required capital investments, legal issues, and national policy issues.

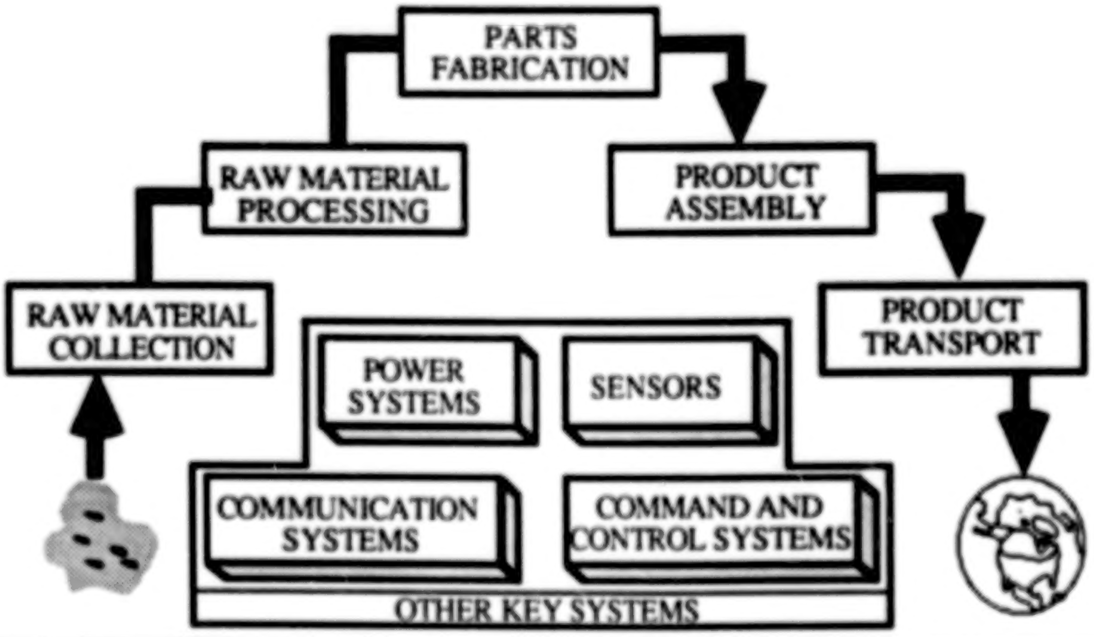
The cost of access to orbit is typified by existing launch vehicles, which cost in the range of 1,000 to 10,000 dollars per kilogram of payload.[11] Technologies are being considered to reduce launch costs considerably. For example, the Advance Launch System which is now under study, may reduce launch costs to about 300 dollars per kilogram. The theoretical minimum cost for delivering payloads to space can be calculated based on the mechanical energy associated with a given orbit. Assuming 10 cents per kilowatt hour, and accelerating payloads to about 8 kilometers per second, the theoretical minimum cost for delivering payloads to low Earth orbit is a little over a dollar per kilogram. However, the technical difficulty of approaching this low theoretical cost is enormous, so for the purpose of speculating about 21st-century space industry, it is reasonable to expect that the cost of access to space will remain high.

The hazards of the space environment include isolation, radiation, vacuum, and zero gee effects. These hazards must be well understood if space industrialization is to proceed. Fortunately, in many cases, these hazards can be turned into advantages. For example, isolation from the terrestrial biosphere will allow space industrialists to use processes that are environmentally unacceptable here on the Earth. Radiation can be turned to an advantage as a power source, by (for example) using the 1.4 gigawatts per square kilometer of solar radiation for thermal power or converting some fraction of it to electrical power to drive machinery.

## BASIC FUNCTIONS OF A SPACE-BASED INDUSTRY

The functions a space industry must accomplish include raw material collection, processing raw materials into useful feedstocks, parts fabrication, product assembly, and transport of products to Earth. Dramatic advancements in the technologies of power systems, sensors, and command and control systems will be needed to build an effective extraterrestrial industry. These functions and the important spacecraft subsystems associated with a large-scale space industry are depicted in Figure 3.

Figure 3 - Basic Functions and Systems of a Space Industry Using Extraterrestrial Resources



Referring to the figure, raw material collection is the process of mining or bulldozing extraterrestrial material and introducing it to the material processing plant. The chief difficulty of raw material collection is associated with the command and control of robotic systems on natural extraterrestrial surfaces. Processing raw materials involves extracting the useful component of the raw materials and converting them to forms appropriate for industrial applications. Extraterrestrial materials processing is presently receiving considerable research attention. Once processed, the material must be fabricated into useful parts and assembled into useful products. The final products must then be transported to the Earth.

## REPLICATING SYSTEMS

If the needed technological advancements are made, an important option to consider for harnessing space resources is that of replicating manufacturing systems. A replicating system can use the energy, information, and matter present in its environment to make copies of itself and some useful product. A key aspect of any proposed replicating space manufacturing system is that of closure. Three types of closure have been identified as important: energy closure, information closure, and matter closure. If perfect closure in each of these areas could be achieved, the result would be a von Neumann machine.[12] Although the prospects of producing a true von Neumann machine are not good, it is likely that within the next century replicating systems with imperfect closure can be made to effectively support space industrialization.

The first replicating factory introduced to space to initiate a space industry is called a space seed. The following discussion is based on a space seed concept developed in a 1980 NASA summer study which represents the most careful investigation to date on the subject of replicating space factories.[13] As part of that study, it was estimated that a space seed, which could be constructed using reasonable technology and delivered to the moon or an asteroid, would have a mass of about 100 metric tons and could replicate itself in about one year.

Figure 4 is a schematic designed to communicate the basic concept of a replicating machine. Based on the assumptions given above, after one year, two factories would exist; after two years, four factories would be present; after three, eight factories would be available, and so on. At the end of 30 years approximately one billion replicating machines would be present. This assumes that the first replicating system is placed in an environment where it can replicate and has sufficient quantities of appropriate material nearby for it and its offspring to replicate. If such material is not present in acceptable forms, some small quantity of mass will have to be delivered to the machines from the Earth. These delivered materials are called "vitamins," in analogy to biological vitamins. An additional assumption is that the transportation system needed to return products to the Earth is produced by the replicating system and consumes half of the mass of the system's products, for example in the form of rockets and propellant.

Figure 4 - The Concept of Replicating Machines

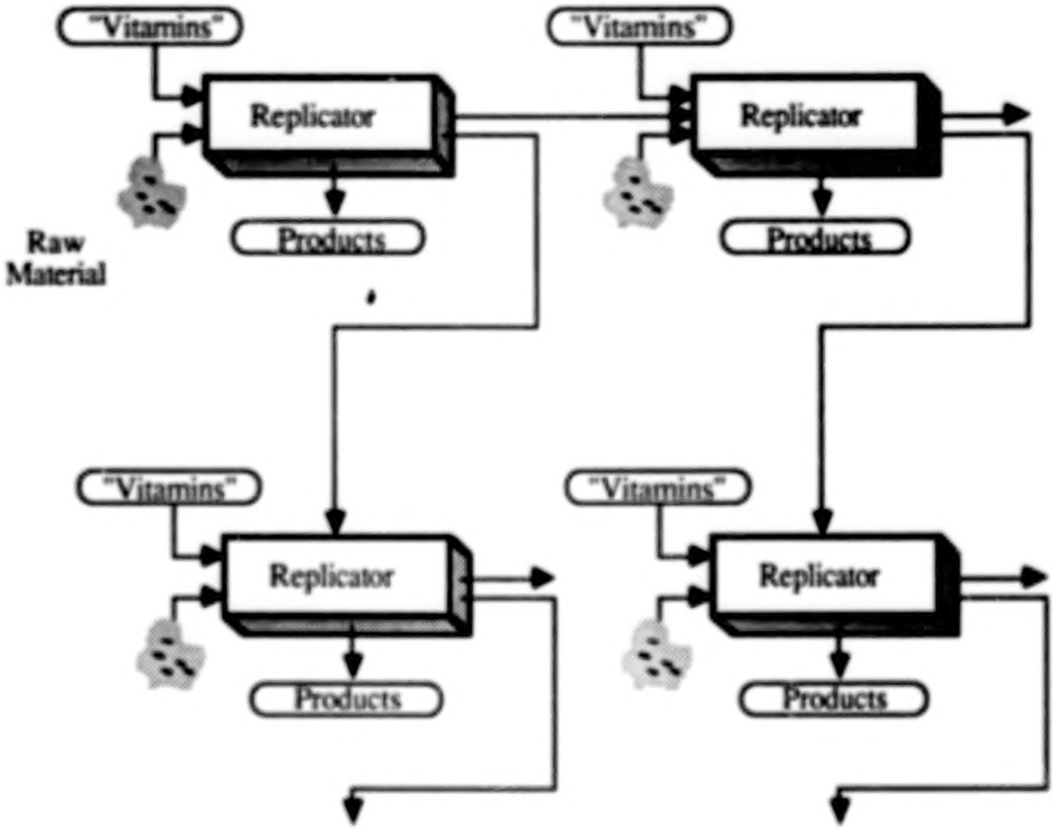
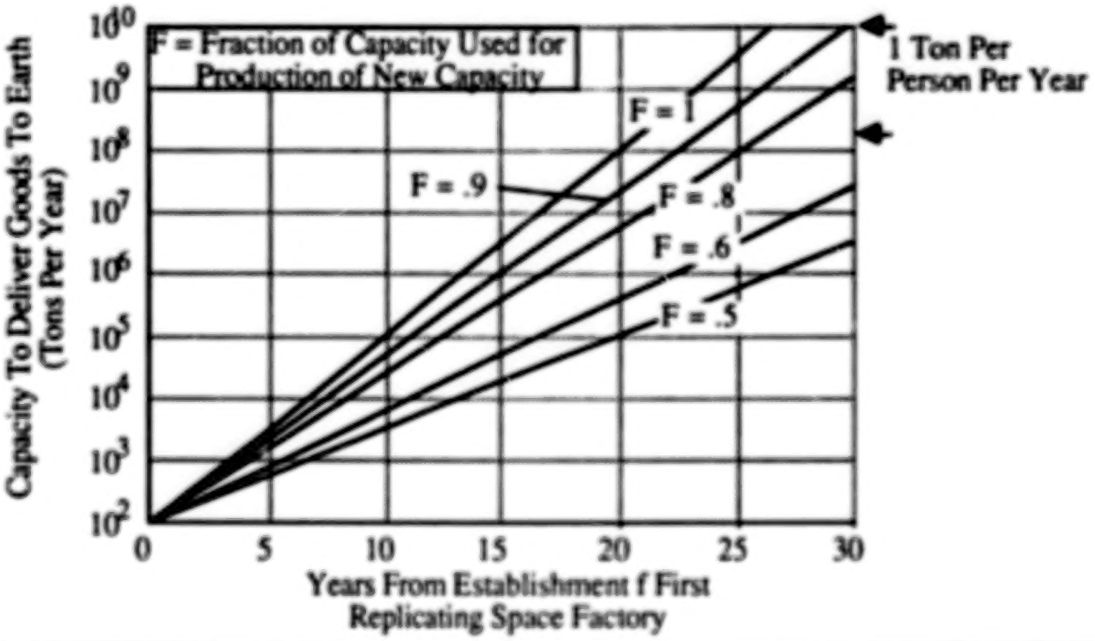

"Vitamins"
Replicator
Products
Raw Material
"Vitamins"
Replicator
Products
"Vitamins"
Replicator
Products
"Vitamins"
Replicator
Products

Figure 5 presents the capacity of an extraterrestrial industry based on replicating machines as a function of time after the establishment of the first replicating space factory. The vertical scale presents the capacity to deliver goods to the Earth in tons per year. The horizontal scale shows the time in years from the establishment of the first replicating factory. Five different lines are shown for different fractions of the capacity used for the production of new factories. The top line, which corresponds to the highest level of capacity to deliver goods to Earth, is based on the assumption that the entire production capacity of the replicating factories is used to produce new industrial capacity in space. The second line corresponds to the case in which 90 percent of the output of the original space factory (and its offspring) are used to produce new factories, and 10 percent of the output is used to produce useful products. The parameter F is used to describe the fraction of factory output devoted to replication. As can be seen, values of F near 1 give the highest rate of development of industrial capacity.

Figure 5 - Industrial Capacity of a Replicating Space Manufacturing System



These same assumptions are used in Figure 6, which presents the total quantity of products delivered to the Earth as a function of time from developing the first replicating factory. Note that the F=1 line is not present. This is because the F=1 condition corresponds to no useful goods returned to the Earth, certainly not an advisable operating condition. Another interesting point to be drawn from Figure 6 is the somewhat counter-intuitive observation that the larger values of F correspond to the largest return of goods to the Earth in time periods greater than a few years. This suggests that replicating factories should be designed to focus the majority of their capacity on replication, especially in the first few years after deployment.

Figure 6 - Accumulated Product of a Replicating Space Manufacturing System

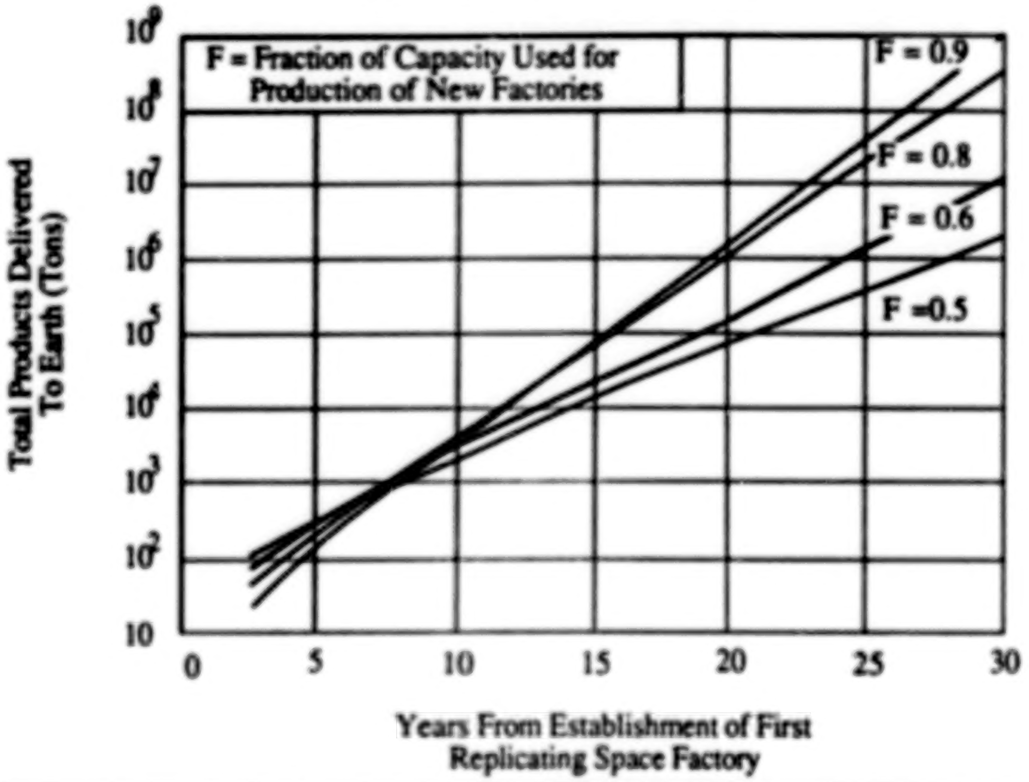
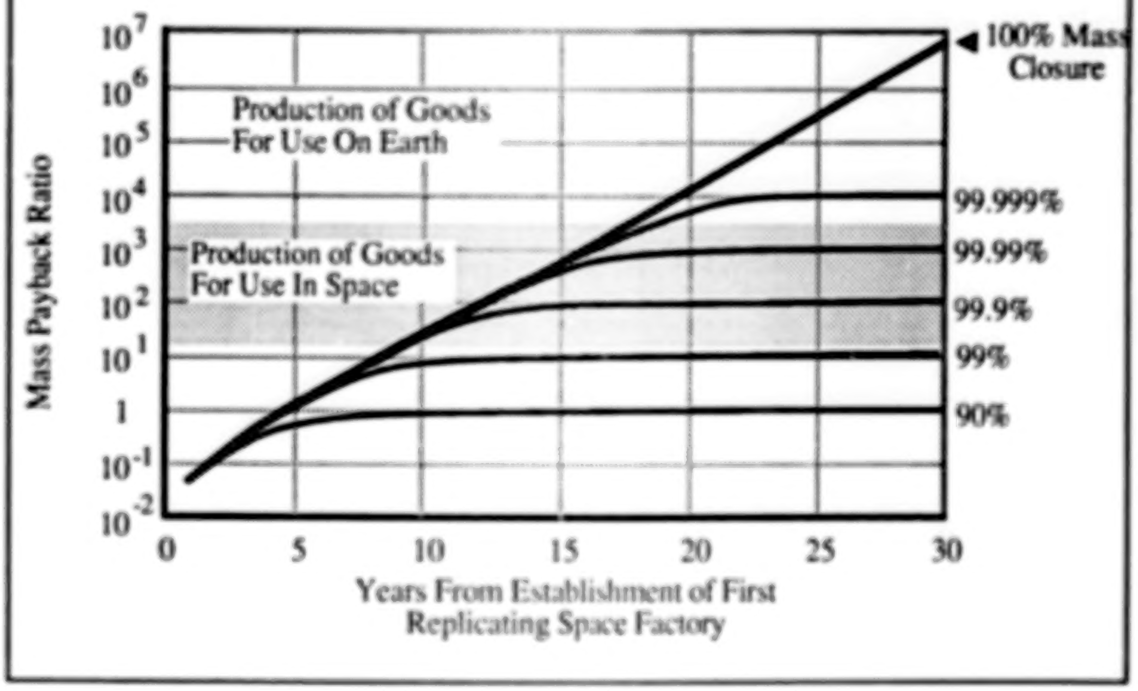


Figure 7 is an attempt to communicate exactly how important mass closure is to any potential replicating space manufacturing technology. The vertical scale shows the mass payback ratio of the replicating space manufacturing system for the case of F=.9. The horizontal axis shows years from establishment of the first replicating space factory. Mass payback ratio is a term commonly used in the study of extraterrestrial resources and is defined as the ratio of the mass of useful products returned, in this case industrial goods, to the mass which had to be launched into orbit to make those goods available. Figure 7 depicts six different curves for six different mass closure efficiencies for the replicating machine. The top curve represents a 100 percent mass closure and shows what the mass payback is as a function of time for a von Neumann machine with an initial mass of 100 tons and a replicating period of one year. The other curves correspond to the labeled mass closure rates.

Figure 7 - Mass Benefits of a Replicating Space Manufacturing System (F=0.9)



A mass payback of less than 1 is unacceptable. Such a condition would imply that for every kilogram of material returned from the space industry, more than 1 kilogram of material would have to be launched into space. To justify development of some level of replicating machine technology, a promise of a mass payback ratio of a least 10 or so will be needed. For example, if one were making communication satellites, a mass payback ratio of 10 would suggest that for every ton of vitamins launched into orbit, replicating machines could return 10 tons worth of communication satellites. For a highly cost-effective replicating space industry, mass payback ratios of at least 100 to 1,000 would be desirable. On the other hand, if considering the return of manufactured products to Earth for terrestrial use, then the required mass payback ratio will have to be quite large, probably well over 1,000.

Figure 7 also suggests that for replicating systems to make sense for a space manufacturing industry, virtually 100 percent mass closure is needed. If studies are done that convincingly show that high rates of mass closure are very unrealistic, then we probably shouldn't be too interested in space applications of replicating machines.

An important caveat applied to the discussion of Figures 4 through 6 is that these figures were generated based on the simplest possible strategy for replicating space systems. Specifically, the value of F was assumed constant in time. In all likelihood, this assumption will be found to be far from optimal, and the conclusions drawn here should be extended only with care.

If the development of a replicating machine technology is undertaken, there are a number of steps that the 1980 NASA summer study suggested would be appropriate. Table 2 is an adaptation of the development milestones suggested by that study.[13]

Table 2 - Development Milestones for a Replicating Space System

1. Design and construct a system which, when supplied only with parts and subassemblies, can duplicate itself.

2. Design and construct a system which can duplicate itself, and, in addition, produce some useful product.

3. Design and construct a system which, when supplied only with feedstock, can duplicate itself.

4. Design and construct a system which, when supplied with raw materials only, can duplicate itself.

5. Design and construct an automated, reprogrammable, multiproduct system which can, from raw materials, duplicate itself.

6. Design and construct an automated, reprogrammable, multiproduct system which, using only materials available on an extraterrestrial surface and using only processes possible in the space environment, can duplicate itself.

7. Design and construct an initial automatic "seed" system which, if placed on an extraterrestrial body, can deploy itself as a functional automated, reprogrammable, multiproduct system and replicate itself using only processes possible in the space environment.

8. Design and construct an initial seed which can produce useful products on an extraterrestrial body.

9. Design and construct an initial seed which can produce useful products on an extraterrestrial body and manufacture a transportation system to deliver those products to useful locations such as Earth orbit or the surface of the Earth.

A 1988 National Research Council report titled The National Challenge in Computer Science and Technology, recommended that the United States adopt the development of replicating machine technology as what they called a "grand challenge.[14]" The council had several interesting observations about why such an undertaking would be advisable. Specifically, they felt that if successfully developed, the technology for replicating machines could create significant spin-offs in industry and government. They suggested that the solution to the problem of developing a replicating machine falls within many disciplines. Some of the disciplines they sighted included knowledge capture for reverse engineering, design for manufacturability, and robotics. Further, the council suggested that the development of replicating systems technology would require and would generate significant breakthroughs and fundamental advances in computer science technology. They also pointed out that success or failure would be clearly established and appreciated by nonexperts and would require long-term, stable funding at significant levels. The funding levels that they refered to were on the order of a hundred million dollars per year, but they also suggested that success in this endeavor would be guaranteed because the payoff, even if this development were to achieve far less than total success, would be substantial. Besides advancing the state of the art, the pursuit of a challenge like this would create a new generation of leading computer researchers and engineers who would, in turn, contribute to the creative and effective use of technology throughout the nation. They called it a grand challenge not only because of the immediate accomplishment it would entail, but because achieving replicating machine technology would give rise to immense technological spin-offs benefiting industry, defense, and society.

## CONCLUSIONS

- The near-Earth asteroids contain tremendous material resources which, if harnessed, could have a beneficial effect on the world economy and the environment of planet Earth.

- Autonomous manufacturing technologies techniques may be needed to harvest the wealth of the asteroids.

- Given the magnitude of the potential benefit of a full-scale space industry, development of advanced automation technology and extraterrestrial-materials-processing systems should be vigorously pursued.

- Developing replicating manufacturing technology could be a grand challenge on which to focus American energies in the 21st-century.

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# Speculations About Goals and Challenges in a Millenium of Space Ventures

**Paul B. MacCready**

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## Summary

One characteristic that seems to be shared by the major exciting ventures in space is need for the commitment of huge resources over decades. Increasingly competing for such commitment will be initiatives focused on the earth, as rapidly growing human pressures are found inconsistent with preserving our agricultural base and the ecosystem on this non-expanding globe Therefore the best, and perhaps only, chance for significant pioneering in space will arise from first helping establish a sustainable situation on earth.

## Introduction

The following speculations come from the only person at this conference who has had practically no professional involvement with space ventures. From this vantage point of non-involvement, I am able to look at the opportunities from a personal perspective, and, without an investment in being right, present views that may well be wrong. If the views stimulate discussion, they will have served their purpose.

The views evolved from an unusual decade of experiences with vehicles that operated on land, sea, or air, with extremely low power -- the power of human muscle, photovoltaic cells, or small batteries. All of these slow, fragile devices are " impractical," by any ordinary definition of the word, but they may nevertheless have considerable value:

- By focusing attention on doing more with less, they broaden perspectives and help raise expectations and standards about conserving material and energy -- a topic of fundamental importance as civilization moves into the era of limits.

- Those that meld biology with engineering prompt us to think about balancing nature and technology so that technology remains a servant rather than becoming our master.
- As examples of innovation they stimulate us to think about how to remove mental blinders in order to foster creativity, perceive the real world more honestly, and contemplate the future with more vision.

Thus these vehicle projects have been the stimulus for getting into very serious subjects, such as the meaning of life and the future of civilization. The exercise of looking at the major issues focuses one's attention on goals. A dominant goal many people share is "mankind reaching a comfortable accommodation with the flora, fauna, and resources of this limited earth". This is not a universally accepted goal. The pragmatists realize it is unachievable and therefore unrealistic. Some people dedicated to space ventures consider a more appropriate destiny for earth is to launch "intelligence" into the vastness of the cosmos; the fate of life on earth is deemed less significant. The ultimate goal is not definable by science or technology. The choice derives from philosophy or religion, or from circumstance that decides for us.

## The Present and Future Situation on Earth

Few people really comprehend the magnitude of the transient in which civilization now finds itself. The individual sees that the present is rather similar to last week, and to last year, and so does not perceive the bigger picture automatically

We are all products of the recent industrial and technological revolutions. We forget that just 200 years ago the human population was small and virtually all humans got by on their own power, supplemented just a bit by the muscle of domesticated animals and some occasional wind and water power. In 1959, British industrialist Henry Kremer established a sizable prize for human powered flight -- a remarkable challenge that connects modern technology to our biological roots and so gets us thinking about the changes over the last two centuries. Our Gossamer Condor won Kremer's prize in 1977, and this got me onto the lecture circuit, having to sort out thoughts about the project's meaning and where it leads. This was the catalyst for an emphasis on the subjects of change and the future.

To put population growth in perspective, I note that in 1925 when I was born there were 1.7 billion people. It had taken life on earth over 3.5 billion years to reach this human population level. Now, in my brief lifetime, the population has tripled to 5.1 billion. At the same 1.8% increase per year, there will be another 1.7 billion in just 16 years, when your nursery school toddler is starting college. In 50 years at this rate the population calculates out to over 12 billion; in 100 years over 30 billion. In 1700 years, at this exponential growth rate the mass of people will exceed the mass of the earth; it will be people all the way down. Obviously a limit will be reached sometime earlier, but I doubt that the maximum will be down in the 8-10 billion range as many people argue. They ignore the increased longevity that modern science is likely to provide, the increased food production that new technology can help generate, and the pressure for population growth in the culture of some lands and religions.

Omitted in this discussion of human population increase is the impact on the global flora and fauna, the other 30 million species now so threatened by one very recently arrived species, homo sapien. The latest estimate is that we are now causing extinction of species at one every four minutes! Mankind's conquest of nature is increasingly effective . The tragedy is that we are winning.

The relative importance of mankind vs the rest of nature's species is for us to decide. Unfortunately the evaluation is made by humans, who are not unbiased and instinctively assume a human importance. Exploring such questions is helped by getting our minds opened by science fiction literature and by scientific cosmological investigations. To start from a broad perspective, we can imagine discussing the subject with some galactic explorer that visits earth every 100,000 years or so.

However, we humans will decide these issues, by careful thought or by default. Nature will probably get short shift, except to the extent that we perceive a direct, short term benefit to us from respecting it. We are starting to think about the inconveniences to us as top soil erodes or trees for fuel disappear, or as the demise of rain forests disturbs us globally via the green house effect. Let's say technology can somehow overcome these troubles. There is still the deep question of whether a person is really human when divorced from the ecological base in which our evolution took place -- similar to the question of whether an animal species still exists when all its members are confined to zoos, and not connected to its natural habitat. Humans are so adaptable they can live in large numbers in a confined, prison - like atmosphere. Without a change from our growth - oriented culture, this may be the prevailing situation in not too many decades. Obviously the subject of goals for civilization deserves an emphasis it is not presently receiving. The crystal ball is cloudy beyond even a few decades; a millenium is beyond realistic comprehension.

## Conclusions

The exploration of space opens up thinking about big issues related to the meaning and goal of human life, and the destiny of civilization. When one looks at the growing pressures an expanding population with expanding per-capita expectations puts on a non-expanding earth with a shrinking ecological and resource base, one concludes that society will soon be paying more attention than at present to trying to achieve an acceptable situation on space ship earth. The subsequent allocation of resources of dollars, brains, and interest will likely preclude the mounting of any giant space initiatives, even if the initiatives can be made international. If a long term comfortable accommodation with earthly limits is achieved (and there are only a few decades before a favorable outcome is precluded), the space missions can then become reality. If the accommodation is not achieved, big space ventures will not be carried to completion. It therefore behooves the space groups to delay the push for big programs and devote more resources to the uses of space to help global survival: environmental monitoring, communications, solar terrestrial relationships, exploration of our (and the earth's) past, and our role in the cosmos.

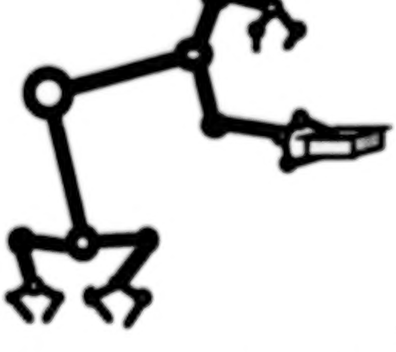
# PROPOSAL FOR A REMOTELY MANNED SPACE STATION

**Marvin Minsky, MIT**

The United States is in trouble in space. The costs of the proposed Space Station Freedom have grown beyond reach, and the present design is obsolete. The trouble has come from imagining that we have only two alternatives: manned vs. unmanned. Both choices have led us into designs that do not appear to be practical. On one side, we simply do not yet possess the robotic technology needed to operate--or assemble--a sophisticated unmanned space station. On the other side, the manned designs that are now under way seem far too costly and dangerous, with all of its thousands of EVA hours. We'd accomplish more at far less cost--by proceeding in a different way. Here is what we ought to do to achieve this third alternative

*Design a space station made of modular, Erector set-like parts.*
*Develop mechanical telerobots to be remotely-controlled from Earth.*
*Train earth-based workers to build the station in space using simulators.*
*Launch a small preassembled spacecraft with a few of the telerobots.*
*Ferry the telerobots into orbit , along with stocks of additional parts.*
*Instruct the trained terrestrial workers to remotely assemble a larger station.*
*Launch materials for additional power, and life-support systems.*
*Finally, send human scientists and explorers.*

The initial cargo would begin with a conventional pre-assembled system for propulsion, power, and communication. The novel aspect is to equip the station with three or more remote-controlled mobile mechanical hands that can move themselves from place to place. These manipulators--call them "telerobots"--are controlled by human operators who use "power gloves" and "control suits" to translate their movements into the corresponding telerobotic acts. Each telerobot, in turn, provides a sense of "telepresence" to its operator by returning visual, auditory and tactile sensations, using head-mounted visual and manual force-display technology. Simple such systems already exist, and better versions could be developed in a very few years. In less than a decade, the project would be years ahead of what's being planned now. If we use a suitably modular design strategy, we should be able to use these telerobots to maintain and repair one another--as well as other components of the space station. Our proposed "tree-robot" design has but a few types of components, each made on scales that differ in size only by factors of two.

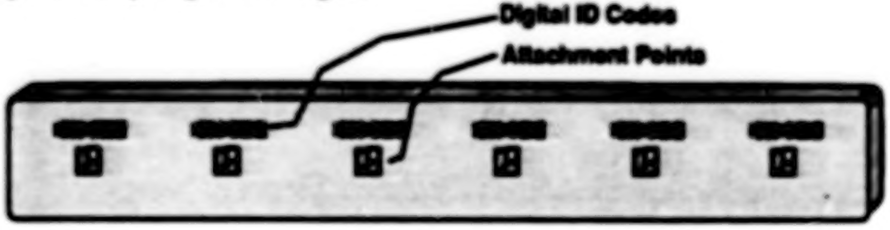


A Binary-Tree Telerobot

The initial cost of such a space station could be very modest, because it lets us postpone costs of safety and life-support systems until manned operation becomes desirable. The first human operators will work on Earth; later they'll be on the station itself--and then, before long, they can work on the Moon.

# THE MICRO-MODULAR SPACE STATION

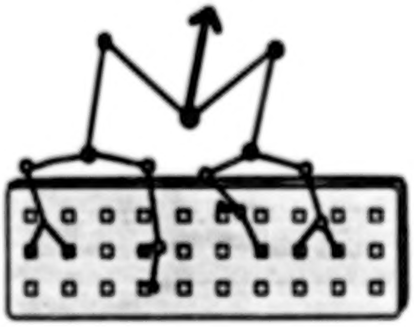
The use of remotely manned telerobots will let us re-think all our old concepts of space technology. Now we can aim toward making even the smallest structural components so modular that, in effect, the entire station can be built with elements like the kinds used in construction toys like Erector™, Meccano™, FischerTechnic™, LEGO™, or TinkerToy™. Every surface of each component should be studded at regular intervals with standardized "attachment-points" each labelled with a unique, machine-readable identification mark. This policy has many large advantages:



A typical structural component

- *--It enables a computer to keep track of all spaceborne materials.*
- *--It permits re-use of the same parts for different purposes.*
- *--It simplifies simulation, assembly, and design.*
- *--It reduces the total inventory mass of material and spares.*
- *--It simplifies training for assembly operations.*
- *--It simplifies subsequent development of autonomous robotic operations.*
- *--It facilitates both telerobotic and manned mobility.*
- *--And it simplifies converting lunar or asteroidal materials into useful components.*

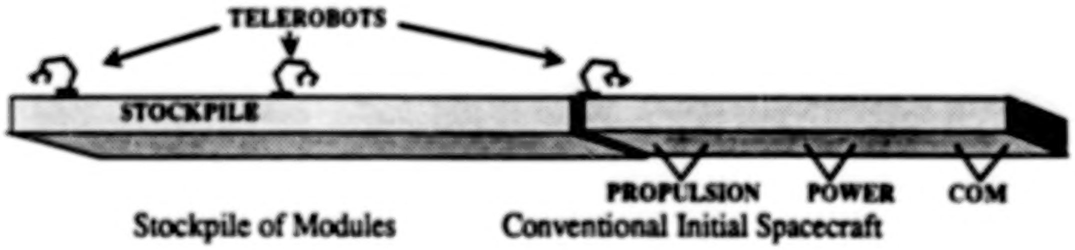
The use of micro-modularity will make it easier to design and debug both the structures themselves and the skills involved in assembling them. The availability of attachment-points will make it easy for the telerobots to move from place to place.



Telerobot traversing a beam.

# STAGES OF DEVELOPMENT

The initial configuration should begin with a conventional spacecraft equipped with power, propulsion, and communication facilities--and bearing a stockpile of modular parts. Then a much larger station can be remotely assembled in space.



## I. PREPARATION FOR INITIAL LAUNCH:

- *Develop the modular components and connectors for space structures.*
- *Develop the modular components and connectors for the telerobots.*
- *Develop the telepresence communication systems for the sensors and actuators.*
- *Use two Telerobots to assemble a third, under neutral buoyancy conditions.*
- *Start training workers on Earth in operational and maintenance skills.*
- *Establish minimal-delay satellite communication network.*

## II. INITIAL LOW EARTH ORBIT CONFIGURATION.

The initial unmanned station in LEO is equipped with conventional packages for power, orbital maneuver propulsion, and satellite communications. Its principal payload should consist of three or four telerobots and a stockpile of parts. All further construction will be done by telerobots controlled by workers on the ground, who will reconfigure and extend structures as necessary. Because it is both desirable and feasible to move slowly at first, the initial power requirements will be small; the telerobots should need less than 1KW of power.

## III. DEVELOPING LEO SPACE ASSEMBLY OPERATIONS.

- *Assemble and test larger structures.*
- *Operate instruments for scientific research.*
- *Practice Telerobot disassembly and repair.*
- *Launch materials for life-support systems and living quarters.*
- *Launch materials for commercial prototypes.*
- *Experiment with tethered and free-flight transfer operations.*

## IV. BEGIN LARGER SCALE OPERATIONS.

- *First manned residence and industrial operations.*
- *Astronauts practice local, delay-free control of telerobots.*
- *Introduce semi-automatic assembly operations, using planning programs.*
- *Assemble and test larger life-support and residence systems.*

## V. BEGIN LUNAR OPERATIONS.

- *Proceed with similar procedures to assemble a lunar base.*
- *Experiment with refining lunar materials.*
- *Begin preparing interplanetary or asteroidal exploration vessel.*

# THE CONCEPT OF MICRO-MODULARITY

In contemporary NASA jargon, the term "module" is applied to any self-contained system, even one so large as an entire space shuttle payload. Here we shall use the term "micro-module" for the idea that every structure should be composed of standardized parts --like those of children's construction sets --wherever this is feasible. Even simple containers should be assembled from smaller plates and beams, except for imperative reasons. This policy may sometimes cause small increases in spaceborne mass, but will usually yield large economies when the same parts are later reconfigured for other applications.

**ATTACHMENT-POINTS.** Every micro-module should be covered with Attachment-Points, marked with unique and precisely located machine-readable optical identification patterns. We must also provide suitably standardized connectors for assembling larger structures. Each connector device must be easy to apply, test, remove, and firmly lock. Re-usable rivets might suffice, but we should also seek a reversible welding technique.

**SPECIFICATION REGISTRY.** Adopting a uniform attachment-point identification scheme would enable us to maintain an international register in which every object launched into space could have a unique ID. Whenever a new structure is needed, a CAD system could then locate required components, even considering those in other, already assembled systems. These spatial ID markings could also be used to locate remote instrumentation devices. For example, passive vernier strain gauges could be located at appropriate points, because optical scanners could easily read them--at no additional hardware cost.

**AUTOMATING TELEROBOTIC OPERATIONS.** What if we wish a telerobot automatically grasp and assemble a certain set of objects? We cannot automate many such functions today, because our present-day robotic technology is not mature enough to do such things reliably. In particular, the technology of Machine Vision is still too weak. But adopting micro-modularity, we could do this today, by exploiting the precisely located ID markings of our micro-modular components to access data bases that precisely specify the spatial shape of every registered micro-module. That knowledge-base would make it easy to build software to reliably locate and assemble the needed parts. Such programs could be made extremely robust by testing the match, at every step, between the sight, and the feel, of the actual scene with what our simulators predict. At the first sign of discrepancy, the system can stop and revert to remotely-manned operation.

**MOBILITY.** How would our telerobots move from place to place? This would be very hard to do in a conventional spaceship, where each change in location poses new mooring problems. But if every surface of the micro-modular spaceship is studded with attachment-points, the telerobots can exploit these for mobility. At each point, then, a Simulator could plan ahead, locating ID-points for further steps, so that the telerobot can move by grasping one attachment-point after another: each new one being verified, both by vision and actual touch. At the same time, the simulator would also confirm the suitability of every new attachment-point for anticipated loads and strains.

**CONCURRENT SIMULATION.** Adopting micro-modularity would simplify full-scale simulation of the entire station, under both actual and hypothetical conditions. The goal should be to maintain a data base that holds, and constantly verifies, the location of every known component--including all available knowledge about the physical states of every part: their stresses, velocities, temperatures, fatigue histories, etc. Such a system could be used both to plan and debug each new construction, and also to train the telerobot operators, to support them with anticipatory feedback (to reduce the apparent time-delay), and to automate routine procedures.

## TELEROBOTS SHOULD BE MODULAR, TOO

The telerobots themselves should be made of only a few pre-assembled components, with each skeletal element equipped with its own motors and sensors--so every unit is easily replaceable. The motors need not be very strong, but each joint must include a fail-safe brake that locks when local power fails. A single docking connector should complete both mechanical and electrical connections. Supplying power and signals has always been hard, in designing terrestrial robot arms. But this should be somewhat simpler in space, where the power required is so much less. Because a typical motor needs less than one watt, it might suffice to run a simple two-wire bus throughout the tree, treating every sensor and motor as a single, separate network node.

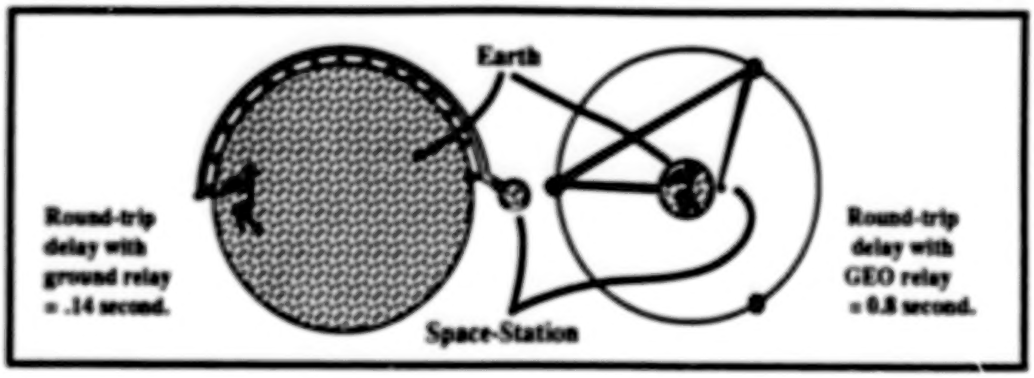
Such telerobots would be dependable, because the tree-like design has enough "redundant" degrees of freedom to be usable in spite of occasional joint failures--provided that they lock when they fail. Earth-based manipulators, have always been designed with as few joints as possible--because gravity demands such relatively massive motors and beams. Indeed, to develop the tree-robots on Earth, we may have to test them in neutral buoyancy conditions.

A more radical approach to mobility would equip each joint module with an independent communication system and power supply. A one-watt motor with 25% duty cycle can run a full day from a single size D rechargeable cell. NASA's free-flying teleoperators will be impractical for large scale work because they consume too much reaction-mass. But self-contained telerobots could propel themselves by the ballistic exchange of reaction-mass objects--including batteries. If an object is projected slowly enough, its trajectory can be verified before it exceeds the reach of the throwing arm; in any case such objects could be retrieved by tethers. For larger scale operations, we could surround the entire workspace with a tethered tetrahedral skeleton, tensioned by the momentum flow of masses exchanged between vertices.

## TELEPRESENCE AND TIME DELAY

Why have telerobots not have been used more in space? This seems largely because of a widespread belief that no one can work effectively through systems involving time-delays. But I am convinced that this is wrong--if the feedback delays are large enough! People are telerobots, too, because our bodies and brains must always cope with internal time-delays of the order of 0.2 seconds between sensation and action. You cannot hope to catch a ball by "keeping your eye on it." Because of your reaction-time, your brain must anticipate its trajectory, at least for that final interval. This critical subject is discussed at more length, in this essay's Appendix.

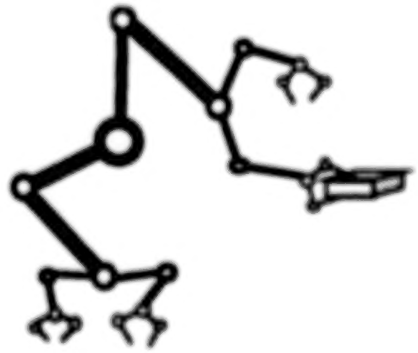
Transmitting a round-trip signal between Earth and a low-orbit satellite would take at least 1/7 second, and would require a very expensive equatorial belt consisting of a dozen or more earth-based communication stations. A more economical system would relay signals between Earth and three geosynchronous relay satellites. This would involve a longer delay of nearly 1 second.


Earth
Round-trip delay with ground relay = .14 second.
Round-trip delay with GEO relay = 0.8 second.
Space-Station

With suitable training, remote operators should be able to learn to perform useful work at "one-fifth real-time" speeds--and eventually faster, when some of the effects of delays are reduced by exploiting computer-supplied "anticipatory feedback" and "supervisory control" modes of operation. In the appendix to this paper, we argue that this mode of work is not entirely unlike that done by people who operate large construction-cranes--systems that impose similarly slow reaction times.

## PRODUCTIVITY ADVANTAGES

A common objection to these ideas is that these time delays will force the work will proceed too slowly. But simple arithmetic refutes that view. Each pair of remotely manned mobile hands should be able to accomplish as much as a space-borne astronaut. For, even in the unlikely case that space-suited workers could tolerate 6 hours per day of EVA operation, it is hard to imagine this yielding more than the equivalent of two to three hours of earthside work. Consequently, we must compare 20-hour human weeks against 168-hour telerobot weeks--and even this is conservative, because anticipatory feedback computer enhancement should at least double final efficiency. So, even proceeding at 1/10 speed, each telerobot could accomplish a human equivalent of work--at perhaps one percent of the other's cost, and an infinite gain in safety.



What would such systems cost, in mass? Each telerobot need not weigh more than about 10 Kg. In contrast, each human EVA operator needs on the order of 2500 Kg, when we include not only the person's own weight but also that of the spacesuit, consumables, and life-support equipment, as well as the mass needed for reinforcement of pressurized living quarters--to say nothing of ferrying astronauts home. Remotely-manned operations offer a hundred-to-one advantage in cost.

## THE US SPACE PROGRAM IS A HOSTAGE TO SAFETY

Our manned space expeditions have been wonderful accomplishments, but were expensive, risky, and limited. And although it is claimed that humans on board made it possible to do emergency repairs, the actual record is not so impressive. The Apollo 13 crew was unable even to examine the damage. The Skylab parasol repair involved a mechanically simple task; yet the crew was able to restore only a portion of the lost function. STS crews have also managed only rather simple repairs. And as for automated unmanned missions, some of them did indeed work remarkably well, but mainly because of conservative plans, with almost everything planned out years in advance.

Today, though, Space seems tedious. A hidden cost has been overlooked: neither manned nor unmanned ships permit extensive repairs in flight, hence we're forced to depend far too much on maintaining reliability. Trustworthiness, not resourcefulness, has become the program's centerpiece--and the name of that game is constrained design. This makes us pay a crippling price--*of having to freeze our plans years in advance*. That's what we did in the early years, when we simply had no alternative. But now this has restricted us to obsolescent technologies, and institutionalizing a sluggishness that virtually bars us from challenging Space. Nothing new can be tried any more.

This problem has grown in the past few years because of our increasing concern for human safety. The Challenger disaster substantially delayed the entire space program, to reduce the chance of one accident. Prior explorers were careful, indeed, but not to any so drastic degree. Our astronauts now play the roles, not of leaders, but of "hostages", because we will do virtually anything to protect their safety. This is no mere concern of NASA alone, but part of a broader phenomenon in which people demand outlandish constraints on every aspect of daily life. Even in medicine--the technology of life itself--we have become so concerned with guarantees that we won't dare to save a hundred lives at the risk of losing any of them. This poses for NASA a dreadful dilemma: a perception that the public will support nothing less adventurous than manned exploration--but will never forgive any accident. This new cultural context provides no way to provide NASA with the "liability insurance" it needs. Manned flight is too risky and expensive, while unmanned operation is too inflexible and unsensational. Like many physicians in recent years, this dilemma has led NASA virtually to retire from practice, albeit without admitting it.

We might escape from this double-bind by adopting this alternative--of remotely manned operation. Using that way to go into space, we can prepare each expedition by using earth-based workers to do what would be, in space, much more dangerous, costly, and difficult. Because the initial station is unmanned, yet still able to exploit the intelligence of its remote operators, we can use it to try more experiments and develop new technologies. Using remotely manned operation, we can achieve our goals while gaining versatility--with at least a tenfold reduction in cost. As for safety, *no one gets injured when no one is there*. Nor should we reject this as a step away from manned exploration. On the contrary, it would speed up developing what we'll need for more ambitious voyages.

## THE ISSUE OF POPULARITY

When they hear this proposal, most people say, *"I agree that this might be a good idea, but I'm sure that the Public won't buy it"*. But it seems to me that this belief is based on a wrong perception.

The public WILL buy it. Those objections might have been valid in the 1960s, but now they're out of date. In fact, the real problem is the opposite: *the public has grown weary of two decades of dull and non-productive man-in-space activity*. These days, few people learn the names of astronauts; today, it is Robots that are "in", like Artoo-Deetoo, HAL and Terminator. The youngsters adore Transformer™ toys, and spend their fortunes on new interactive game cartridges--new virtual realities to spark their personal game-machines. It started out with those TV games--old Breakout, Pong, and PacMan stuff, and then evolved from Zaxxon and Megaroids to Mario Brothers. This year, a million PowerGloves work in our children's homes, just waiting for their owners to manipulate some things in space. Those skeptical critics are out of date!

A remotely-manned space program will give the public a chance to share in the fun. Using simulators, we can post competitions to recruit people talented at assembling tricky systems and mechanisms. Soon there could be thousands of telerobot operators, working in local communities. Then Space will seem accessible, no longer only for strangers in far away places. This program will be more exciting in any case, because the new constructions and experiments can be more adventurous, and can proceed so much more rapidly, than could any involving risk of life. New structures and experiments will be completed much more frequently, attracting more active public interest. For reasonable fees, even non-technical persons will be able personally to experience the operation of actual telerobotic systems, first in far-away places on Earth, next in near-space, and finally, right there on the moon. Many people can thus get involved in active exploration roles.

Furthermore, the new telepresence simulation technology will contribute to new forms of entertainments for the public at large--of the kinds called "virtual realities". Imagine teams of players on moons (whether real or imaginary) engaging in strange new contest-games--building, or fighting, or playing games--or simply exploring and making friends. Among the many individuals engaged in these new practices, popular "stars" will start to emerge, as in all domains of human enterprise--and as our old heroes fade from mind, we'll adopt new idols of different kinds.

Acknowledgment: I wish to thank Robert E. Maas for many suggestions; also Dale Amon and Nathan Ulrich. The basic idea of tree-robots was first proposed by me in the 1960s, and later developed independently, and in much more detail, in Hans Moravec's *Mind Children*, (Harvard University Press, 1988)

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## Appendix on TELEPRESENCE and TIME-DELAY

*How could we start a new program based on such an immature technology? The project would have to wait too long before such telerobots become available.* People have said this for two decades. There has actually been substantial progress in systems that "reflect" the joint-forces back to the human operator. The general experience has been that people can perform gross operations at normal speeds, but delicate operations may take 5 to 10 times longer to perform. This is partly because previous manipulators were so clumsy, having only plier-like grippers; more dexterous multi-fingered hands are now in development. Also, previous teleoperators lacked tactile feedback at their fingertips--and small improvements in this domain will yield major gains in performance.

*Conventional manipulators have only 5 or 6 degrees of freedom, but you propose dozens of joints. How could we possibly depend on such complex new gadgetry?* Increased complexity does not imply less dependability. Additional (redundant) degrees of freedom are actually desirable! A person with an injured joint usually can manage things; even if one can't walk normally, one may still be able to limp. Terrestrial robots have never been designed to exploit this possibility, because it has been so hard for them to support their own weight against gravity. But redundant design is more feasible for low-torque work in low-gravity space, where it will actually increase reliability--if every joint has been equipped with fail-safe brakes.

*Won't time-delays make it impossible for people to use remote manipulators.* This is the most common objection to remotely-manned space operations. People often cite instances in which time-delays cause difficulties. Let me paraphrase some examples from my E-mail files:

> "In a certain experiment, a TV camera and remote driving set-up was installed on a Go-Kart, with a five second time delay between the camera's transmission and the driver's video display. The driver then had to negotiate an obstacle course of moderate difficulty. Nobody was able to successfully negotiate the course."

To this, Paul Dietz replied, *"I don't think this proves anything, except that a 5-second delay means you cannot operate a Go-Kart at normal speed. Need I remind you that teleoperation of a lunar rover was accomplished years ago by the Soviets?"* In a similar vein, Joe Dellinger rejoined, *"I remember reading an article describing how they train pilots of Oil Tankers. They sit them in a very small very slow motor boat in a small pond, and put HUGE delays on all the controls, like 30 seconds. The article said that at first the pilots would crash the boat against the walls, etc, etc, but with a little more practice they would learn to pilot it exactly where they wanted to go without thinking."* And Dellinger went on to add, *"Mammalian nerves carry signals faster than "more primitive" life's did, and yet 100 foot long dinosaurs whose nervous system probably took half a second to carry a signal from their hind feet to their head and back evidently walked around on irregular terrain at respectable speeds without tripping over their own feet."*

When first you learn to drive a car, you find that it takes time to change direction--but soon you learn to anticipate. Similarly, at first it seems hopelessly awkward to operate a large boat, an airplane or a construction crane, or for pianist at first to play a pipe-organ. Eventually, though, most people learn. It seems to me strange to hear such concern with telepresence time-delays, when all around us we see successful control of delay-constrained systems. In most cases, all we'll need to do is--***slow down.***

> "Some international phone calls introduce short (1/2 second) delays into your conversation, and this makes your standard speech protocols break down. It causes collisions in which both sides hear a dead area, start to speak, and then collide again with each other. I've found these conversations to be very fatiguing."
>
> "It is difficult to work with computers over a heavily used INTERNET trunk that imposes large delays. It is frustrating to get several characters ahead and then find that I've made a typo. If I were working with a process where each action had an effect on the following ones, I could be far off track by the time I got feedback that I'd made a wrong action!

Again, we merely need to slow down. Those telephone calls would work quite well, if both parties were willing to "over and out". That typist would have no trouble at all, when typing at a slower rate. We ought to remember that humans, too, are telerobots connected to brains. We all adhere to a cultural myth that mind is directly connected to world. A sounder view would recognize that human sensory-motor loops are not instantaneous, but take some time--of the order of $T = 1/6$ second. Therefore, when we introduce, not a *delay*, but an *additional delay* of magnitude D, we should expect our performance speed to be reduced by the factor $T/(D+T)$. Thus, a one-second *additional delay* should permit one to work at 1/7 real time--for example, when operating devices on an earth-orbiting space station, as seen through a geosynchronous relay satellite system.

In any case, so far as large-scale space operations are concerned, the additional telepresence delays should present no trouble at all--because we have to slow down in any case. For example, to rotate a 10 meter beam, one would normally apply very small accelerations, because spaceworthy structures have flimsy parts that cannot tolerate earth-scale stress. Large-scale work can *only* be done in slower than normal "real time".

*But we can't always slow down. What about work that must be done in "real time".* The phrase "real time" should not be used to mean "instantly"--because *no* system can react without delay. That expression should refer, instead, to the sorts of things that a person can do in the order of 0.15 seconds. Consider, for example, that popular party-trick of suspending a dollar bill by holding the top about one inch above and between a victim's fingers. When you release the paper and it falls, no one can catch it. Human reactions are simply too slow. But many such problems will be simpler in space--if we think of vacuum and lack of gravity as assets instead of antagonists. That dollar bill's fall is determined by G--but no such bound applies in space, so our telerobot operator will find no problem in catching it. "Real time" can go slower in space!

> "Suppose that a tele-operator turns a valve that releases coolant from a reactor. The operator on Earth waits for one second, then realizes the mistake and turns the valve back. One second's worth of coolant loss may not be so important on Earth, but on the Moon, that coolant may be irreplaceable."

This problem has little to do with time-delay. Even an astronaut right at the scene would lose some of that coolant. One answer lies in imposing appropriate protocols--for example, to slow down and institute safety checks, such as *"do you really want to do that"* dialogs. But every computer user has learned that no such schemes ever keep working for long. It is easy to talk about slowing down--but can we train people to work both slowly and reliably? This problem, again, is not one peculiar to teleoperation. Human attention is hard to maintain through any slow and uneventful task. We see the results right here on Earth, with oil tankers running aground. The problem is one of vigilance. Indeed, in the case of remotely-manned operation, we could always shorten sessions and rotate crews, because so many terrestrial workers are available, whereas that option is rarely open in space--and there is no way to maintain perpetual discipline among a limited staff of weary, overworked astronauts. We just can't expect to find a way to make people maintain constant vigilance. Ultimately, this problem can be solved only by more automation.

*Even if people could be made to slow down, won't things then take too long to be practical?* The cost-advantages of remotely manned operation would seem overwhelming because, as we argued earlier, even if the work proceeds 10 times more slowly, the on-location productivity of each telerobot will rival the equivalent of a full EVA work-week--at a hundred-fold smaller cost of launch.

*Why do certain ranges of time delays seem especially disturbing?* We have encountered peculiar difficulties in dealing with telepresence time-delays in the range between 0.05 and 0.5 seconds. For example, most people suffer a peculiar experience when they try to speak coherently through acoustic feedback systems that return an echo a fraction of a second later. This often results in a devastating stutter, in which the subject repeats many syllables. We see similar phenomena with delayed visual feedback. For example, when you try to write on a graphics screen, through a system with the same order of time-delay, you find yourself stuttering now with your hand--by repeating the loops in some characters. I have the impression that these aberrations are most disturbing for small delays, but they are not so apparent with larger delays--such as those we'll encounter when working in Space. I suspect that this could be because our brains have evolved specialized ways to deal with internal delays of those magnitudes; then these "stroboscopic phenomena" might arise from sub-systems of the brain having corresponding time constants. Consider that one would expect various parts of the brain to contain machinery specialized for comparing what actually happens, after each motor action, with what was planned or expected to happen. Schemes of that sort would seem quite indispensable, for providing information both for controlling, and for learning about, coordinated sensory-motor activities. Now each such mechanism can be presumed to engage some type of memory buffer and some variety of time-gated comparison scheme, both actuated at intervals comparable to that of that particular brain-system's sensory-motor loop-delay. (If such comparisons proceed at regular intervals, they might be detectable as local brain-wave frequencies.) Consequently, we might well expect "stuttering" types of disturbances whose magnitudes peak when those sensory cues are delayed by those singular intervals. But these special effects ought to weaken in strength with larger delays, and perhaps disappear entirely, when larger delays force workers to switch over to other, more deliberate modes of operation. If this is right then, paradoxically, we might get better performance using the 1-second GEO relay system than with the faster, more expensive 1/7-second earth-based relay alternative.

# Space Travel for the Next Millenium

Theodore Taylor

Transcript of invited talk for Vision-21 Symposium

For some eight years there were about two dozen of us at General Atomic at San Diego who were literally packing our bags to go off and explore the solar system. Maybe one way of summing up what we saw in our immediate future was our goal for 1970–which was Ganymede. We saw a way of getting from Earth anywhere in the solar system fast in spacecraft that were roughly one-third engine, one-third propellant and energy source, and one-third payload. We designed a lot of vehicles, and they all came under what we called the Project Orion. The project started the night of the announcement of Sputnik 1. It was an effort by a few of us to, as it were, recover our lost face because the Russians had been putting zoos in orbit and we were struggling trying to get something up there. We said "What do we really want to do?"

I want to come back to that question a couple of times in what I'll have to say, that question of "What do you really want to do and what is in the way of doing it?" And, if there is nothing in the way, then do it. The basic idea of Orion actually evolved out of a thought that Stan Ulam, the co–inventor (along with Edward Teller) of the H–Bomb, had (apparently) the day after the trinity explosion at Alamogordo. He was thinking of propelling things at ICBM velocities. This was in 1945– "Getting up to those speeds takes a huge amount of energy. This is the most energetic think we've got, what we saw yesterday. Is it crazy or not to think of a series of explosions that come from nuclear explosives that are carried on some kind of a thing that, somehow, explode behind it and it goes up to whatever speed is of interest?"

I've always been a space buff, ever since I can remember. And it didn't take an awfully lot of thrashing around that night in October 1957 to say that Stan was right, that that was the way to go. There followed a lot of things that were unbearably exciting and particularly unbearably difficult to give up when the project died about eight years later in 1965. I just want to say a few things about that. Mostly in the context of what we thought was a very real vision of our future. Most of us working on the project were in our late 20's, early 30's. I was 32 when it started; I was 40 when it died. And we saw our future very clearly– go out there and explore it.

It took us about six months to find money beyond a rather plush General Dynamics in those days. They gave us the resources to get going and put together a fairly persuasive proposal on what to do. At that time NASA didn't exist yet and it wasn't at all clear who was in charge as far as space activities were concerned. To make a long story short, Roy Johnson, who was the first director of ARPA--what was then the Advanced Research Projects Agency in the Pentagon--took hold very hard the first time we went in to see him. He had just hired Herb York as his Chief Scientist and Herb was an old friend from Los Alamos days, and so on. So we started a formal project for ARPA in July of 1958–a million dollars for one year.

That one year was packed with excitement, and I think some real accomplishments. One of those accomplishments was the uccessful flying of the first object that, as far as I know, has ever flown that way–and also the last that has ever flown that way. It was a one meter diameter model which had five charges, of about 2 pounds each, of high explosive inside. It fired them out sequentially– 5 of them– and it got up to about 200 feet and a little parachute opened and the thing came down.

A key thing happened as we got going. This was very soon–a few days–after Sputnik was announced. Freeman Dyson, who was at the Institute for Advanced Study at Princeton and with whom a number of us had worked on a whole variety of things at General Atomic, heard about this and it took him about five minutes to decide "this is it". He took a leave of absence from the Institute and came to General Atomic, and made a huge difference in what happened there. I can't resist saying that at a point when it wasn't clear what was going to happen to us, he had to make a decision; and that was whether to continue to be a very good theoretical physicist ,or to switch and become what might be recognized eventually as the greatest engineer, ever.

Well, he was one of the key people in that project for about eight years. The idea is basically fairly simple. The idea is to carry several thousand nuclear explosives inside, stacked up about half way to the end of that thing (they weren't H–bombs, they were fission bombs). And then to fire those sequentially, typically at a rate of about one a second, at a point about a diameter and a half from the bottom from the center. The nuclear explosives were not spherical A–Bombs. They were shaped charges. You can show on the back of an envelope that if you try to enclose a nuclear explosion in anything with any structure it will blow it to pieces.

So, we focused on the explosive charge itself, trying to conserve momentum and direct as much momentum as we could through the solid angle that was subtended by the bottom plate, which we called the pusher plate. This was a metal plate–we finally wound up settling on aluminum. So the explosion would go off and slam this plate upward with a speed of something like 15 or 20 meters per second. Then the problem was how to connect that with the structure. From the very beginning we designed for at least a half dozen people up front in this thing. It was always, always–without exception–a manned space vehicle. So, we had to cushion the shock and there is an analogue here which I think is accurate. It is sort of like a car riding along on a rough road. First you need tires. The tires were toroidal gas-filled assemblies on top of the aluminum plate. Then there had to be something which was the analogue of wheels connected to shock absorbers. That was a structure just above those toroids connected with some long nitrogen filled shock absorbers. Then, at the top of the shock absorbers, things had been smoothed out so that the ride was cushioned. Just pretty much like a car.

Above that, depending on the way in which this shock absorber–tire system was driven, you'd either get pulses of a peak of something like 4 g's up front or–this is what we finally settled on–you could drive it at resonance so that you'd squeeze the pusher plate up–into the tires and the shock absorbers. And then as it bounced you'd stop it and return it. Since you're talking about 2000 pulses to get into orbit ,we were sure that some of them would fail, so we had to arrange that the pusher plate would be stopped and pulled back every time there was a failure to restart the cycle. That took a lot of doing.

What we focused on principally for most of those years was something that would take off from the Earth's surface; it was 135 feet in diameter, gross weight 4000 tons; payload through a very difficult mission, brought back to Earth orbit (we wouldn't bring it back down to the ground) about 1000 tons. The idea was that we'd mount this on some towers a couple of hundred feet high, probably from Jackass flats or Yucca flats in Nevada. And then start of with some very low yield explosions, because the air mass between the explosions and the bottom of the vehicle acts like propellant in a way that's a little bit like a ram jet. So while you're in the sensible atmosphere the yields are quite low. It turned out that to get up and out of the atmosphere took about two hundred kilotons of total yield. In those days most people, certainly people in the business of nuclear weapons, weren't particularly concerned about fallout. The reason that we didn't worry about it, in the beginning at least, was because 200 kilotons was to be compared with several *megatons* of fission (half fusion/half fission) in the big H bombs we were setting these off all over the place, mostly out in the Pacific. The Soviets by that time were, too. So we said ,another 200 kilotons for each flight? Who cares?

The performance of this thing was 4000 seconds nominal specific impulse, about 40 kilometers per second effective exhaust velocity, which depended critically upon how well we could shape the charges. It began to look less and less crazy the more we looked at it. By the end of that first year a lot of people were taking it very seriously. It was all secret–some vague descriptions of it were made public, but it was generally not much in anyone's consciousness except for maybe 50 or 75 people in the United States. We kept going and got more and more persuasive that this wasn't crazy and, in fact, was something that could be done. About 1962 NASA asked us to do some mission studies. NASA had not become involved in the project, even after NASA was formed, because basically nobody in NASA knew anything about nuclear explosions. And this was at the heart of what we were doing. But we did start some mission studies that were very important and gradually moved away from some of the really outlandish versions of this that we started out with. In particular we did start worrying about fall-out . What we settled down to was a 34 foot diameter set of modules (the shock absorbers) about half the charge (propellant systems as we called them–we had a tendency to call them bombs but they were shaped pulse units) and then the payload.

The idea was to put each of these parts, in sequence, on top of Saturn V and put them in orbit. Then assemble the whole thing. The number of packages depended upon what you wanted to do. Our favorite mission by that time was the round trip to Mars in 250 days: roughly 30 kilometers/second mission velocity. We dropped the specific impulse to about 2500 seconds and what we wound up with was a departure weight from orbit of about 500 tons. Then two components of the payload–roughly fifty/fifty: 70 tons to be left at Mars and 70 tons brought back, witg a crew of 8 to 12 people. It looked as if that was going to happen in about 1962.

Then along came the Nuclear Test Ban Treaty with the Soviet Union, which forbade any nuclear explosions except underground. We reacted to that with a proposal that was actually made by Neils Bohr, who was invited to be the principal person at the dedication of General Atomics' very fancy laboratory in La Jolla. Marshall Rosenbluth and I spent a whole night to –I don't know, 4 o'clock in the morning– hearing Bohr as best we could (because his English was terrific,but he mumbled. I don't know anybody who ever had ease in listening to Bohr talk, whether it was in Danish or in English.) In any case, he poured out this passionate feeling about having tried to get Stalin and Churchill and Roosevelt, before we built the bomb, to agree that it would never be built. And he failed. When he heard about Orion we couldn't tell him about it in any detail because it was secret and he was a Danish citizen. But he decided that it did make sense. So he decided that what we should do is go to the Russians and say "let's do this together". Now this was opening up the door to the Solar System really wide open, but it was also to get rid of all our bombs. So that double attraction got Bohr very strongly promoting the idea of a joint project with the Soviets.

When the Test Ban Treaty came along we proposed exactly that. But we said that there are still some loose ends in this and we have to do some testing. If we can't do it in space we will do it on the ground. Not repeated flight tests underground, but there are some key questions remaining after a lot of experimentation, mostly with high explosively-driven lead plasmas that we used to mock up the conditions of stagnating debris–the propellant we call it–against the bottom of this thing. The key question was, "what's to prevent heating up and essentially destroying the whole ship?" And the answer was very simple–pulse. Pulse everything. If a glowing ember ever pops out of the fire place, you don't pick it up and put it back in the fire place–you just flick it. And the reason is you can deliver the same momentum in a very small fraction of the time of actual continuous pushing. And during that time, heat flow is strongly inhibited at these very high temperatures (about 100,000 degrees Centigrade) by a build-up of an opaque layer of whatever it is that it is slamming up against. That opaque layer is very protective. Just like moistening your finger to test a hot iron–the same general idea.

So, pulsing and controlled ablation came to be the answers in great detail to the question "Why doesn't this whole thing burn up?" That concept needed some testing with a nuclear explosion and we proposed in detail how to do this underground. For about three weeks in 1965 there was a joint decision by the Defense Department, the Atomic Energy Commision and NASA to proceed for three years in what we call an Engineering Practicality Demonstration Program. And assuming it was successful, and we presumed it would be, then to go to the Russians and say "let's do it together". There actually was for three weeks a decision to do that!

Then the whole thing started becoming unravelled. I counted 13 sort of fancy committees that were called together to review all of this in detail--the Air Force Advisory Committee, several committees of NASA, a couple of ad hoc ones, Congress looked at it– and nobody recommended stopping it; and more than three quarters of them recommended going full blast. But, a large part of the aerospace engineering community, some in the government, some not, said "Look we've got to learn how to walk before we run. This is really running, and running fast. That's great, but we've got to walk first".

For example, proceed with the nuclear rocket. I think the people in the nuclear rocket program felt threatened by Orion. So the net result was that the first person to really be convinced that this was really kind of crazy to proceedwith, was Jim Webb. And, so NASA fell out of bed. Very quickly the Defense Department did. Harold Brown was very nervous about this thing and I think was greatly relieved when NASA decided to pull out. The AEC interestingly hung on a little bit longer, but then they finally dropped out and so the project died.

I'm not describing this in a little bit of detail because of any strong urge to revive Orion, in that form, at least, but because I think it's important to know what it feels like to be planning, in the next 10 years, to go out and explore space in a huge scale. And that is an experience that few people have actually had. What comes from that is a vision of a future. And that vision I'm finding is coming back with some important changes right now.

I want to spend a few minutes talking about that vision. What I'm going to do is to just sort of tersely present some features of the world which this modified vision (it has some connections with Orion) consists of. I don't expect to be persuasive. I think most of what I have to say is provocative–some of you may find it very provoking– but I feel this so strongly that I think I need to get it out.

These are not predictions. Neils Bohr said many things that are very wise. One of them was "You can't predict the future, especially when it hasn't happened yet." So, these are possible features of the rest of the 90's and the early parts of the next century.

The first is that I see coming a global consensus about how the rest of a vision like this may actually come to reality. I see that happening because of the enormous urgency that it does happen before the end of the 1990's. Pretty much world-wide, there is a sense of what to do and how to do it to avoid what could be extermination of the human species–nuclear war, a big one, or, whether we have a nuclear war or not, to ruin our habitat, just ruin it, if we continue what we are doing now globally. So, number one in this vision is a consensus about what to do about all this in a lot of detail. Much of this work has been done, but there's a lot that remains to be done. And I think it had better be pretty clear before the end of the century what this is going to be–we've got 9 years.

The second is it has to be clear to most human beings that having large families is not, as it has been traditionally among most parts of the world where large families still appear, not a way to achieve security. This has to get out: that in fact more children, like more nuclear weapons, make you not more secure but less. I think there has to be coupled with that something that's technical (bio–technical) and that is a really satisfactory method of birth control. For starters, I'd focus on males, just because there's been much less focus than on females.

The next characteristic of this vision is that the threats of wars, particularly indiscriminate wars of retaliation with weapons of mass destruction, will be much less than they are today. As far as nuclear weapons are concerned, for years now I've been a staunch promoter of abolition altogether, as soon as possible. A lot of that can be verified by all kinds of measures, but not perfectly. We can't do anything perfectly, but we need a global taboo that it is absolutely repugnant human behavior to be any part of acquiring or maintaining nuclear weapons of any kind. I think that needs to shift over to biological and chemical weapons as well. This whole idea of deterrence by maintaining a situation in which a country can flat-out murder a large population of people that have nothing to do with the decision to proceed with an attack--I find that monstrous. And I think that there's a very good chance that public pressure world–wide will bring that about. I don't see that coming universally from government leaders. I certainly don't see it coming from the leadership of the United States. There have been–some people say somewhat half–hearted attempts by Mikhail Gorbachov and later Rajiv Ghandi, who is now out of power of course, to press for that ,but a lot of that has been ignored and laughed at. I think it is dead serious and we had better DO IT.

Unfortunately, abolishing nuclear weapons but pressing for and expanding nuclear power seems to be incompatible. The reason I say that is that right now in about 40 countries in aggregate there are about 100,000 tons of spent power reactor fuel. The plutonium in that fuel can, counter to popular wide spread opinion, be used for making efficient, light–weight or heavy–weight nuclear warheads of all kinds. The total quantity of that plutonium is about 1000 tons, which is roughly five times the total amount of plutonium in all the world's 60,000 nuclear warheads. So how is it that we are going to arrange things so that a couple of countries, let's say the United States and the Soviet Union, continue to maintain 200, 500, 2000 minimum deterrent (so–called) nuclear weapons in a world in which the hardest part of making the nuclear weapons has been done in 40 countries with these huge numbers and somehow say "it is good for us but not good for you"? "You can't have them, we can have them!". That doesn't fly if you talk to a few Indians or Pakistanis or Brazilians or Argentines or Mexicans or Iraqis or Iranians or whoever that don't have them or even some people that do but secretly (like Israel). And the idea that somehow the two superpowers (or really five) can continue to behave as though they are much safer with nuclear weapons than without them but nobody else can do the same. How do you enforce that in a world that is awash with plutonium? That's one problem I've never seen addressed in such a way as to say it really is solvable.

What I see happening is that for a lot of reasons, but the main one of which is the weapon connection, we'll find that we're not mature enough yet as a species, and may not be for a very long time, to handle wide scale use of nuclear power.

The focus of what we must do, and do vigorously, has to be ways to find how to live in harmony with our environment and with each other to the extent that we can. And yet meet basic human needs in ways that are pretty much accessible to most of a population of a little over five billion and probably seven, eight, nine billion. And then I would hope we would taper off and maybe come back down a little bit.

Now, what about space? Number one on my list is that as soon as it can be arranged between those countries that are now active in space that all activities in space are internationalized without exception. And I include a lot of what goes under the name of military activity, which I think is very healthy, that has to do with keeping an eye on what's going on down below. I guess I could sum up one version of the Open Skies Proposal as far as surveillance and so on is concerned–satellites and other means– is to go back again to Orion days when Harrison Brown made an impassioned plea that we continue to expand what we can do in space with satellites but that we take every bit of raw data and every bit of processed data and put them in a sack–it turned out to be a very big sack– and put it on the front steps of the U.N. Building every Tuesday afternoon at 4 o'clock. No secrets. No secrets in space activity. Why? Secret activity in space is extremely threatening, and I see no way for that to change. We've got so much to do to clean up the messes which we have left behind that secrecy is something that maybe whould be taboo also. In other words, superglasnost.

So far as how things in space get done is concerned, if one accepts the idea that everything is basically internationally organized and carried out, then that calls for intimate cooperation between countries that choose, have the resources, have the will, to do whatever it is: go to Mars, go to Mercury, get out to the major planets. The value of doing some things in a big way, particularly connected with surveillance, in space is that there's a strong connection with arms control and disarmament, and that is verification of disarmament and arms control treaties. Trevor Gardner, who basically started the U.S. missile program, hawkish as he was, proposed an international set of arms control satellites. He proposed this strongly in 1960, 1961, and pretty much until he died in the late sixties.

Now, how to get into space and to how to move around in it? A great deal has been said, some of it I think absolutely fascinating at this conference, about all kinds of different ways to go beyond what we have done; beyond in the way of developing not just propulsion systems but whole transport vehicles and the infrastructure to support them and everything else. I just want to pick out two tasks which I think it's clear we can accomplish in such a way that those who wish to can participate in space activities on a huge scale. The first is how to get into orbit.

There is a kind of holy grail out there, and I have no idea how to take hold of it, tied up in free radical chemicals. If we could store atomic hydrogen in some stable way that we were quite sure couldn't explode, we've got a good solid 1700 to 2200 seconds of specific impulse. That's enough for all the high thrust things that I could think of doing–landing on the moon, getting up from the Earth, doing a soft landing on any place that we want to land on: Mars, Venus, Mercury (particularly) and all the moons, and so on. So we can carry along high thrust chemical propellant that's really up to the job.

Now until that time happens, I'd like to revive something. I was just talking to people here at the conference who are of the same mind: we ought to go back to what was being proposed in the late fifties, the early sixties. And that is, pick out some launch vehicle. I would argue for that launch vehicle being hydrogen and oxygen, from the ground up. That's not a new idea. There was a serious set of firm proposals to build single stage hydrogen oxygen launch vehicles to go into low Earth orbit and you can go through the arithmetic, anyone can do that, to overcome gravity and drag losses and so on. You can put somewhere around ten to fifteen percent of the launch weight into low Earth orbit. Of that weight maybe a third will be tanks, structures which are now in orbit. By tethering and all kinds of other things they become now a resource to be used in space. Trying to get the main weight of boosters back down for refurbishing is silly if you can get them up there and leave them there to use for other things.

Now, there are some missions in which what you really want to do is go very fast up to escape velocity and not orbit. For that. one very real possibility is a two–stage hydrogen oxygen rocket. The first stage moves things along at about 6 kilometers a second, and the second stage adds another 6 kilometers a second and there you are just barely hanging on– maybe at a libration point. But you've gone, in effect.

But now, how about moving around once you're up there? You don't need high thrust. I think that it's quite clear that the big winner is going to be solar electric propulsion. And that will do everything that we might want to do at low cost, very fast, out to about the first one third or so of the asteroid belt. From beyond Mars, all the way into the sun. What might this look like? Well I was astonished to see a diagram downstairs of Geoff Landis' bicycle wheel solar array which I think is a specific embodiment of how to go. This is just like a bicycle wheel–it's got spokes and it's got the analogue of a rubber tire on the outside, two kilometers in diameter. In that roughly three square kilometer interior are very thin film photovoltaic cells that are (at Earth orbit) picking up about 1350 watts per square meter. It seems to me quite fair to talk of those thin film cells, whatever they are, in the near future having efficiencies of 20 percent.

That electric power then goes into ion thrusters. That's not my field. I keep picking up things from people, particularly from the Soviets, about how efficient and light–weight those can be. You'd like to get into the ballpark of several kilowatts of input energy per kilogram of thruster. Now you can certainly do that for the solar cells. In fact, it's quite credible for a structure that is stable to deliver 10 kilowatts per kilogram of solar cells . I am not saying we know how to make the structure. Back to the bicycle wheel you see, the ship itself is the analogue of the rubber tire around the wheel. It rotates–this is a one kilometer radius–to give the ship and its contents about a quarter "g". Now that means it rotates once in about 90 seconds. And the payload: the people, the shielding from solar flares or whatever, is all out there. I am sure that a lot of people here could decide within less than a day, where to put the propellant tanks for the ion thrusters. My guess is that the best place to put them is at the hub. The thrusters, at least most of the thruster capacity is also at the hub; but able to point over the best part of 180 degrees on either side so that the solar array faces the sun but the thrust is whatever direction you want.

What kind of performance can one get? Well, if you really look at what's out there now in terms of the weight of substrate on which these solar cells can be deposited, it's a thickness that can easily be less than mil--a thousandth of an inch. The penalty for going to very high specific impulse is lower accelerations, so it takes longer to get up to speed. But, if you'd settle for , let's say going out to Mars or, what I get much more excited thinking about, going in to Mercury (I think Mercury is a lot more interesting than Mars, but that's a separate question) to pick up about 15 kilometers a second you can do that in a week, two weeks. Your talking about roughly a milli–g, maybe a couple of milli–g's.

If the exhaust velocity is somewhere equal to or maybe twice the mission velocity, then you're talking about mass ratios of maybe two. And a division of weight between roughly four -enths payload, two-tenths essentials, for the space craft that are not really connected with the payload. And then the rest is tanks and engines and the photovoltaic system. One can see these things not being very mission oriented. All they need is propellant and you go wherever you want. Reusable. How long? Who knows? But I think we could find out very quickly how long thin film photovoltaic cellswill keep operating more or less the way they are supposed to. You can fill them full of holes–one percent hole–and you've lost one percent of your electric power. At ten percent holes, which is a lot of holes, you have to worry about short circuiting and other things. I don't want to trivialize the problem, but it's not clear at all that there is any really severe, basic problem in doing this from an engineering point of view, and certainly not as far as the basic science is concerned. Rule of thumb: mission velocity about half the ion beam velocity.

That is fine out to somewhere between Mars and Jupiter. What do you do about the major planets? Orion could do it. I think there are a lot of side things about Orion. The one I worry about the most is athe potential for destructive use of anything that carries 5000 nuclear explosives, which is what it takes to make a fast round trip to Ganymede, Titan, what have you. But, I think the answer is probably some form of nuclear power. Maybe thermonuclear power. It may be stretching things a little bit to consider thermonuclear power from Helium-3 and deuterium (which produces no neutrons, which is a big help.) But whatever that it is, a point of departure for thinking about nuclear propulsion beyond the asteroid belt is a constraint that for fission power or fusion power in which there are lots of neutrons, they go to clean cold starts way out there. Then they go back and forth and pick up payload; we can certainly get on our way to Pluto at very high speed. The problem is what do we do when we get there? We can break various ways. You've thought about that a number of things at this conference.

But, then how do you get back? And I think the answer is probably going to have to be nuclear. Another possibility is laser beams, generated from ferociously potent solar panels, let's say at Mercury where you get six times the insolation that you do on Earth. Unless we can somehow get around the laws of optics, you are stuck with roughly a kilometer aperture for something with a wavelength like sodium light–which may turn out to be possible. Then you can just beam energy out there. Something like that thing with pulses may be sort of close cousin to Orion in its original form, where you'd pulse energy on to it.

I am not suggesting that exploring the major planets and going clear out to Pluto is something not to think about; it's way off in the future. It may be much closer than we think. It was very close with Orion. But it takes some doing to put things together and if people really get serious about it we'll find ways to explore safely, without weapons connections and all that, out beyond Saturn.

Let me mention one other thing which may be a direct use of space–not exploration–on which everything else can ride. And that is disposal of the actinides, the 200 thousand tons of irradiated reactor fuel that the world is going to have at the end of this decade. We have a hundred tthousand ons already and we'll get a hundred thousand more in the next ten years. You can visualize space disposal of nulcear wastes in a little detail, as has been done before, some years ago. Frank Rom told me about this yesterday.

You set criteria on the launch process that the payload cannot break–no matter what! Just to get a little specific, if you add up the total amount–total quantity of actinides in all this 200,000 tons of nuclear fuel, the answer is about 3 million kilograms of, mostly, plutonium. Suppose you use 200 launches to get rid of all this stuff. You're talking 5,000 kilograms plutonium per launch. You have to use dry fuel reprocessing. That's a whole other story–you can't have a lot of low level waste, and that's not easy, but you can do it. The weight of the plutonium is doubled with things like tungsten and cadmium, for example, that capture high energy fission neutrons and thermal neutrons so that it cannot sustain any kind of a chain reaction. Having done that with this package, that corresponds to 200 launches to get rid of everything. You've got something a little bit less than a meter in diameter. It's pretty hot–I don't mean thermally (what I've looked at says that you can get the heat produced by this easily out without going to very high temperatures). But you do have to seal it with some 10 or 12 centimeters of tungsten around this sphere, which is now 6 feet across.

The next thing you have to do is make it buoyant so it floats. The ideal launch is if you go straight up off an island somewhere out in the ocean, so that if the engine stops, or blows up, or starts going off course, it falls back down roughly a little off to one side from where it lifted off and falls in the ocean. So it has to float. And how do you do that? Well, you add titanium honeycomb and then big heavy case around the whole thing. You wind up kith a total weight of about 50 tons for this package, of which 30 tons is the shielded nuclear waste. That's what you want to deliver out to just barely hanging on by its skin of its teeth. Then you connect with that with a solar propulsion system brake with 30 kilometers a second and drop it right in the sun.

Now there may be terrible flaws with all this. But there's a chance it could turn out to be the only acceptable thing to do with this stuff. I think we can find out whether that's the case in a very short time–before the end of the decade. Then what have we got? The launch vehicles for each of these packages is the 2 stage hydrogen oxygen rockets–maybe we can do better than that. Each vehicle is a little bit smaller than Saturn 5. And 200 of them! We really settle down, which I think we should have done long ago, to pick a vehicle and use it over and over– and I don't mean re-use it, but use the same type of launch vehicle as though you really meant big business. Not go to the moon and then sigh and wonder what to do with the leftover Saturns and so on... we can't do that again! So there just may be something which, of itself, would call strongly for good launch vehicles into high orbit–or low orbit for that matter– and high performance solar electric propulsion. Then everything else rides on that. And a few of these things don't pick up these packages–they go off and go to Mars! Or go to Mercury, or whatever.

All this may sound like Pollyanna. All I can say about that is two things. First, I'd far, far prefer to be pursuing a kind of world that may turn out to be too good to be true than to keep drifting, as we are, toward a world that is just too awful to contemplate. The second thing I want to say is as a guiding principle on how to get to something like this vision of a stable, harmonious future–lots of things going on in space–is to give you the motto of the Pugwash movement. Pugwash was organized originally by Albert Einstein and Bertrand Russell. The first meeting was in the town of Pugwash in Nova Scotia. The drive was to make sure that no matter what happened, at least some American and Soviet scientists would keep talking to each other. Einstein died before that first meeting, but Russell came up with this motto: and you think about it, it's a way to sort of keep steady. Simply: "Remember your humanity and forget the rest!"

Thank you.

N91-22143

OFFICE OF AERONAUTICS, EXPLORATION AND TECHNOLGOY

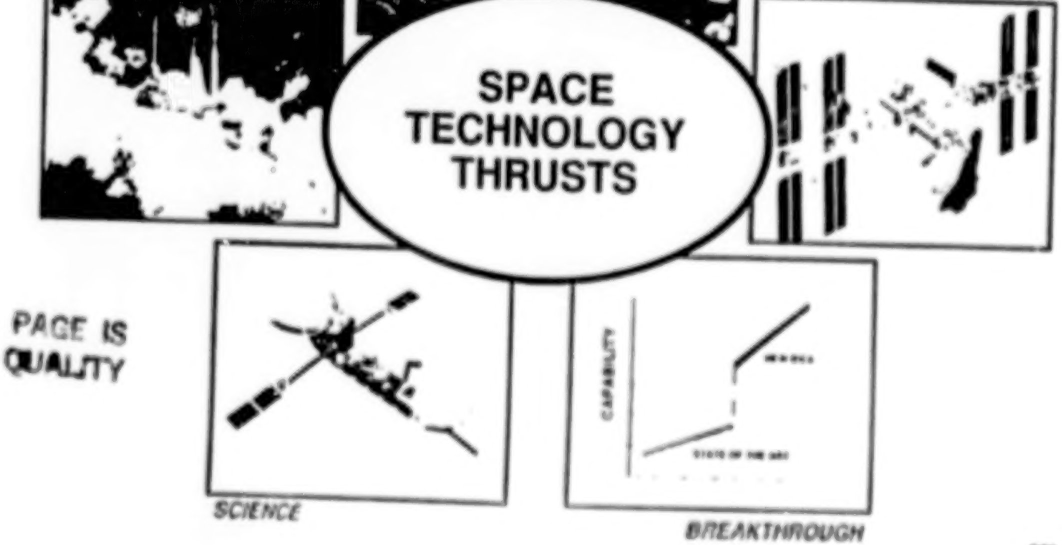
# ADVANCED INTERDISCIPLINARY TECHNOLOGIES

JOHN L. ANDERSON
Manager
Advanced Interdisciplinary Technology

Presentation to
VISION 21 Symposium
at NASA Lewis Research Center
April 3-4, 1990

## Advanced Interdisciplinary Technologies

- BREAKTHROUGH THRUST (Space R & T Assessment)
- BIONICS (Technology Derivatives from Biological Systems)
- BIODYNAMICS ( Modeling of Human Biomechanical Performance Based on Anatomical Data)
- TETHERED ATMOSPHERIC RESEARCH PROBES


EXPLORATION
TRANSPORTATION
SPACE STATION
SPACE TECHNOLOGY THRUSTS
SCIENCE
CAPABILITY
STATE OF THE ART
BREAKTHROUGH

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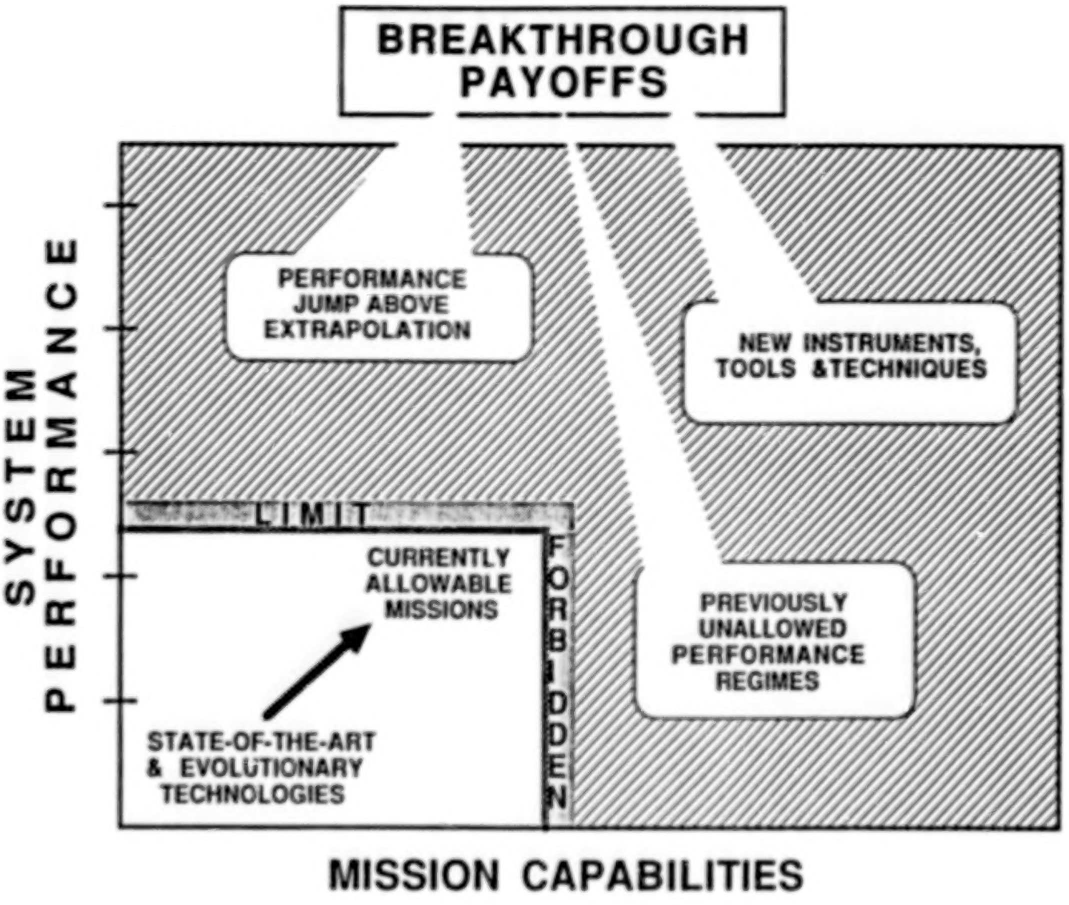
BREAKTHROUGH PAYOFFS
PERFORMANCE JUMP ABOVE EXTRAPOLATION
NEW INSTRUMENTS, TOOLS &TECHNIQUES
LIMIT
CURRENTLY ALLOWABLE MISSIONS
FORBIDDEN
PREVIOUSLY UNALLOWED PERFORMANCE REGIMES
STATE-OF-THE-ART & EVOLUTIONARY TECHNOLOGIES
SYSTEM PERFORMANCE
MISSION CAPABILITIES

## BREAKTHROUGH
### PROGRAM ELEMENTS

**ARTIFICIAL INTELLIGENCE**
- MISSION OPERATIONS
- SCIENCE DATA ANALYSIS
- AUTONOMOUS ONBOARD CONTROL
- LARGE KNOWLEDGE-BASED SYSTEMS
- NEURAL NETS
- DISTRIBUTED KNOWLEDGE-BASED SYSTEMS

**ADVANCED PROPULSION CONCEPTS**
- ELECTRODELESS PLASMA THRUSTERS
- ELECTRON-CYCLOTRON RESONANCE (ECR) THRUSTERS
- HI-TEMP SUPERCONDUCTOR APPLICATIONS TO ROCKET ENGINES
- ADVANCED PROPULSION CONCEPT MISSION STUDIES
- ADVANCED FISSION / FUSION
- REMOTELY ENERGIZED (BEAMED) PROPULSION
- HI-ENERGY DENSITY PROPELLANTS
- ANTI MATTER

**ADVANCED POWER CONCEPTS**
- AMTEC (ALKALI METAL THERMAL-TO-ELECTRIC CONV.)
- FUEL CELLS (IN-SITU REACTANTS)
- LASER POWER BEAMING

**ADVANCED MATERIALS & STRUCTURES**
- ADAPTIVE STRUCTURAL CONCEPTS
- LONG DURATION SPACE LUBRICANTS
- ADVANCED INTERMETALLICS / COMPOSITES (HYPERSONICS)

**ADVANCED SENSORS & PROCESSORS**
- MICROELECTRONIC DETECTORS
- PHOTONICS (OPTICAL PROCESSORS)

**ADVANCED RESEARCH TOOLS**
- COMPUTATIONAL ANALY. & SYNTHESIS OF MAT'LS PROPERTIES
- TETHERED SATELLITE SYSTEM (TSS 2)

# "THE HUMAN FUTURE DEPENDS ON OUR ABILITY TO COMBINE THE KNOWLEDGE OF SCIENCE WITH THE WISDOM OF WILDNESS"

**Charles A. Lindbergh**

## BIONICS

### OBJECTIVE

IDENTIFY ADVANCED, NOVEL TECHNOLOGY APPROACHES FOR FUTURE SPACE SYSTEMS BASED ON PHYSICAL AND CHEMICAL PROCESSES, STRUCTURAL PRINCIPLES, AND INTEGRATED FUNCTIONS USED BY BIOLOGICAL SYSTEMS

### DESCRIPTION

IDENTIFY AND EVALUATE APPLICABILITY OF FORM, FUNCTIONS, AND PROCESSES OF SELECTED BIOLOGICAL SYSTEMS TO FUTURE SPACE TECHNOLOGY REQUIREMENTS.

DEVELOP PROCEDURES FOR DERIVING SPACE TECHNOLOGIES AND ENGINEERING APPROACHES FROM SELECTED CHARACTERISTICS OF BIOLOGICAL SYSTEMS.

### APPROACH

CONTRACT A STUDY TO SURVEY & ASSESS THE STATE OF KNOWLEDGE OF BIONICS APPLICATIONS TO SPACE TECHNOLOGY (AWARDED TO RESEARCH TRIANGLE INSTITUTE - MARCH, 1990)

CONDUCT A NASA-WIDE TUTORIAL WORKSHOP WITH CHIEF SCIENTISTS, ADVANCED PROGRAM AND RESEARCH OFFICES AS PARTICIPANTS

ARC - MICHAEL McGREEVY

# TECHNOLOGY & ENGINEERING DERIVATIVES FROM BIOLOGICAL SYSTEMS (I)

## HYDRONAUTICS

**FLOW EFFICIENCY - REDUCED DRAG**

- O RIBLETS - Shark Skin (Stars & Stripes (S&S)yacht)
- O CRESCENT - Tuna, Whale (S&S keel wings)
- O POLYMER SECRETION - Dolphins
- O BARNACLE GROWTH RESISTANCE- Synthetic Fish Surface Protein Added to Ship Paint

## AERONAUTICS

- O RIBLETS - Shark Skin
- O WINGLETS - Bird Wing Tip Feathers
- O CRESCENT WINGS - Whale Flukes, Shark & Tuna Tail Fins, Swifts
- O SERRATED TRAILING EDGES - Tuna, Swifts
- O VORTEX ENERGY UTILIZATION - Dragonfly

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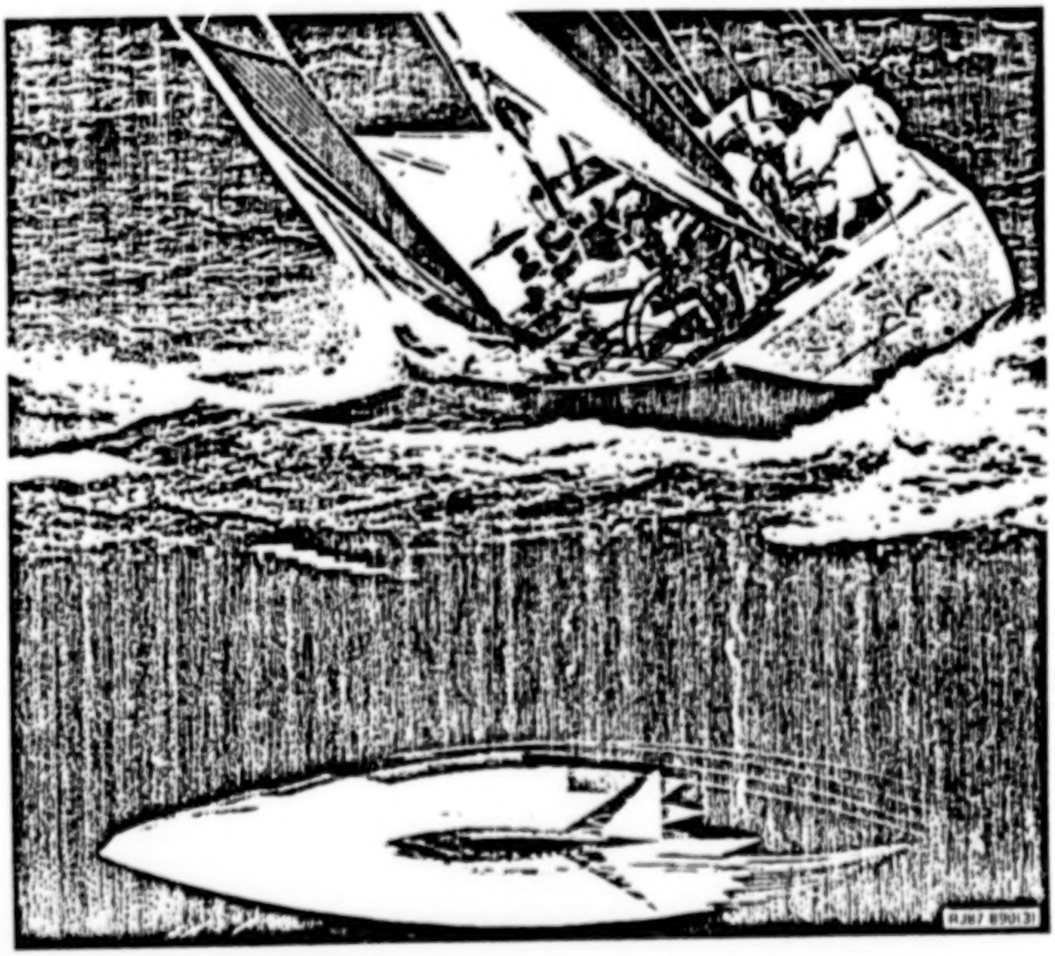
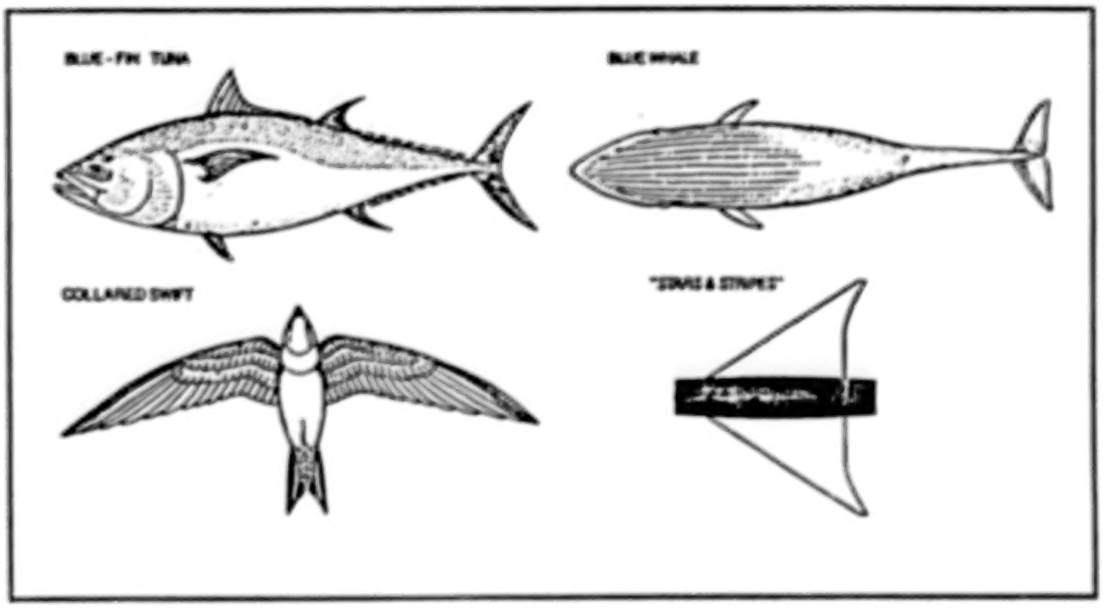


# CRESCENT SHAPES FOR FLOW EFFICIENCY


BLUE - FIN TUNA
BLUE WHALE
COLLARED SWIFT
"STARS & STRIPES"

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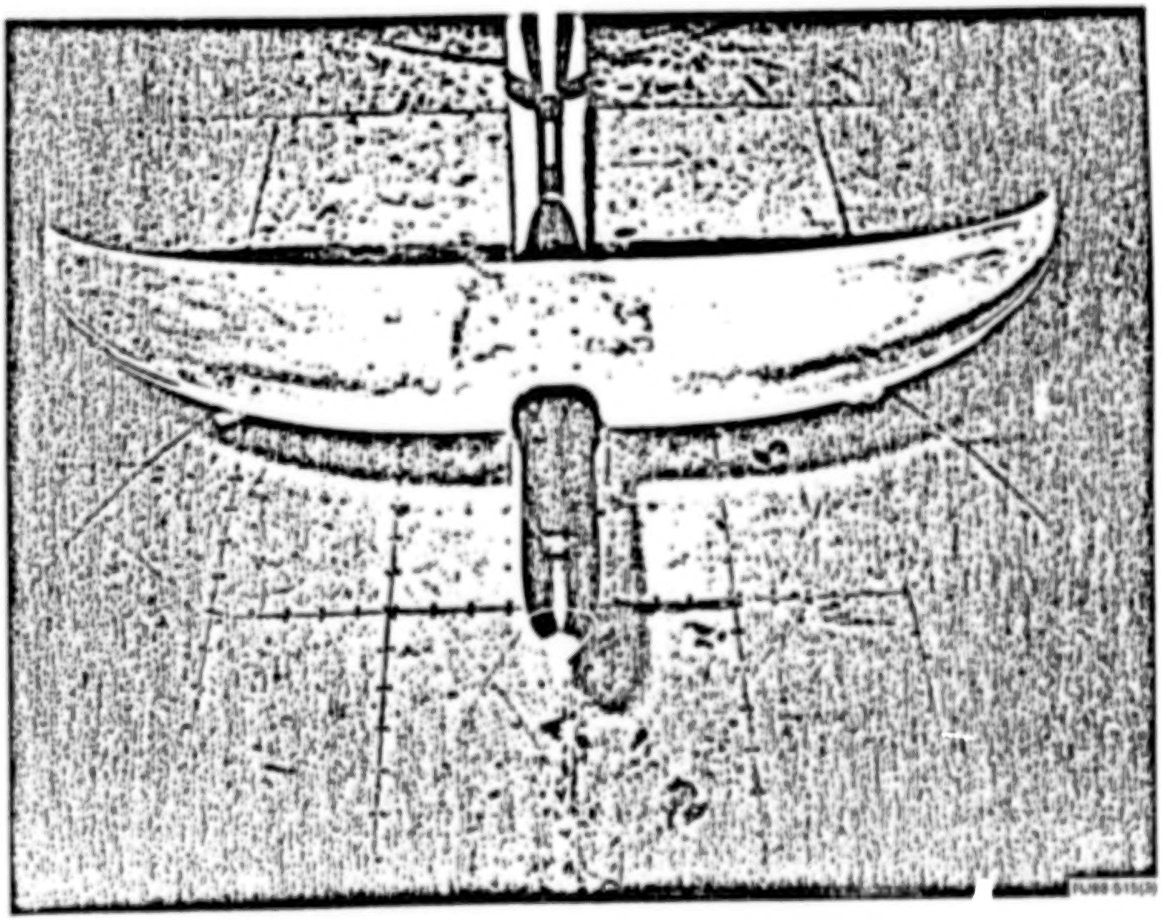
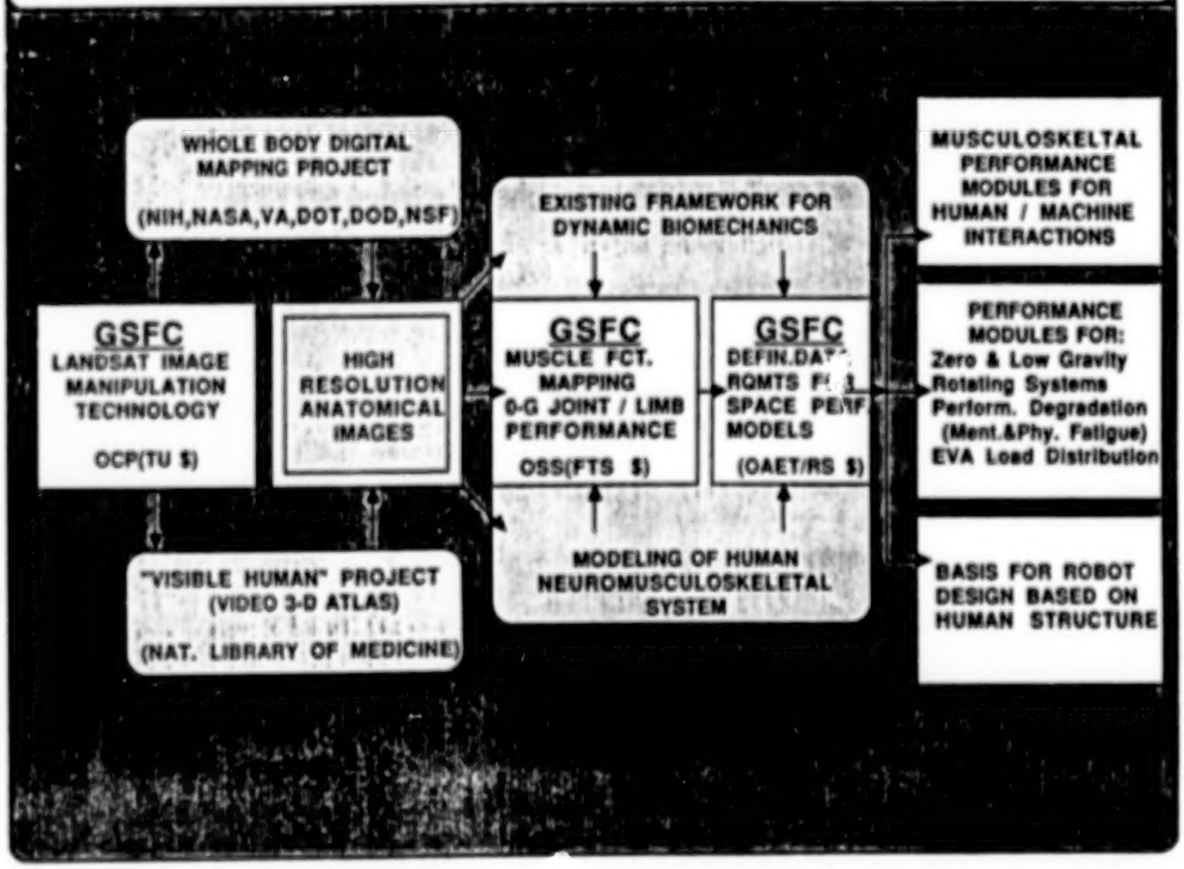


# TECHNOLOGY & ENGINEERING DERIVATIVES FROM BIOLOGICAL SYSTEMS (II)

## ASTRONAUTICS

- SENSORS - Eagle/Hawk Eye UV Protection, Moth IR Detection, Shark EM Field Detection
- NEURAL NETWORKS/PARALLEL PROCESSORS Slug Brain, Rat Inner Ear
- O2 & CO2 PROCESSES - Fish, Aquatic Mammals
- SURFACE CHEMISTRY - (ADHESIVES) - Mollusks, Barnacles
- SYSTEMS DESIGNS (STRUCTURAL STRENGTH/WEIGHT) Deep Sea Fish, Birds
- LOCOMOTION - Insects, Spiders
- CRYSTAL ENGINEERING (BIOMINERALIZATION) - Sea Urchin, "Magnetotactic" Bacteria, Shells, Bones, Teeth

## BIODYNAMICS



WHOLE BODY DIGITAL MAPPING PROJECT
(NIH,NASA,VA,DOT,DOD,NSF)

**GSFC**
LANDSAT IMAGE MANIPULATION TECHNOLOGY
OCP(TU $)

HIGH RESOLUTION ANATOMICAL IMAGES

"VISIBLE HUMAN" PROJECT
(VIDEO 3-D ATLAS)
(NAT. LIBRARY OF MEDICINE)

EXISTING FRAMEWORK FOR DYNAMIC BIOMECHANICS

**GSFC**
MUSCLE FCT. MAPPING
0-G JOINT / LIMB PERFORMANCE
OSS(FTS $)

**GSFC**
DEFIN.DATA RQMTS FOR SPACE PERF. MODELS
(OAET/RS $)

MODELING OF HUMAN NEUROMUSCULOSKELETAL SYSTEM

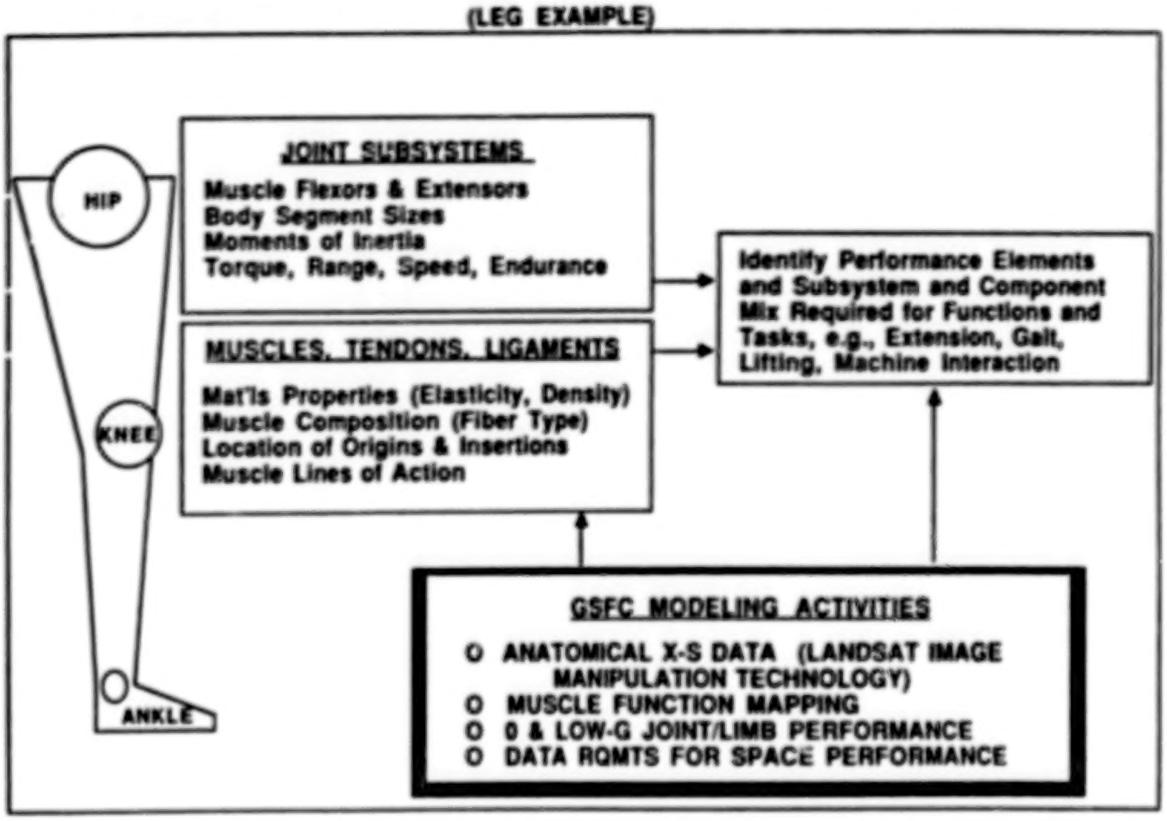
MUSCULOSKELTAL PERFORMANCE MODULES FOR HUMAN / MACHINE INTERACTIONS

PERFORMANCE MODULES FOR:
Zero & Low Gravity
Rotating Systems
Perform. Degradation
(Ment.&Phy. Fatigue)
EVA Load Distribution

BASIS FOR ROBOT DESIGN BASED ON HUMAN STRUCTURE

# HUMAN NEUROMUSCULOSKELETAL SYSTEM ANATOMICAL MODELING OF DYNAMIC PERFORMANCE

(LEG EXAMPLE)


HIP
KNEE
ANKLE

**JOINT SUBSYSTEMS**

Muscle Flexors & Extensors
Body Segment Sizes
Moments of Inertia
Torque, Range, Speed, Endurance

**MUSCLES, TENDONS, LIGAMENTS**

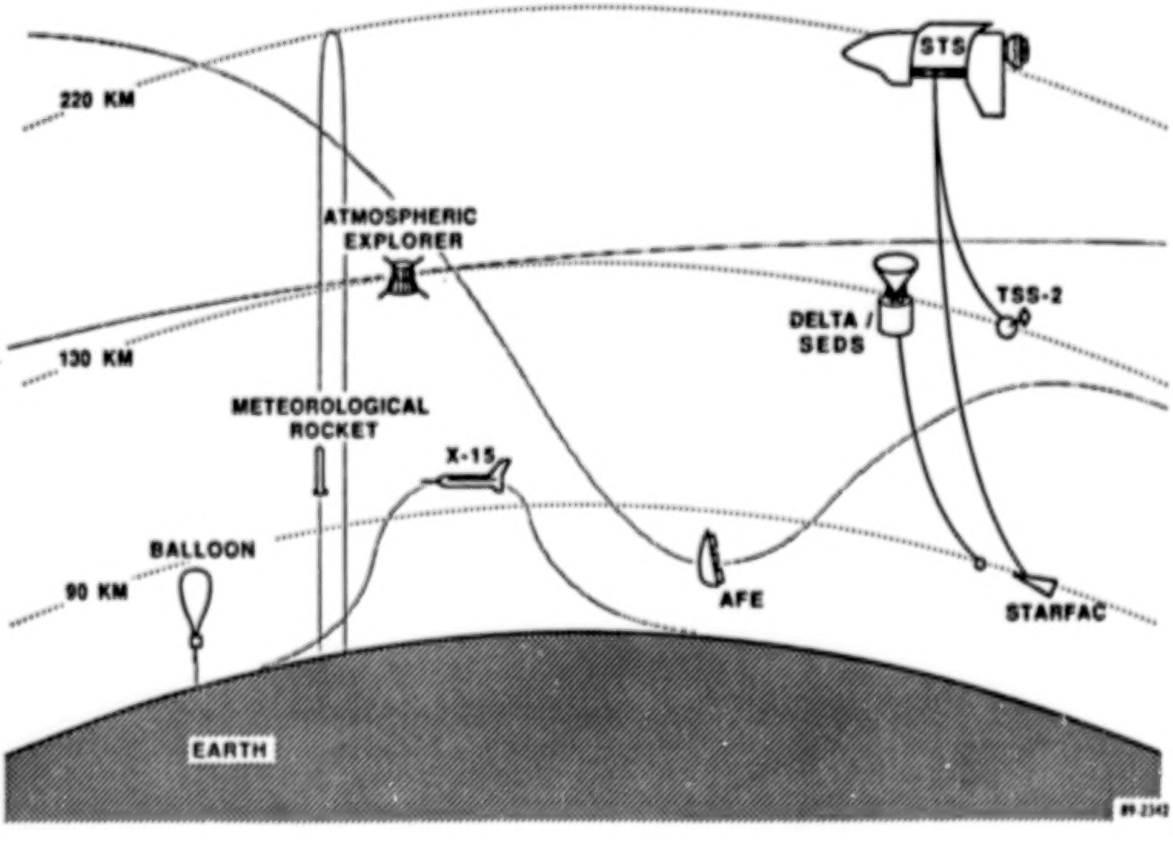
Mat'ls Properties (Elasticity, Density)
Muscle Composition (Fiber Type)
Location of Origins & Insertions
Muscle Lines of Action

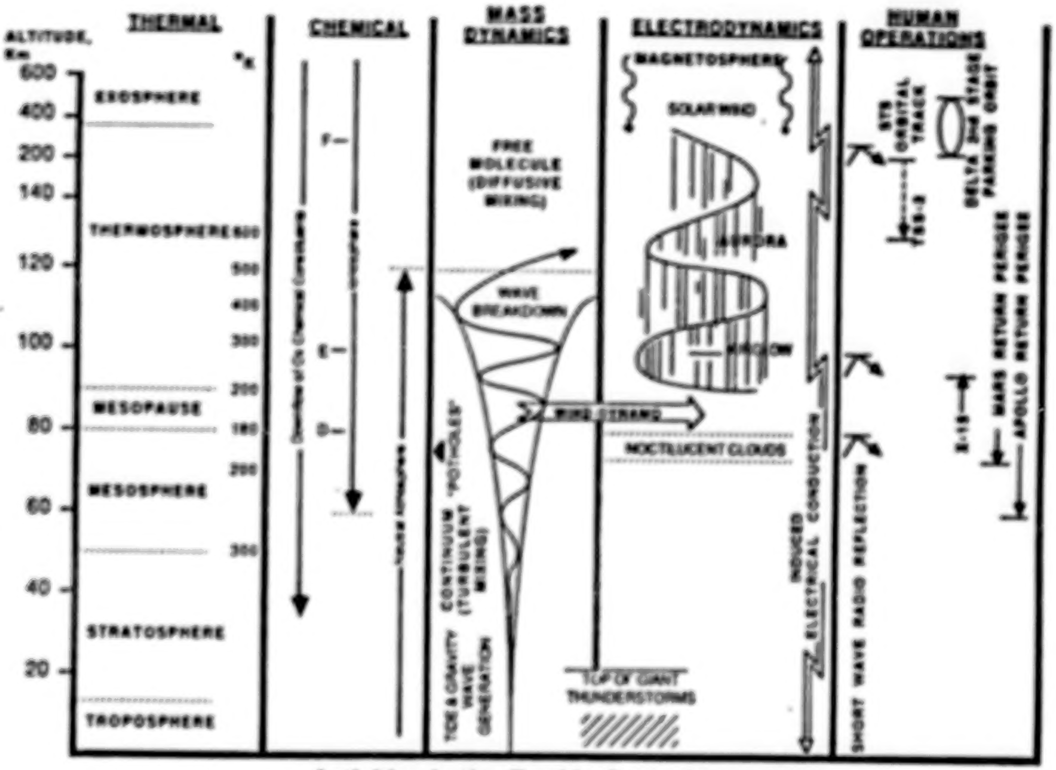
Identify Performance Elements and Subsystem and Component Mix Required for Functions and Tasks, e.g., Extension, Gait, Lifting, Machine Interaction

**GSFC MODELING ACTIVITIES**

- O ANATOMICAL X-S DATA (LANDSAT IMAGE MANIPULATION TECHNOLOGY)
- O MUSCLE FUNCTION MAPPING
- O 0 & LOW-G JOINT/LIMB PERFORMANCE
- O DATA RQMTS FOR SPACE PERFORMANCE

## VEHICLE ACCESSIBILITY TO THE OUTER ATMOSPHERE


STS
220 KM
ATMOSPHERIC EXPLORER
TSS-2
DELTA / SEDS
130 KM
METEOROLOGICAL ROCKET
X-15
BALLOON
90 KM
AFE
STARFAC
EARTH
89-2342


THERMAL
CHEMICAL
MASS DYNAMICS
ELECTRODYNAMICS
HUMAN OPERATIONS
ALTITUDE, Km
°K
600
400
200
140
120
100
80
60
40
20
EXOSPHERE
THERMOSPHERE
MESOPAUSE
MESOSPHERE
STRATOSPHERE
TROPOSPHERE
600
500
400
300
200
180
200
300
F
E
D
FREE MOLECULE (DIFFUSIVE MIXING)
WAVE BREAKDOWN
CONTINUUM "POTHOLES" (TURBULENT MIXING)
TIDE & GRAVITY WAVE GENERATION
WIND DYNAMO
TOP OF GIANT THUNDERSTORMS
MAGNETOSPHERE
SOLAR WIND
AURORA
AIRGLOW
NOCTILUCENT CLOUDS
INDUCED ELECTRICAL CONDUCTION
STS ORBITAL TRACK
TSS-2
DELTA 2nd STAGE PARKING ORBIT
X-15
MARS RETURN PERIGEE
APOLLO RETURN PERIGEE
SHORT WAVE RADIO REFLECTION

Activities in the Earth's Outer Atmosphere

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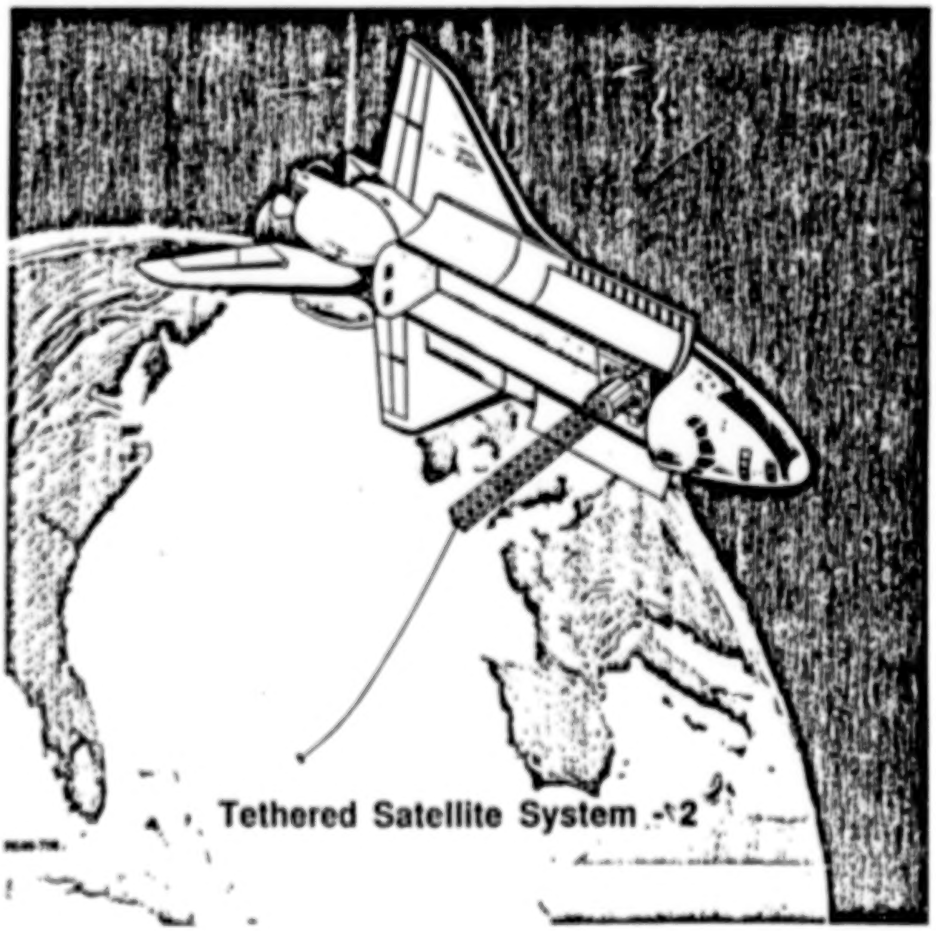

Tethered Satellite System - 2

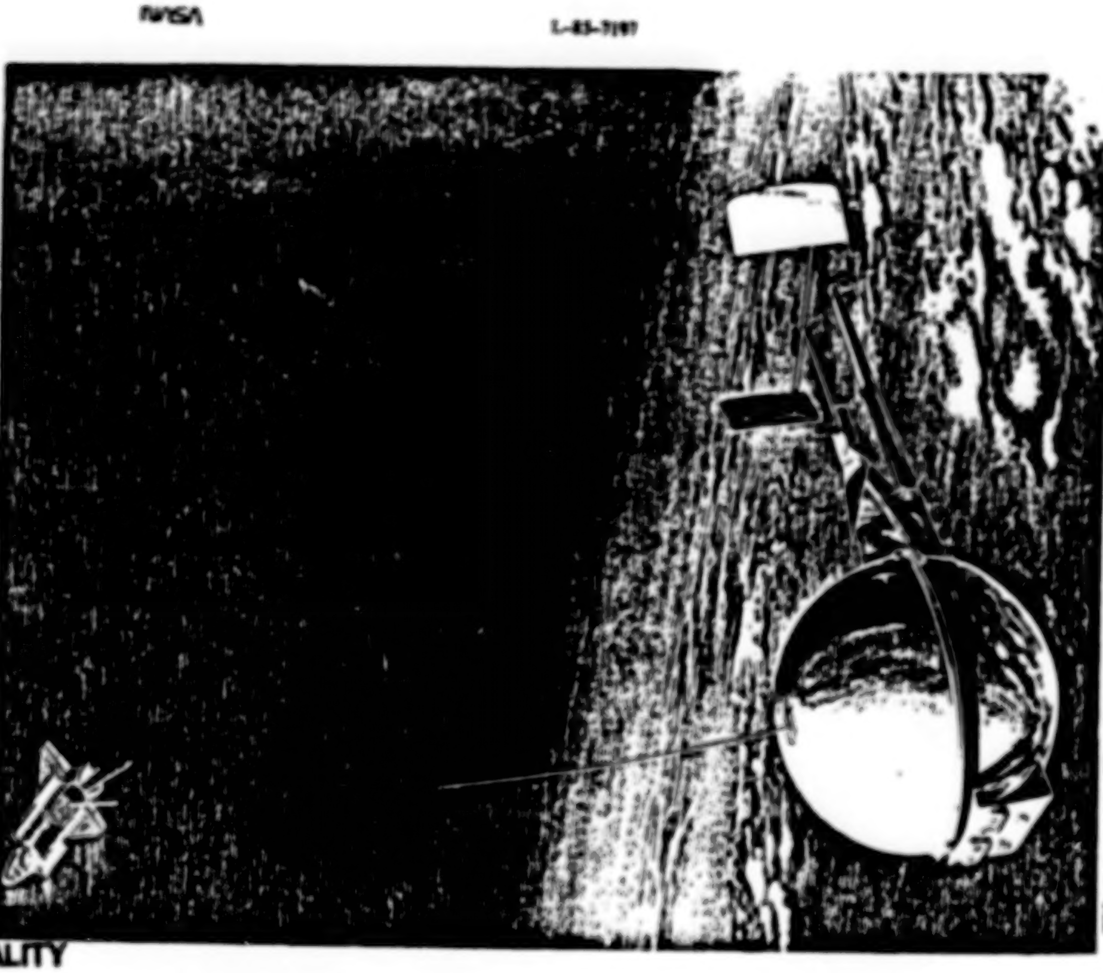
NASA
L-83-7197

Figure 1. Tethered Satellite System (TSS-2) (Concept)

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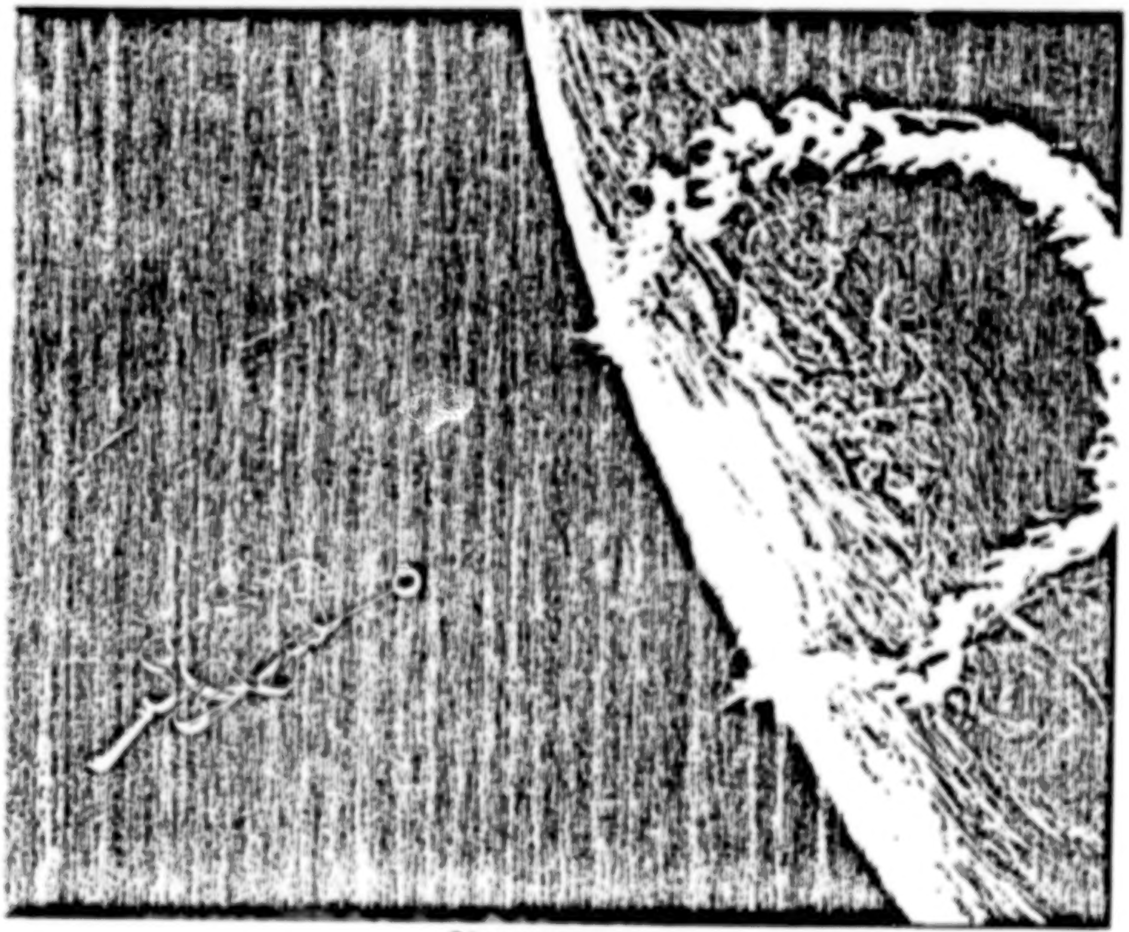

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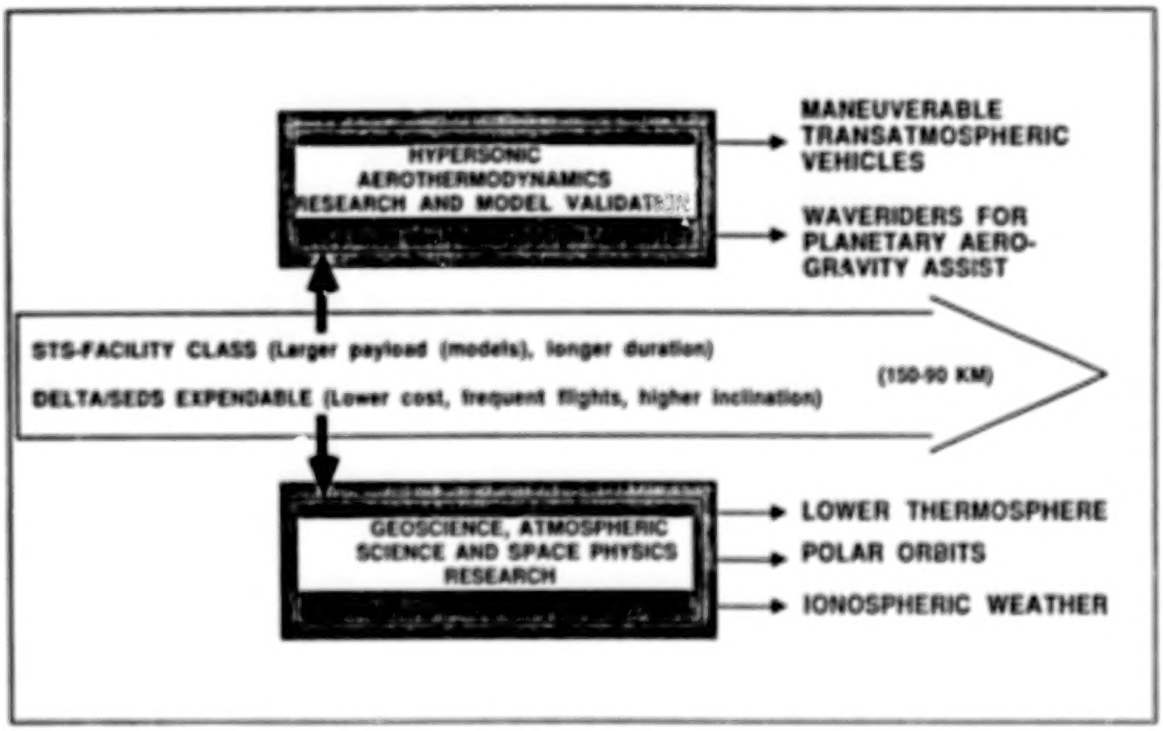
# USES OF TETHERED ATMOSPHERIC SYSTEMS



HYPERSONIC AEROTHERMODYNAMICS RESEARCH AND MODEL VALIDATION

→ MANEUVERABLE TRANSATMOSPHERIC VEHICLES

→ WAVERIDERS FOR PLANETARY AERO-GRAVITY ASSIST

STS-FACILITY CLASS (Larger payload (models), longer duration)

DELTA/SEDS EXPENDABLE (Lower cost, frequent flights, higher inclination)

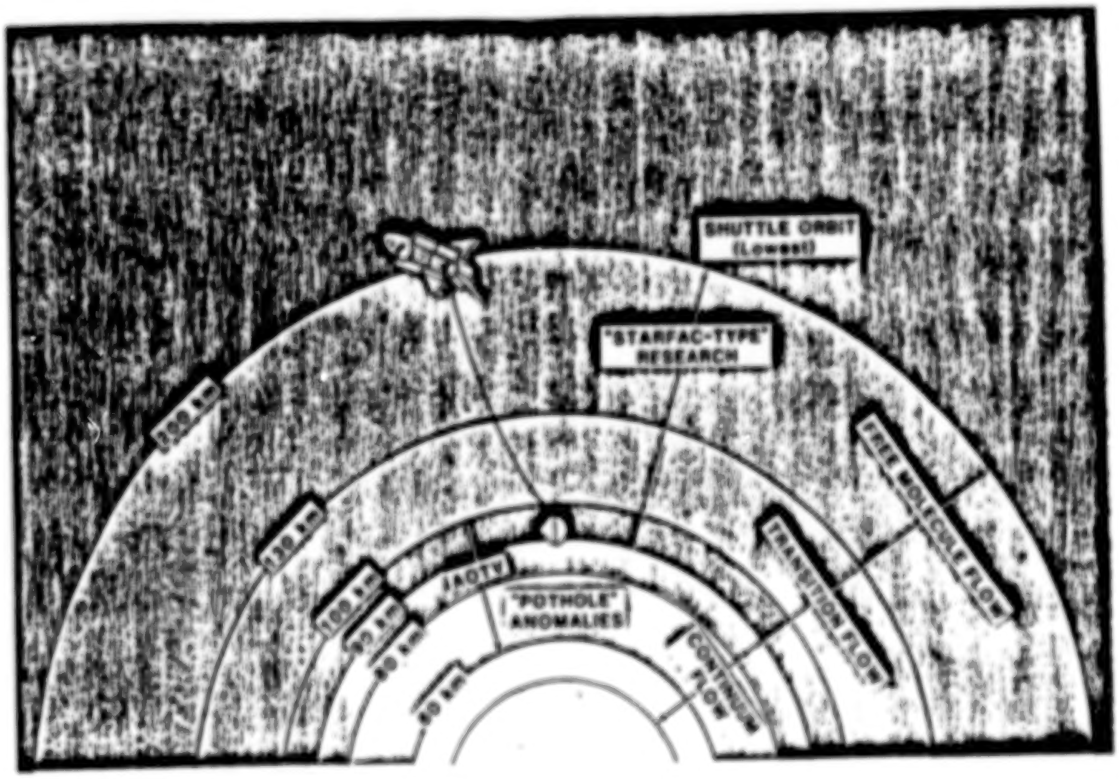
(150-90 KM)

GEOSCIENCE, ATMOSPHERIC SCIENCE AND SPACE PHYSICS RESEARCH

→ LOWER THERMOSPHERE

→ POLAR ORBITS

→ IONOSPHERIC WEATHER


SHUTTLE ORBIT (Lowest)
"STARFAC-TYPE" RESEARCH
200 km
130 km
100 km
90 km
80 km
60 km
AOTV
"POTHOLE" ANOMALIES
CONTINUUM FLOW
TRANSITION FLOW
FREE MOLECULE FLOW

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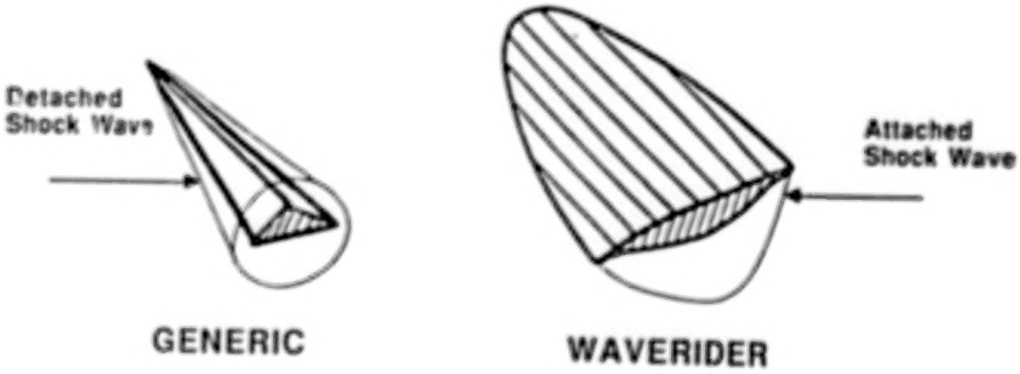
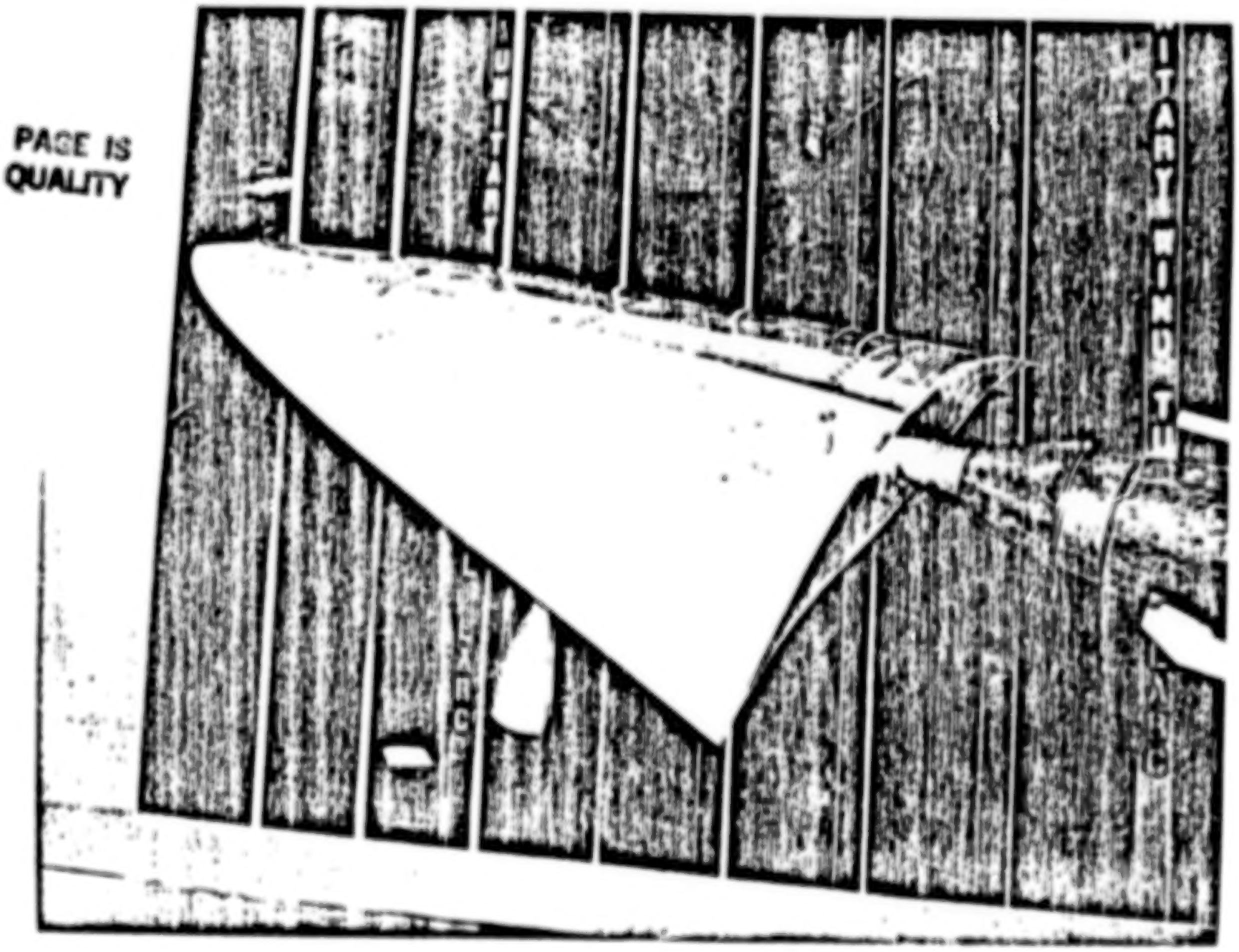

Detached
Shock Wave
Attached
Shock Wave
GENERIC
WAVERIDER

**HYPERSONIC CONFIGURATIONS**

NASA



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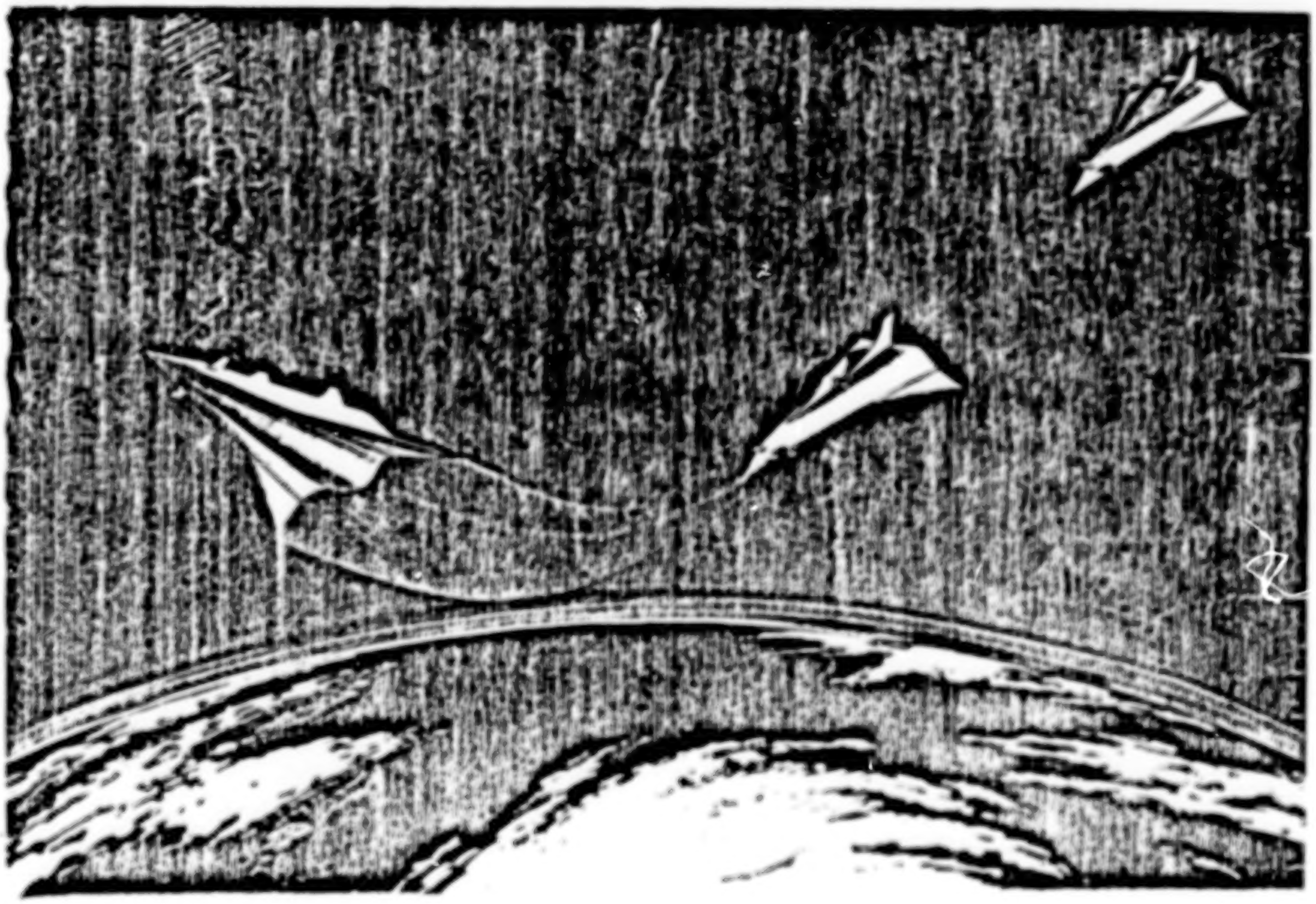
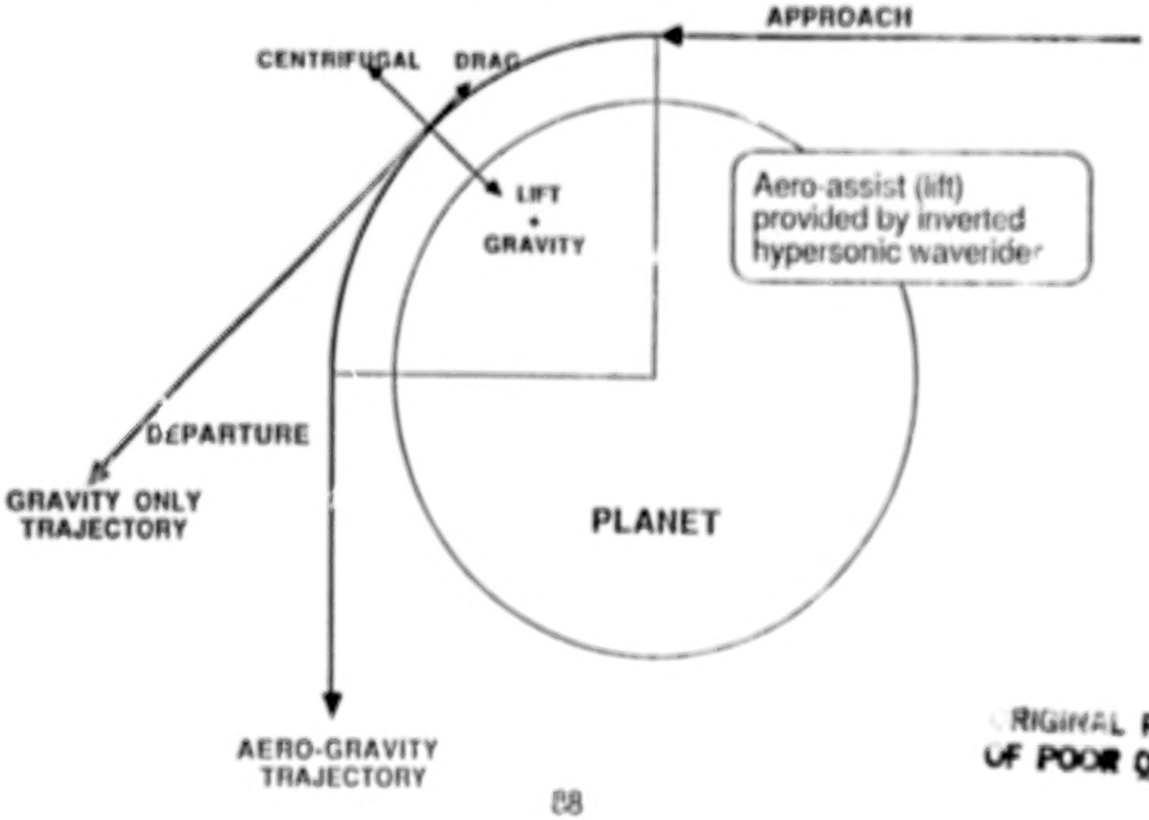


# ATMOSPHERIC TRAJECTORY-ASSIST FOR PLANETARY SPACECRAFT


APPROACH
CENTRIFUGAL
DRAG
LIFT
+
GRAVITY
Aero-assist (lift) provided by inverted hypersonic waverider
DEPARTURE
GRAVITY ONLY TRAJECTORY
PLANET
AERO-GRAVITY TRAJECTORY

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# SOLAR SYSTEM MISSION TIMES WITH HYPERSONIC WAVERIDERS

| MISSIONS | JUPITER GRAVITY ASSIST | TERRESTRIAL PLANET AERO-GRAVITY ASSIST |
|---|---|---|
| SOLAR PROBE | DVEJS - 5 YRS | EVS - 4 MOS<br>EMS - 5 MOS<br>EVES - 6 MOS<br>EVMS - 9 MOS |
| PLUTO FLYBY | EJP - 12 YR | EMP - 10 YR<br>EVMP - 5 YR |

HYPERSONIC WAVERIDER WITH L/D ≈ ~10 AT M ≈ 20 - 30
ATMOSPHERIC FLIGHT TIMES ~ 200 - 500 SECONDS

SPACE-TETHERED WAVERIDER MODELS MAY PROVIDE THE ONLY WAY TO DETERMINE FLIGHT CONDITIONS & VALIDATE CONFIGURATIONS AT THE EXPECTED MACH NUMBERS

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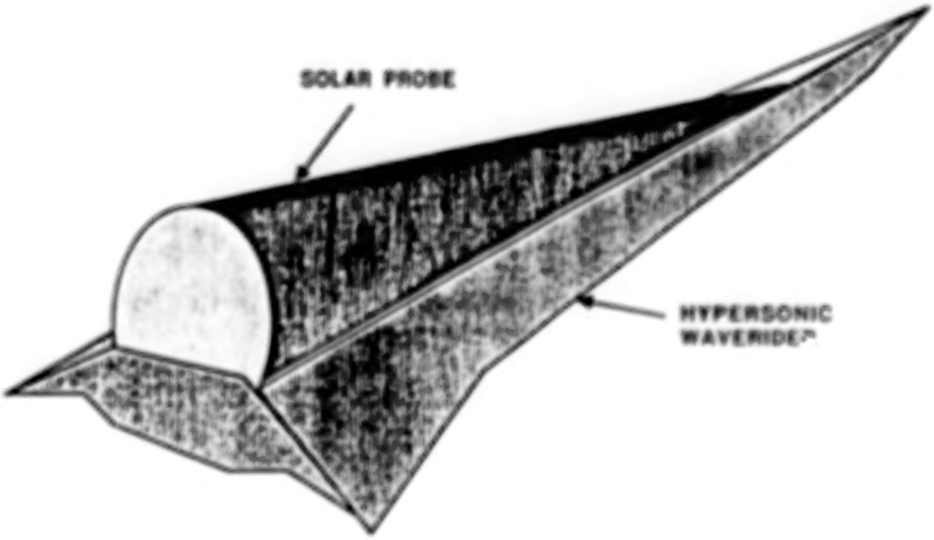
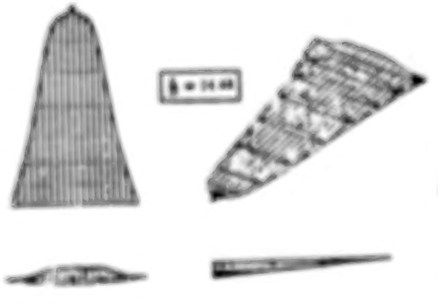
SOLAR
PROBE

# WAVERIDER - SOLAR PROBE SCHEMATIC


SOLAR PROBE
HYPERSONIC WAVERIDER


(a) h=50 km.

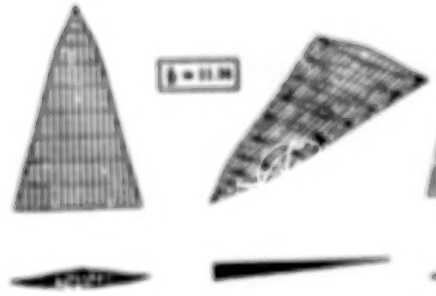
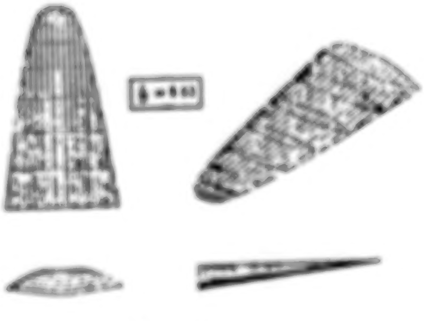

(b) h=70 km.

Figure 9. Waveriders designed for the Venus atmosphere, $M_\infty = 20$, $\theta_{min} = 5°$.


(c) h=50 km.

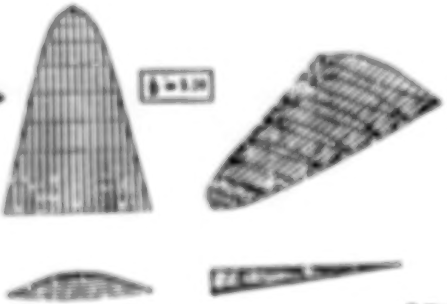

(b) h=30 km.

Figure 10. Waveriders designed for the Mars atmosphere, $M_\infty = 15$, $\theta_{min} = 7.5°$.

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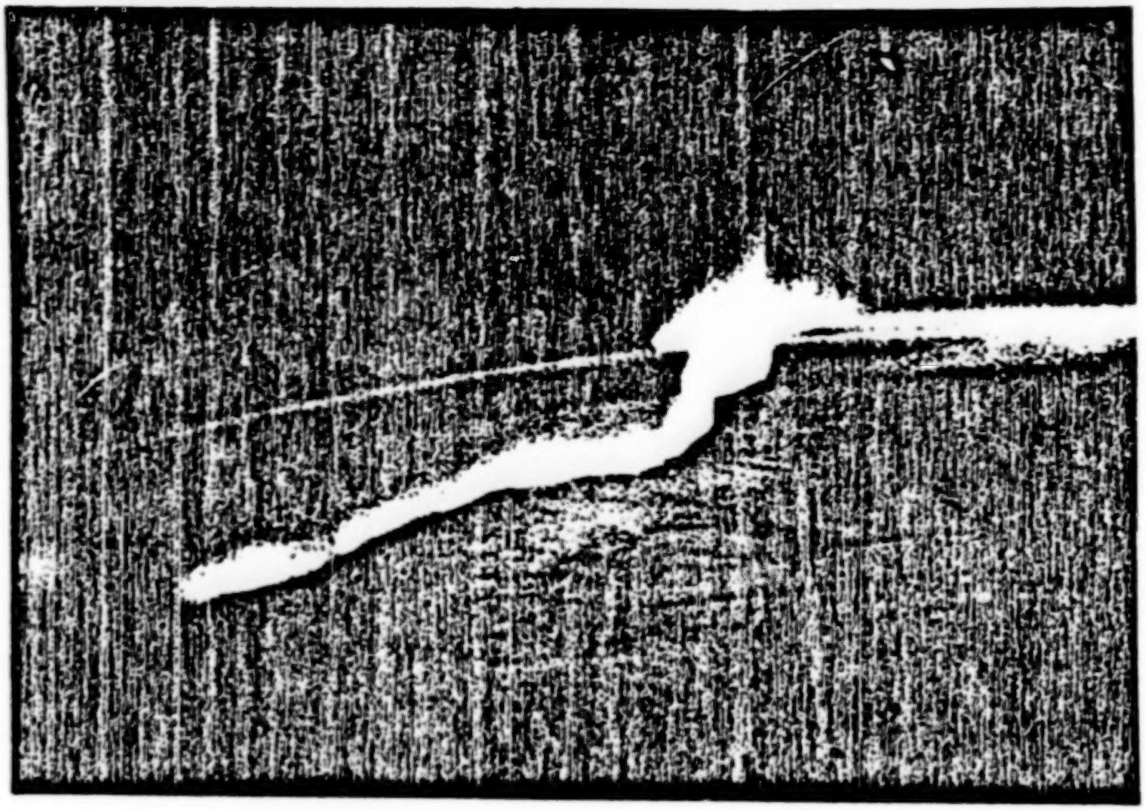
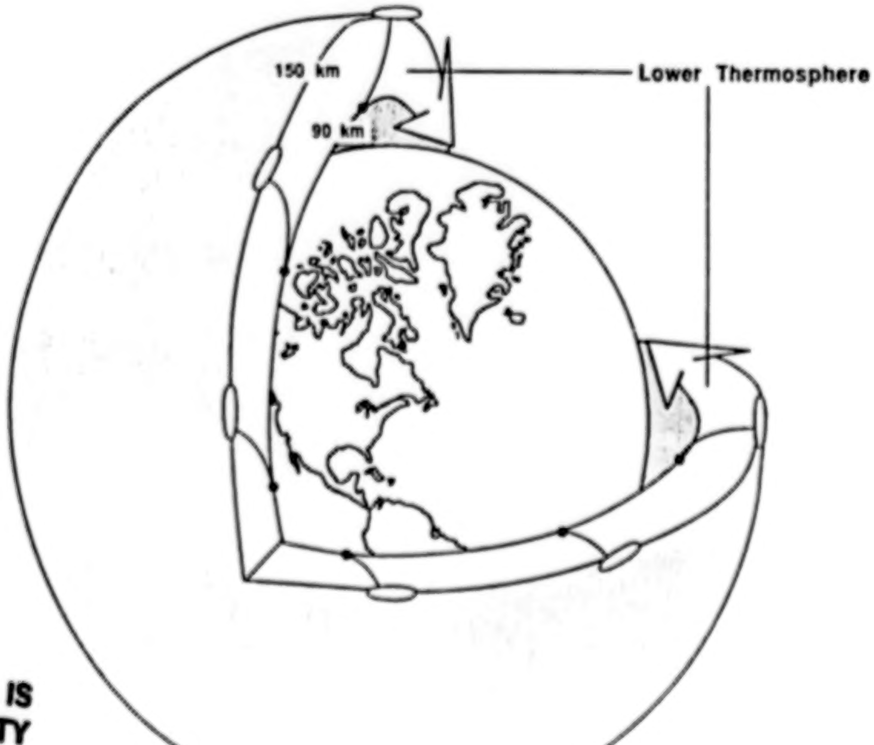


## GLOBAL REGION ACCESSIBLE TO TETHERED ATMOSPHERIC SYSTEMS

OAST


150 km
90 km
Lower Thermosphere

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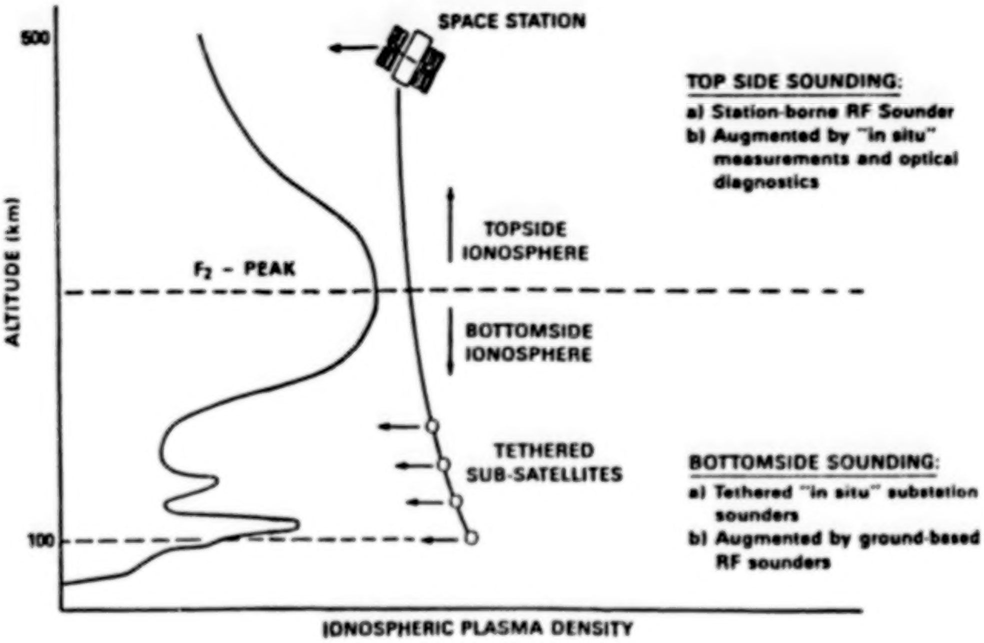
89-2311

# IONOSPHERIC WEATHER SPACE STATION (FUNCTIONAL CONCEPT)


500
SPACE STATION
TOP SIDE SOUNDING:
a) Station-borne RF Sounder
b) Augmented by "in situ" measurements and optical diagnostics
ALTITUDE (km)
TOPSIDE IONOSPHERE
$F_2$ - PEAK
BOTTOMSIDE IONOSPHERE
TETHERED SUB-SATELLITES
BOTTOMSIDE SOUNDING:
a) Tethered "in situ" substation sounders
b) Augmented by ground-based RF sounders
100
IONOSPHERIC PLASMA DENSITY

Taken from "Technical Issues in the Conduct of Large Space Platform Experiments in Plasma Physics and Geoplasma Sciences", by Ed Szuszczewicz, an invited paper published in "Space Technology Plasma Issues in 2001", NASA JPL Publication 86-49, October 1, 1986.

# III. Plenary Panel Discussion

# Space Travel For the Next Millennium
## *PANEL DISCUSSION*

participants:
Geoffrey A. Landis (moderator)
Robert Forward
Marvin Minski
Theodore Taylor
Joel Sercel
Paul MacCready

Landis: The purpose of the panel discussion is to allow all of the panelists to comment on the various questions, and to argue with each other, I mean to discuss with each other, some of the various topics and get interaction among the various panelists. I have far more questions than we are going to have time to discuss during the panel, but, we will try to do as many as we can. Again, I'd like to mention that we would like all the panelists to discuss. If you have something to say about it we'd like you to just pop right in.

The top question on my list is directed to Bob Forward, but I will open it up to everybody. The question is: If chemical rockets are too expensive and nuclear propulsion is politically unfeasible then what is the most likely propulsion choice for manned travel in the solar system?

Forward: Well, it's solar electric and maybe solar thermal. But that is going to start limiting us as we get out further in the solar system and get further away from the sun. After that, why, there's sails and tethers. I would like to have both of those subjects not only explored in terms of building up our pile of paper stack of studies but actually get some real operational experience in space with some demonstration experiments. I think that's what we need more of is technology demonstration tests rather than complete scientific experiments. Unfortunately the present budgets of NASA aren't designed that way so we have a real problem that we need to turn around.

Sercel: I guess I would first question the assumption.

Forward: Of course, that's true. First question the assumption.

Sercel: The first thing we need to do is study that assumption; and the problem with nuclear propulsion is two-fold: First we have to identify the technical issues required to absolutely assure that it is safe and then we have to take the story to the public and make a convincing argument that it's safe. Now if we lose one or both of those battles then we need to consider other options. Solar energy is probably the best other option. As Bob pointed out, studies in solar electric propulsion show that performance is comparable to solar sails. As you know; there is a rough trade between solar electric and solar sails. And, for operations for launch vehicles in near Earth space, maybe beamed energy.

Landis: Ted, you may be unique in being an expert in both nuclear and solar propulsion. What are your thoughts?

Taylor: My thought is this: that I think it's likely that we can do everything we want to do out to somewhere around the middle of the asteroid belt with solar electric; possibly solar sails—and won't need nuclear energy for high performance. Beyond the asteroid belt there is some possibility of beaming by laser from close to the sun, but the receiving end of this (if it's Saturn or even Jupiter) gets awfully big; just because of the diffraction limit.

So what sounds good to me is to relegate nuclear propulsion to exploring, or doing whatever else we do, at the outer planets and make all of that, without exception, international. It is a little bit like the Baruch Plan for development of nuclear energy in the first place--that it should all be done under international auspices. But I would go a little further and say there is no reason even to

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have any nuclear power systems that have been started up before they get way out there. And then as long as it's not threatening, because it is being done internationally, we could make full use of it.

Minsky: There is a way of increasing the priority for nuclear, I think. If there is some urgency, like a war, people operate with different criteria. If there is genuinely going to be a threatening asteroid out there that was going to do some huge devastation, all the countries would band together and rational choices would be made with less politics to it. However to know that in six years something is going to come and zap you (I think that is beyond our forecast capability, but maybe we can convince somebody that even if it turns out to be false) then we might be able to accomplish your mission.

Landis: The next question is: "What are the pro's and con's of international cooperation versus international competition?"

Minsky: I think it's getting harder because nobody keeps their promises. It's not much help for the United States to offer people launching space when we don't have any. I think Ted is right, if we can get efficient fabrication on the moon I think that should be the first priority, although maybe asteroids are easier. Then you only have to send up unstarted nuclear. You're sending up materials that haven't gone critical yet and they are pretty harmless (although it could be hard to convince anyone of it). But, you know, if we wait about ten years it looks like everybody will starting to be nuclear again despite themselves. The threshold will have gone down just because of the fuel shortage.

Sercel: We looked at that with the Odyssey concept. Technically, it appears that one could use nuclear electric propulsion to do pilot exploration of just about any interesting target in the solar system this side of Saturn with round trip times of something like 5 years. So we took that analysis and said "What could we do with that with a space project that would challenge this nation well into the 21st century?" And one of the aspects that came up early in our look at it was the international aspects. So, I went back and reviewed, did some historical reviews, and if you look through history societies that have a high degree of contact with other societies have always been the ones that developed at the fastest rate. Presently, the world is sort of smaller than it has been in the past so there is always that kind of contact. But if you could design an international program that had the United States as sort of the center of it with each of the partners contributing parts, then that would put us at the center of this network. That would potentially be very enhancing for technology development.

Landis: On the topic of speculative ideas, what methods do you use to get new ideas seriously considered? A lot of people here have some wild ideas that they'd like to get some work done on. What do you do? Where do you go?

Forward: Well, the first thing to do is do a very serious study and get it published. The very fact that you've gone through the peer review helps a lot. But then you have a real long up-hill battle of going around and getting people interested in it. And keeping them interested in it and keep getting invited back and keep talking about the subject. I mean, after all, why do you think I wear these fancy vests? I want you to remember me, OK? And hopefully, remember the ideas. And if you keep talking to enough people, and enough *young* people, why, pretty soon the idea will get accepted.

But it takes years. Decades.

MacCready: I don't have any great ideas of how to do it; every case is a different case. You have to be lucky, and dogged and the times have to be right. The big problem is that so much of the resources get linked into long term giant programs. There aren't a lot of resources for getting started. Especially missing are discretionary funds that lab directors can allot to worthy causes. And when you've got to get into next fiscal year's budget, or the one after that, even to get started on the project, then things quiet down. So somehow diverting a significant amount, 5% or so of budget to high payoff and maybe low probability projects is very important, though very hard to do politically.

Forward: There is a program called the Small Business Innovative Research program which gets a certain percentage of every DOD and NASA budget. This money only goes to small businesses. A lot of people that I know who are interested in advanced propulsion have gotten some interesting programs started. The first year the first contract is $50,000 or less, but the second or third contract can be of the order of a quarter a million a year, and you can do a lot with a quarter million a year–if you can get past that first phase. So that is a program and those that are interested in furthering ideas of advanced propulsion, or something else, can use that program and people with ideas can form a separate company even though they still work for somebody else. In fact in one of the programs in antimatter research, this guy got together with a university professor and formed a small company and won a bid on a certain part of antimatter research. So that is a way to do it.

Landis: I might mention, however, that the Small Business Innovative Research program is very competitive and that if you are going to try to get a SBIR grant you had better put together a very solid proposal that that isn't something with big holes in it. You've better have figured out just what you want to do, and not propose ideas that you haven't thought out yet.

Minsky: When I was an Assistant Professor one day a person from Exxon showed up and gave me this check for $10,000 which he said was for discretionary research. It was pretty thrilling and lasted four years and ended up being worth about $300,000 because some student would want to do something and I'd say "well okay what do you need?" and he'd say "I need this gadget, or that," so we'd get it, and then it almost always turned out that something useful came, so I could charge it to the ARPA [Advanced Research Projects Agency] contract or something. But the leverage of small amounts, just trifling amounts, of money—so you can get the materials, so the student can prove that it's feasible, and then the visitor comes and says "OK we want that"; ONR comes around and says "Oh, we'd love you to include that under the real contract". Real contracts don't give you five minutes or five dollars to spend on "crazy things," so somehow you have got to get that $10k from some philanthropist, just a bit of discretionary money outside of the contract so that the auditor doesn't see. If you wanted you could take it home, I suppose, but that's not the point.

MacCready: That SBIR money really is important. Yesterday I met the guy who started the program and he's absolutely delighted with it and feels proud that he is a public servant that got that going and said that they have doled out more than two billion dollars now in how many years it has been going on. My company has done about a dozen of them, so far. I think the percentage of winners is not bad--it's one out of four or one out of six, not one out of thirty. And they really are after innovation and you do it as a small company and you are able to keep the proprietary rights for whatever you do which is a strong inducement.

For certain topics a prize is a very good way to stimulate development. Somehow if you put up a prize of a certain amount of money, it harnesses work a hundred times that amount of money--or a thousand times. It provides a focus and I don't know if people are after the fame or the money or what, but suddenly a lot of people start working on it. It turns out to be kind of hard to find just those challenges that you can have a prize with a good simple set of rules and the right criteria. That's not easy, but when you can then a prize is pretty good. You have to figure out somebody to put up the prize money, of course, but it's got some nice flavor and you only pay for winners.

Minsky: Recently a Japanese person came to me very discreetly and said "if they offered you a prize would you accept it?" which I thought was pretty funny. They don't want to make any mistakes. So he told me about the Japan Prize and I said "Oh, I suppose it is for my new Society of Mind theory." He sort of blushed and said "No, it's mostly for that paper you wrote in 1961!" So, I am not sure that prizes are that much of an incentive.

MacCready: People do some things for other reasons also.

Landis: The solar sail prize that was recently mentioned--for a race that may never even get started--seems to have stimulated quite a bit of interest.

Landis: Here is a question for Marvin Minsky. Joel Sercel talked a lot earlier about self-replicating robots. What are your comments on what Joel said? Is this something that is likely to happen?

Minsky: There are some very critical things, it seems to me. If you make the kind of robot I was talking about then you would make it out of exogenous materials. You know, maybe you can make all of the parts out of pieces of fused glass or metal--whatever you find on your asteroid or the moon.

The problem in self replication is usually that after a while you say "oh, I missed something." This robot is going to need a vidicon or a computer or a memory. I think that for the foreseeable future you can make self replicating systems that are pretty small and compact if you send the seeds--the vitamins, as Sercel calls them. A capable computer only weights a hundred milligrams. What does a 68040 weigh? It's a few milligrams, it needs some electricity--you're going to need some magic way of getting power. I should think a good robot could make thermal bi-metal junctions, if it knew enough. But I don't see any easy way to make a transistor factory right now. It seems to me you could make most of the mechanical stuff in a robot. Maybe the vitamins include little motors. I don't know how hard it is to make motors, but it seems to me that if we are talking about self replication, with a certain payload that we could do tremendous things right now. You have to ship the inside of the joint and the little computer, and all that is only a kilogram. And then you make all the gross stuff. These robots, clumsy as they are, ought to be able to fabricate the other clumsy parts of robots, and that's a big leverage. Maybe if you send five or six of these to the moon and a hundred kilograms of chips and sensors that's enough to make a big lunar factory--except for a few little critical things.

Landis: Joel, do you agree with that?

Sercel: Sure. When we were talking before I asked Marvin how long he thought it would take to make the first self replicating machine. My guess was fifty years; Minski said ten years, so I guess I am too conservative.

Minsky: But I think you are including making the vitamins too.

Sercel: No, I'm not, I'm assuming importing the vitamins.

Minsky: Well, I don't want to stick to ten years because when you are doing something ten years seems like a short time. I know I had this experience a couple of times in research. One of my really great students, Pat Winston, wrote a wonderful learning machine program--learning structural descriptions by example. It's sort of a classic in artificial intelligence. And this was a little program that would learn to build little structures; a little house, or arch, or a tower out of children's building blocks and we all thought that was a great thing and we just looked forward to the next graduate student who would take it another step and it was ten years. I don't know why, but if you have a good idea you can't order--or at least I don't order--a student to work on it. That never worked anyway. So, three or four years later another student understands the thing and starts to work on it.

Our PhD's usually took about six or seven years because they liked it so much hanging around the lab. So, you could think of ten years in leisurely basic research as just the average time between each idea and the obvious next step. So, when I said ten years I don't think I really meant it. And, fifty years might sound like a long time but it is just five steps of that sort. How many years between Newton and Feynman? Just about 300? And no one could say physics was crawling along in that time. It's a short time for major things, so who cares?

Taylor: One thing I feel compelled to say about self reproducing automata. There are pretty persuasive arguments that say that the gestation period, once we learn how to make these things, is likely to be nine months, or a year. The litter size could easily be ten. Question: what happens, when that population explosion takes off, a hundred years later? It seems to me that there are likely to be other people out there somewhere who at some stage come across this and a hundred years later, out goes a paving and reshuffling and redoing enterprise that goes out more or less spherically at about half the speed of light, or maybe closer to the speed of light, and redoes everything in its way. That must have happened. Question: are we in the middle of it right here? Right now? Were we produced by self reproducing automata?

Landis: That's partly an argument due to Frank Tipler. The point is that if there had ever been civilizations anywhere in the galaxy that sent out self-reproducing machines, they would have been here billions of years ago

Sercel: I think it's clear that you're not going to program a self-replicating machine to reproduce in an uncontrolled way if you're intelligent enough to make one in the first place. I would guess that if some civilization had made self replicating machines, and a self replicating machine came into our solar system, it wouldn't necessarily start reproducing itself and take over the solar system. It might be out there in the asteroid belt watching right now.

Landis: Since we've sort of started on the subject of interstellar travel, the next question is: "Is the Orion, [which is the atomic bomb powered space ship which you see a model of over on the right], is the Orion concept still our best bet for an interstellar mission in the next fifty years, say, for a one-way, un-manned, fast fly by of Alpha Centauri?" And then as an addendum to the question: "Is it feasible to assume that a two hundred year trip time for such a mission could be realized without catastrophic failure of the space craft sub-systems?"

Taylor: Well, the interstellar version of Orion came out of Freeman Dyson. We thought we were thinking pretty big with a space ship that would deliver a thousand ton payload to Ganymede. We were pikers compared to what he did. He had a space ship which was several kilometers across; the bottom of it was several kilometers across. And what made it go was around a million ten-megaton H-Bombs. What this did was to take something perhaps the weight of loop Chicago off to Alpha Centauri. It was a very big concept. I think he did that for the sake of completeness, to say "Well what is the limit of this thing?" I mean, a million H-bombs are not completely out of reason. Johnny Wheeler had been pushing for that for years, that we actually build a million nuclear weapons in case we went to war in Europe.

Interstellar travel seems to require a violation of some of the basic principles that we hold dear. To make something to connect back to Earth within lifetimes... the energy requirements are huge. When you look at those numbers and talk about sending something that weighs, say, a hundred tons (which is awfully small for a voyage that long) up to, say, half of light speed, the energy requirements are on the scale of all the energy that has been consumed by human activity from the beginning of time. It's a whole different scale. Although we sometimes described it as an interstellar propulsion system, Orion never really was.

Sercel: It's worthwhile to point out that about every thirty years or so we double our energy consumption rate. In the process of that thirty years we expend as much energy as we have used in the entire previous history of man. So, if you do the back of the envelope calculation and assume that we continue to increase our energy consumption rate, it's only a matter of a few hundred years before we get to the point where a large interstellar mission is just a small fraction of global energy use. So, maybe the easiest thing to do is just wait.

Forward: Orion is the only interstellar vehicle that we could have built ten years ago. If, for instance we knew the sun was going to go nova or the Earth was going to die or something like that and we had send some of our seed off that's something we could have theoretically built, and built it a long time ago.

My effort over the last couple of decades has been to try to find some other way of going to the stars, other than using rockets. Now, many physicists have taken out the back of their envelope and proved that you can never get to the stars using rockets in ten years and they've done it and showed that you'd have to use up all the deuterium in the world's oceans as energy source and reaction mass, in order to send one interstellar vehicle to some star a hundred light-years away and bring it back. And you can prove that, and that's because you make the wrong assumption!

One wrong assumption is that you are going to accelerate at 1 g. You don't need to go at 1 g. You just need to go at 1 g for a year and then you are up to seven tenths of the speed of light and coast the rest of the way.

Another thing is, you don't want to use rockets for interstellar flight in the first place, and so the rest of my effort in this field has been to try and find some method of moving through space other than using rockets. One of them is the Bussard ram-jet, which unfortunately Dana Andrews and other people have tried to make work and found we can't figure out how to make the hydrogen scoop yet.

Another is to use beamed power. I have written two papers on this. One of them is on a space vehicle, starwisp, that only weighs twenty grams--less than an ounce--and returns color TV pictures from Alpha Centauri. Those kind of things don't violate any physics, and they don't use up all of the world's supply of energy. In fact, all it needs is a solar power satellite to get it there and get the images back. So, you can go to the stars without violating physics and not using up the world's supply of energy, but it's not going to be easy, and it's not going to be fast. These things only get up to twenty per cent of the speed of light, so the round trip mission takes 25 years to Alpha Centauri. So you can talk about going to stars, and it's fun. But we still need better ideas and it is what I hope to inspire in some of you younger guys here.

MacCready: There's another way of looking at going to the stars. It's not the approach that you're interested in here, perhaps, but if you put a small amount of money, (small compared to the amounts for the programs that have just been talked about) into investigation with radio telescopes, IR and optical, that have diameters that are sort of the diameters of the Earth, or by locating things on the moon and planets and so on, that you can get a huge amount of information about what's going on there. You are visiting them, but you are not visiting them by going there and bringing something back. You are visiting them by really looking at every bit of radiation that comes out.

Forward: My last novel was deliberately written to include an alien life form that would never have radio, and yet was very important to find because he had much more intelligence and had developed mathematics much further than we had. I deliberately did that because there are people like Barney Oliver and Sagan that say that the only way to do this exploration is by listening by radio and anybody that talks about flying to the stars is right off the back of the cracker box.

Taylor: I have to say that if the natives out there are friendly and have an urge to get up close to us, they'll come and get us.

Forward: But not if they don't have technology.

Landis: And if they are unfriendly and have an urge to get up close?

Taylor: I think if they are interested, that's a problem.

Forward: Do you think the whales will develop technology?

Taylor: Whoever is out there, if there is anybody out there, the chances of there being only a a hundred years from the invention of radio, and such are infinitesimal. They are either amoebic, or monkeys, or way ahead of us; not right where we are.

Forward: If you're underwater, you may not develop technology.

Landis: I've always wondered why the SETI people keep focusing on radio anyhow because obviously any intelligent life form would use the shortest wave length possible to communicate over interstellar distances. So, perhaps we won't find out anything until we get the gamma ray telescope up.

Forward: Or neutrinos?

Sercel: Well, since we're in the spirit of speculation here, we're talking about maybe you can travel by information is what Paul was suggesting as opposed to physically travelling, well if they've already built their self replicating machines and they are sitting on the asteroids waiting for us then all we have to do is get in contact with one of them and they can send the information required to make a human being back in their home world and we can then have human beings on their home world.

Landis: So they are just waiting for us to develop the receivers that can down load their life forms.

That leads to the next question, which is: "What kind of impact would the space Hubble telescope have on space exploration should it prove that nearby stars do have planetary systems?"

Forward: Is it designed to do that?

Landis: No, actually it is not particularly designed to do that, although some people have been proposing to try. There are other space telescopes coming up that might. One might have been the one that the Europeans launched, Hipparcos, but I guess it's having problems since the apogee kick motor failed to put it into the right orbit. If they don't put another one up, thre should be an Astrometric telescope up in a few years. This will measure the positions of nearby stars to a sufficient accuracy that they should be able to detect Earth-like worlds within I think a hundred light years and, if I am not wrong, Jupiter-size worlds within a thousand light years.

Audience: Haven't they recovered Hipparcos enough to get data despite the failure?

Landis: I've heard that they are getting a lot of data out of it, not nearly as much as they hoped but that they were getting good data and pretty soon we should learn something from this.

Sercel: It's worth pointing out that there is a good deal of evidence already of planetary systems around other stars. And that hasn't resulted in a revolution in our space program. For example, images of the star Beta Pictoris suggest some kind of planetary system, as well as some of the infrared data that came back from IRAS.

Landis: The IR signature really comes from pretty small particles. It's really dust that they're talking about.

Sercel: But it is suggestive of the first stages of accretion of planetary systems.

Forward: I don't know. Once we actually design a telescope that has the right kind of occulting disk to block out the major star and is well designed so the stuff leaking by doesn't louse it up and really finds a green planet..I think that once we have a picture of a green planet a lot of people will be very intrigued, I think, and interested in going there and that would be fun.

Landis: One of Bob Forward's papers on interstellar travel suggested it should be possible within the next many decades to focus laser light distances of light years with a lens a thousand kilometers or larger in diameter, an O'Meara 'para-lens.' O'Meara's intent in studying the possiblity of making such large lenses was to use them for telescopes. With a lens that size, you could not only detect Earth-like planets out to hundreds of light-years, but you could *map* them with hundred-kilometer resolution. So that if the planets are out there and we have the technology to get there via these laser-propelled ships, we'll know where we're going long before we do.

MacCready: Before we get off this, looking at the practical side, such projects are going to have to be government funded. The government is run by people interested in what happens during their term, not some far distant term. And although a few far-sighted things sneak through I think it is going to be very hard to do something where the results are going to come in thirty years later, or two hundred years later, with our political system. Or our economic system.

Forward: In my very first paper in interstellar flight, I pointed out that even if you could travel at the speed of light it would take you 4.3 years to get to Alpha Centauri, and either 4.3 years to come back, or for the information to come back if you decided to stay there. And that is 8.6 years, a half year more than the term of a president. So no interstellar mission will ever fly.

MacCready: Whereas if you have this super telescope you get the information right now.

Minsky: Maybe if we could ban re-election to all offices then this problem would change. That is probably the first priority.

Landis: Kennedy asked for a moon mission within a decade and that happened.

Forward: Yes-- but it's Nixon's signature.

Landis: What do you think the possibility is for non-government funded space exploration: including everything, SETI and all of that stuff? Do you think it's ever going to be possible, or are government agencies like NASA the only organizations that are large enough to fund space exploration?

Minsky: The D.D. Harriman problem. We need more immensely wealthy people.

Sercel: I would point out Orbital Sciences, Geostar, several other start-ups that are doing quite well in the space business. I think it looks better now than it ever has for non-government funding.

Forward: I think it is a real problem. Interstellar flight is non-profit. The real answer is multi-billionaires, yes.

Audience: There are some small scale amateur kinds of explorations are going on now, and I point out that the ham radio community has a number of satellites up that they are using for serious communications. They have been doing it essentially by being hitch-hikers on much larger satellites that have some nooks and crannies and will accept experiments on a non-interference basis.

Landis: Here's a question for Marvin Minsky: "What do you think the likelihood is of AI machine civilizations elsewhere in the galaxy? What do you think is the likelihood that we might evolve into a machine civilization ourselves?"

Minsky: Well, that's sort of Gregory Benford's field these days, and David Brin. It seems to me fairly likely that in a thousand years or so, or less, we'll turn into one. I just can't imagine a people as smart as us tolerating disease and senescence and all that sort of thing once the option becomes available. There's Hans Morovic's script for that in *Mind Children*. Some people will say, "no, I don't want to be machines" and they will have their choice and die out. There is this singularity in evolution and when we understand how to make ourselves into better hardware then some people will do it and some won't and that will be that.

Sercel: I guess I agree. The one thing I wonder about is whether you go green or go gray. The metallic approach or genetic engineer.

Minsky: I guess green is cheaper.

Forward: Yeah, I agree with it too. I usually say we already do turn our world over to little bitsy robots. And we do it because we trained them, brought them up to be human beings and believe in our culture and we trust them, finally, enough to go and retire and turn the world over to them. I don't see any difference between that and the act of training some kind of silicon little being and doing the same thing.

People argue, "but they're not made of meat." Of course they don't exactly say that, but that's really what we are trying to do: run a world with computers made of meat. And I think there are better ways of building intelligent beings.

Landis: I might add that makes the interstellar travel problem much easier because the time frames don't really matter. If it takes a thousand years to get to Alpha Centauri at a small fraction of the speed of light, that's OK, because you turn yourself off and turn yourself back on when you get there.

Forward: Yes, but I think one of the whole points about interstellar travel--that if it takes you more than fifty years to get there and you haven't completely eliminated all the other methods of propulsion, then by the time you're half way there, somebody is going to pass you up. Yyou don't really want to build a space craft unless it can get there in less than fifty years. At least not until we have run out of options on propulsion technology, and we are a long way from that.

Landis: That's OK, it's just different copy of you that's passing you up. So you just download your new copy onto the old one as you go by.

Forward: OK.

Sercel: With regard to machine civilization, one might observe that the transformation between biological civilization and machine civilization is not a distinct one, but it's a gradual process. And it's a process that's already started. We use automobiles for our legs, and we use tractors for our arms, and, in fact, considering the impact we are having on the biosphere (which is purely biological) it may be that machine civilizations and biological are not compatible and it may be the most natural place for a machine civilization to live is out in space where it is not interfering or destroying a pristine, delicately balanced biological environment.

Minsky: Where there is no polyethylene eating bacteria. That's in Larry Niven.

Landis: We're getting close to the end of time. Let me ask this one last question. "Should the Solar Power Satellite be used as a focal point for long range international space efforts? And, if so, what should we be doing right now?"

Sercel: I think the space Solar Power Satellite is a very interesting concept. It has some problems and it has some strengths. Where we stand right now is that we don't know enough about the technology options that can be investigated to really know whether it can be made cost effective and safe. So, if those studies conclude that it can be, then it would be a very interesting thing to pursue, but I don't think we have enough information to make that decision right now.

Forward: I'd rather see that we focus on some other goal. I mean, that particular choice isn't so obviously effective. We just really don't know. It has some very good points, as Joel said, but it's not so obvious as "Let's go to the moon" or "let's go to Mars". Those are things that aren't really trying to make a profit. You're just doing it for the heck of it.

Taylor: Before using solar energy collected in orbit will look in any sense economical for producing electric power, it will have first happened on the Earth's surface. In fact, if you want to see a candidate for a near-term revolution in energy production on the surface, I think a clear front runner is hydrogen produced by low-cost photovoltaic cells and then used as an all purpose fuel. There's a lot of attraction for a hydrogen economy. We can do it on Earth. In fact, we are right on the verge of being able to do it this decade at costs that will compete with natural gas. So, I guess I don't see what burning need the Space Power Satellite meets that we can't do cheaper, better, quicker on Earth.

Minsky: I have one concern. I wonder which projects we ought to do as soon as possible because rising population and the fusion of interests in world governments will come to the point where nobody dares do anything. There is a fear that in another twenty years a curtain will come down on all forms of exploration and everybody will be too careful to launch anything.

Landis: I regret we're out of time and so I have to close the discussion. I would like to thank you for participating, and I hope you enjoy the rest of the symposium.

# IV. Poster Session Papers

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N91-22144

# ULTRA HIGH TEMPERATURE PARTICLE BED REACTOR DESIGN

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Upton, NY 11973

## ABSTRACT

This study is a computer analysis of a conceptual nuclear reactor.

The purpose of this work is to design a direct nuclear propulsion engine which could be used for a mission to Mars.

The main features of this reactor design are high values for $I_{sp}$ and, secondly, very efficient cooling. This particle bed reactor consists of 37 cylindrical fuel elements embedded in a cylinder of beryllium which acts as a moderator and reflector.

The fuel consists of a packed bed of spherical fissionable fuel particles. Gaseous $H_2$ passes over the fuel bed, removes the heat and is exhausted out of the rocket.

The design was found to be neutronically critical and to have tolerable heating rates. Therefore, this Particle Bed Reactor Design is suitable as a propulsion unit for this mission.

## I. PURPOSE

It is desired to design a direct nuclear propulsion engine which could be used for a mission to Mars.

## II. MAIN FEATURES

Two characteristics of the design are high values of $I_{sp}$ and, secondly, efficient cooling. These traits result from the use of a particle bed reactor.

### A. The Specific Impulse

$I_{sp} \propto V_{exh}$, where $V_{exh}$ is the rocket exhaust velocity. To increase the $I_{sp}$:

1. Increase the exhaust temperature, thereby increasing $V_{exh}$, and/or
2. Operate the reactor at low pressure, allowing the $H_2$ to dissociate into $H_1$, reducing the mass of the exhaust particles, causing their velocity to increase. Operation of this design at high temperature and low pressure result in a large value of $I_{sp}$ (see Design Parameters).

Research carried out under the auspices of the U.S. Department of Energy under contract No. DE-AC02-76CH00016.

### B. Cooling

In the Particle Bed Reactor, the spherical fuel particles have a high surface to volume ratio resulting in very efficient cooling.

## III. DESCRIPTION

### A. Reactor Core

The Particle Bed Reactor (Figure 1) consists of 37 fuel elements imbedded in a cylinder of solid beryllium, which acts as a moderator and reflector. The fuel elements are cylinders which are distributed in a hexagonal array.

### B. Fuel Elements

Each fuel element (Figure 2) consists of four co-axial cylindrical shells. From the largest radius to the smallest, they are:

1. Inlet Plenum
2. Cold Frit
3. Fuel Bed
4. Hot Frit

The frits are porous cylindrical walls which hold the fuel bed.

### C. Fuel

The fuel consists of a packed bed of spherical fuel particles (Figure 3). Each particle consists of four spherically symmetric regions. From the smallest radius to the largest, they are:

1. $UC_2$/ZrC Fuel Kernel
2. Porous Graphite
3. Pyrolytic Graphite
4. ZrC Coating, which does not react with $H_2$.

Heat is generated by fission and the system is cooled by $H_2$, which flows radially inward from the inlet plenum, through the porous cold frit, over the packed fuel bed, through the porous hot frit and is exhausted axially out the channel formed by the hot frit.

### D. Design Parameters

| | |
|---|---|
| Power (MW) | 1000. |
| Fuel Bed Power Density (MW/l) | 5. |
| Outlet Frit Mach Number Limit | .3 |
| Outlet Temperature (K) | 2000-4000 |
| Outlet Pressure (ATM) | 5. |
| Estimated $I_{SP}$(s) | 900-1600 |
| Pitch (CM) | 19.43 |
| Height (CM) | 124. |
| Core Radius (CM) | 68. |
| Vessel Thickness (CM) | 1. |
| Fuel Bed Mass (Kg) | 496. |
| Moderator Mass (kg) | 1767 |
| Miscellaneous [Vessel, etc.] (kg) | 700. |
| Estimated Total Mass (Kg) | 3000. |

## IV. METHOD

Various computer codes were used in the analysis.

### A. Neutronics

One set dealt with the neutronics. A Monte Carlo neutron transport code was used to determine the criticality and the heating rates of the system.

### B. Thermal Hydraulics and Heat Transfer

A second set dealt with the thermal hydraulics and heat transfer. Conservation of energy, and transfer of heat, from the solid material to the $H_2$ coolant, were incorporated into a finite element description of the reactor.

## V. RESULTS

### A. Neutronics

1. The neutronic analysis showed that the proposed reactor design is neutronically critical. $K_{eff}$ is easily changed by varying the size of the fuel kernel (but keeping the particle O.D. constant), thereby changing the fuel loading.

2. Despite the neutron streaming out of the exhaust holes, the reactor is critical.

### B. Heat Transfer

1. The structural components and the moderator would absorb about 40 MW of power and would be readily cooled by the $H_2$ before it enters the fuel bed.

2. Despite the low pressure of the system, high power is produced because of the favorable heat transfer properties of the fuel particles.

## VI. CONCLUSION

The Ultra High Temperature Particle Bed Reactor is suitable as a propulsion unit for a mission to Mars.

## REACTOR CROSS SECTION

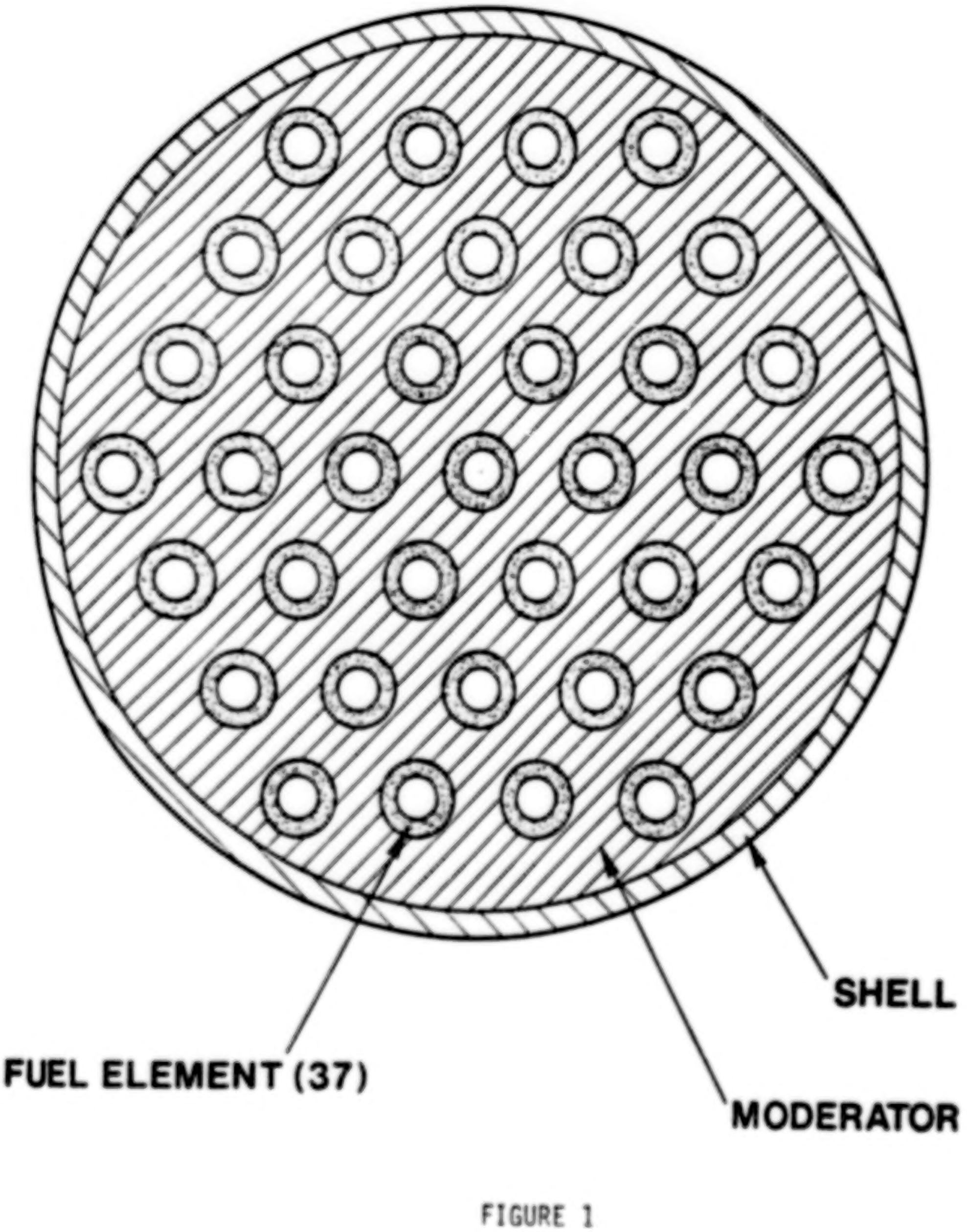
SHELL
FUEL ELEMENT (37)
MODERATOR

FIGURE 1

## UNIT CELL

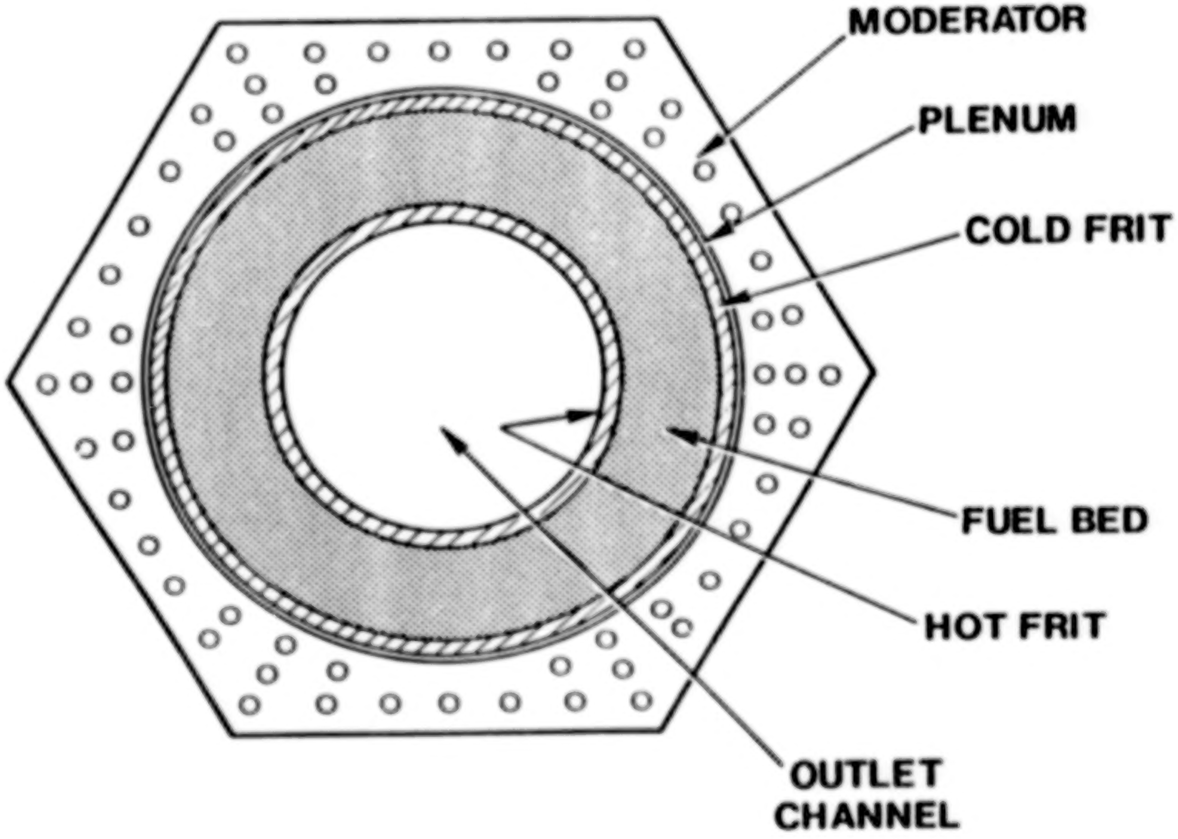
MODERATOR
PLENUM
COLD FRIT
FUEL BED
HOT FRIT
OUTLET CHANNEL

FIGURE 2

## FUEL PARTICLE

OVERALL O.D. 500$\mu$m

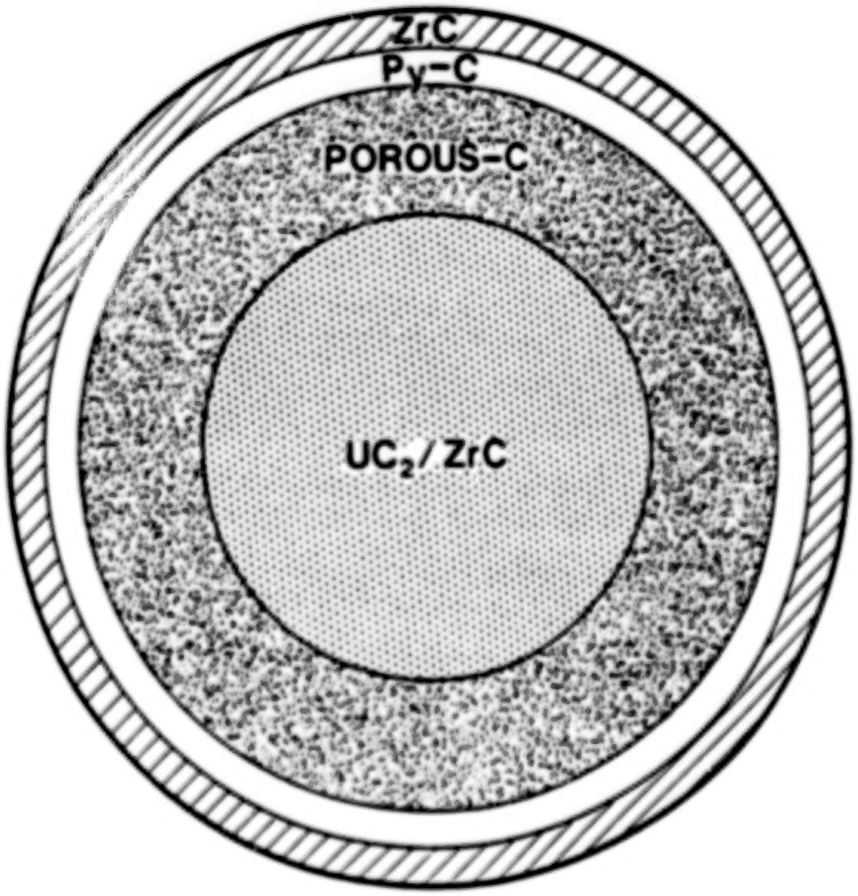
ZrC
Py-C
POROUS-C
UC₂ / ZrC

FIGURE 3

N91-22145

# The Liquid Annular Reactor System (LARS) Propulsion

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Roger Lenard, SDIO
The Pentagon, Washington, D.C.

A new concept for very high specific impulse (>2000 seconds) direct nuclear propulsion is described. The concept, termed LARS (Liquid Annular Reactor System) uses liquid nuclear fuel elements to heat hydrogen propellant to very high temperatures (~6000K). Operating pressure is moderate (~10 atm), with the result that the outlet hydrogen is virtually 100% dissociated to monatomic H. The molten fuel is contained in a solid container of its own material, which is rotated to stabilize the liquid layer by centripetal force. LARS reactor designs are described, together with neutronic and thermal-hydraulic analyses. Power levels are on the order of 200 megawatts. Typically, LARS designs use 7 rotating fuel elements, are beryllium moderated and have critical radii of ~100 cm (core L/D ≈1.5).

## INTRODUCTION

As illustrated in Figures 1 and 2, the LARS fuel element consists of a rotating cylindrical can that holds an inner annular layer of a high temperature refractory material. This refractory contains uranium and an appropriate diluent(s), possibly a mixture of $UC_2$ and ZrC. The thin outer layer of the refractory, i.e., the portion adjacent to the fuel element can is solid, while the inner layer, i.e., the portion adjacent to the central channel where the hydrogen propellant flows, is liquid. Heat is generated in the refractory by fissioning of uranium. The bulk of this heat is transported by convective mixing in the annular liquid layer to the inner boundary between the flowing hydrogen propellant and the liquid. Here, heat flows into a seeded (e.g. with micron size tungsten particles) hydrogen propellant by a combination of radiant and convective transfer. Heat transfer rates to the propellant is very high, typically on the order of 5-10 kilowatts per $cm^2$ of surface area.

The liquid refractory is maintained as an annular layer by rotating the fuel element can at a speed sufficient to stabilize the molten fuel layer. The can is metal (beryllium) and is kept cool by heat transfer to the hydrogen propellant. The thermal conduction rate to the can from the fuel layer is relatively small, on the order of 100 watts per $cm^2$.

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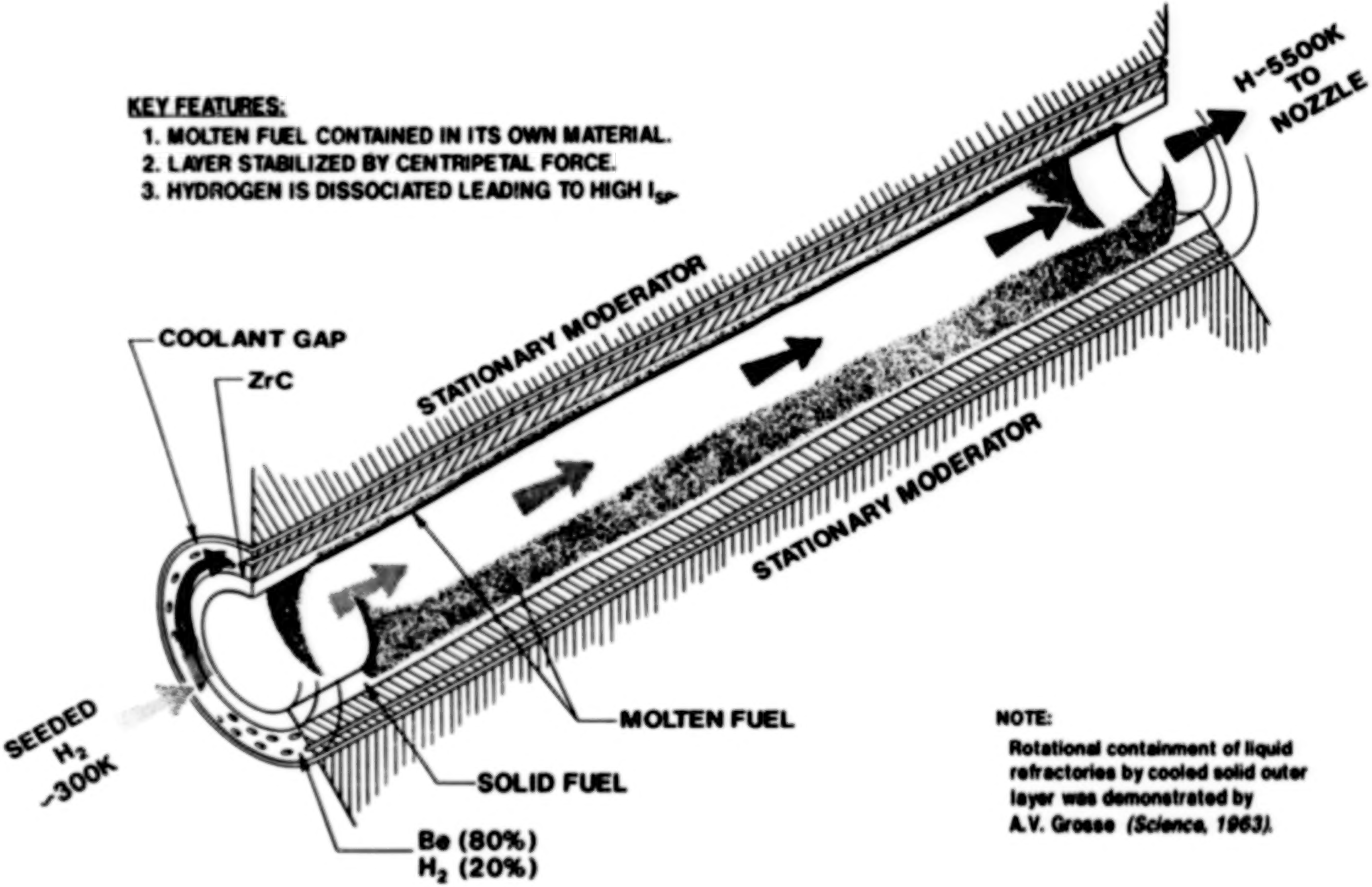
Research carried out under the auspices of the U.S. Department of Energy under contract No. DE-AC02-76CH00016.



COOLANT GAP
ZrC
STATIONARY MODERATOR
STATIONARY MODERATOR
H~5500K TO NOZZLE
SEEDED $H_2$ ~300K
MOLTEN FUEL
SOLID FUEL
Be (80%)
$H_2$ (20%)

**KEY FEATURES:**

1. MOLTEN FUEL CONTAINED IN ITS OWN MATERIAL.
2. LAYER STABILIZED BY CENTRIPETAL FORCE.
3. HYDROGEN IS DISSOCIATED LEADING TO HIGH $I_{sp}$.

**NOTE:**
Rotational containment of liquid refractories by cooled solid outer layer was demonstrated by A.V. Grosse *(Science, 1963)*.

Figure 1 LARS concept.

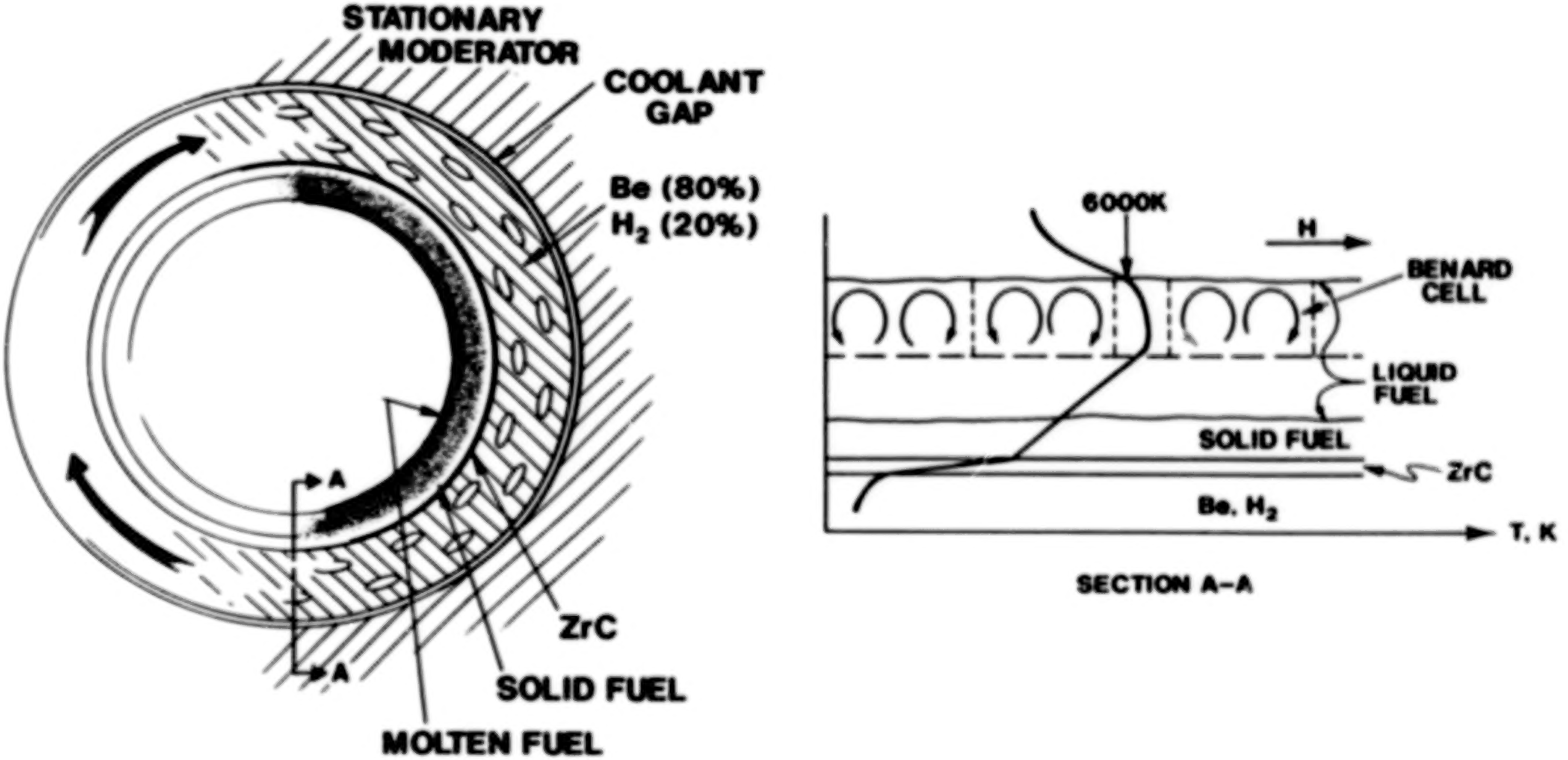

STATIONARY MODERATOR
COOLANT GAP
Be (80%)
$H_2$ (20%)
ZrC
SOLID FUEL
MOLTEN FUEL
A
A
6000K
H
BENARD CELL
LIQUID FUEL
SOLID FUEL
ZrC
Be, $H_2$
T, K
SECTION A–A

Figure 2 LARS rotating fuel element.

Rotational containment of high temperature liquid refractories by creation of a cooled solid outer layer was first demonstrated by A.V. Grosse (ref. 1). Containment of liquid refractories in cooled metal containers is also found in the vacuum arc melting industry, though here the containers are not rotated.

## CONCEPTUAL REACTOR DESIGN

The reactor design is controlled by heat transfer and physics considerations. For a given total power (P) output, the radiant heating of the propellant must be sufficiently efficient to extract the heat at the desired flow rate. Convective heat transfer is, of course, also present, however, for the conditions of LARS, it is an order of magnitude smaller than the radiation. The efficiency of the radiation heat transfer process is controlled primarily by the emissivity of the molten fuel surface and the absorptivity of the gas volume. This efficiency will be represented by a multiplicative parameter (f) which can be varied over a reasonable range. Thus, in order to estimate the required heat transfer area and thus the outlet duct size, the radiant heat flux (Q) will be estimated from

$$Q = f \sigma T^4 \tag{1}$$

where T is the surface temperature.

The factor f is the amount of heat transferred by radiation from the liquid surface to the seed particles divided by blackbody radiation at the same temperature to a perfectly absorbing gas. If we assume that the liquid surface is "gray", f can be expressed as (Ref. 2):

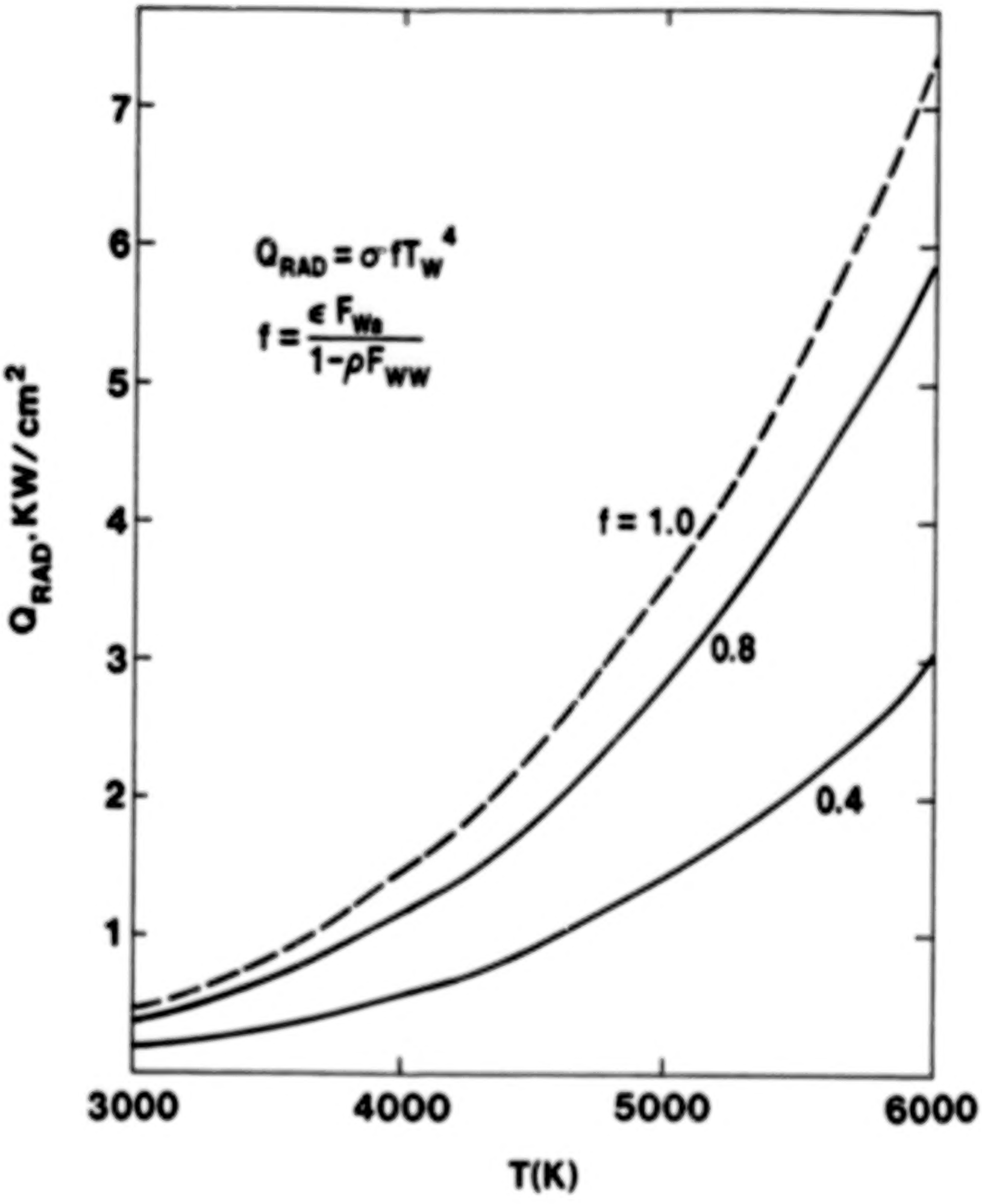
$$f = \frac{\epsilon F_{wa}}{1-\rho F_{ww}} \tag{2}$$

where $\epsilon$ is the emissivity of the surface, $\rho$ its reflectivity and $F_{wa}$ and $F_{ww}$ are the view factors from surface to wall and wall to wall, respectively. Calculated values of the latter for a range of geometries are presented in Reference 2.

Considering reasonable absorption parameters of particle-seeded gases (Ref. 3) and the observed emissivities of molten refractories, it was concluded that a practical range of f is from 0.4 to 0.8. Consequently, the radiative surface heat flux was calculated for these values of f and a temperature range of 3000 to 6000K. These results are shown in Figure 3.

One can calculate the dimensions of the reactor if one assumes that the duct radius and height are related by a multiplicative constant (n). Thus, the duct area (A), for N fuel elements is given by

$$A = 2\pi N n r^2 \tag{3}$$



**NOTE:**

$\epsilon$ = emissivity of liquid wall

$\rho$ = reflectivity of liquid

$F_{Wa}$ = view factor wall to aerosol (Fig. 2 NASA CR-953)

$F_{WW}$ = view factor wall to wall (Fig. 3 NASA CR-953)

Figure 3 Radiative heat transfer from liquid surface to hydrogen.

and the required duct radius for a given P and Q can be calculated as

$$r = \left[\frac{P}{Q\ 2\pi nN}\right]^{1/2} \qquad (4)$$

Based on the above relationships and assumptions, it is possible to investigate the variation of r with T for various values of f. In all cases, it will be assumed that $n = 24$, $N = 7$ and $P = 200$ MW. The implied outlet duct radii are shown on Figure 4. It can be seen that they range from a maximum of 32 cm for the lower temperature, to a minimum of 5.65 cm for the highest temperatures. The implied reactor height (given by $n = 24$) varies from 768 cm for the lowest temperature to 135.6 cm for the highest values of temperature and f values. In order to completely describe the reactor, an assumption must be made regarding the ratio between the outside fuel bed diameter and the fuel element pitch. The appropriate value for the pitch/diameter (P/D) ratio is a function of the moderator and its magnitude is chosen based on experience. Finally, a reflector thickness and a cooling duct geometry within the moderator and reflector must be assumed.

The above information describes the reactor system in sufficient detail in order that an estimate of its multiplication factor ($K_{eff}$) can be made. Due to the extreme heterogeneous nature and the high neutron leakage, these reactors can only be analyzed using an explicit geometrical representation of the core. These analyses were carried out using the MCNP Monte Carlo code. Two reactors, both operating as $T = 6000$K but having f values of 0.4 and 0.8, respectively, were considered. Detailed dimensions and uranium carbide masses for these two systems are shown on Table I. Some of these dimensions are also illustrated in Figure 5. From Figure 4, it can be seen that if the value for f is allowed to vary from 0.4 to 0.8 (dashed line), a family of reactors is defined for which $Q \approx 3.0$ kw/cm$^2$ and whose temperature varies from 6000K to 5150K. In other words, any improvement in the effectiveness of radiative heat transfer f, permits operation at a lower surface temperature. All these reactors are critical and have the same geometrical dimensions. A larger family of reactors can be defined by the shaded area in Figure 4. These should all be critical and operate at various values of temperature T and heat transfer parameter f, i.e. $5150 \leq T \leq 6000$K and $.4 \leq f \leq 0.8$. Although larger reactors operating at lower temperatures are possible, these begin to lose the unique advantages of the LARS concept.

## TECHNICAL ISSUES

The major technical issues that must be addressed in the design of LARS are the following:

1. Stability of liquid layer

    a. Effects of rocket acceleration
    b. Helmholz instability
    c. Convective cells (Bénard Problem)

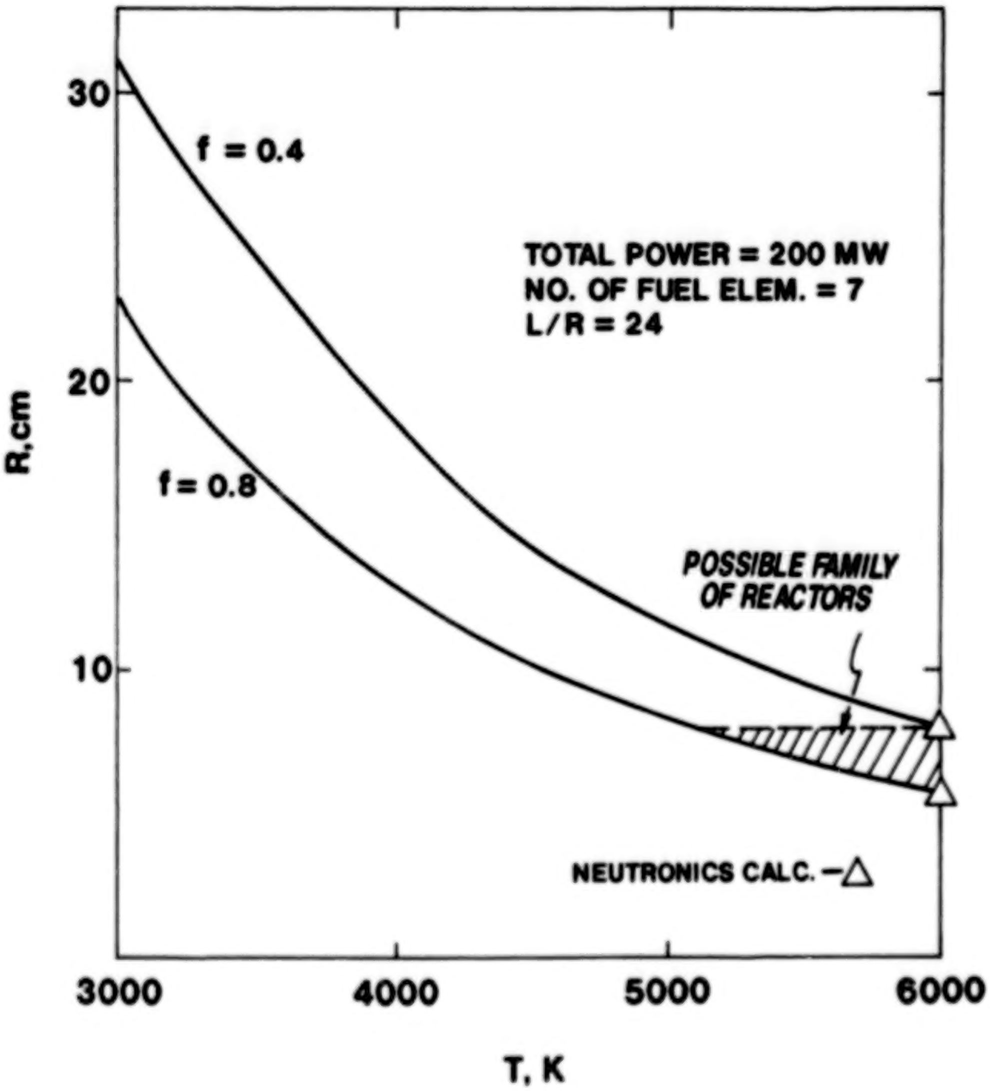


Figure 4 Flow channel radius vs. temperature

## TABLE I. REACTOR PARAMETERS

| | f - 0.4 | f - 0.8 |
|---|---|---|
| TOTAL POWER (MW) | 200. | 200. |
| OUTLET TEMPERATURE (K) | 6000. | 6000. |
| OUTLET PRESSURE (ATM) | 10. | 10. |
| NO. OF ELEMENTS | 7 | 7 |
| OUTLET DUCT RADIUS (CM) | 8.0 | 5.6 |
| FUEL BED RADIUS (CM) | 9.4 | 8.1 |
| ROTATING DRUM RADIUS (CM) | 12.4 | 11.1 |
| ELEMENT PITCH (CM) | 30.0 | 30.0 |
| REACTOR O.D. (CM) | 110.0 | 110.0 |
| REACTOR HEIGHT (CM) | 192.7 | 135.5 |
| TOP REFLECTOR (CM) | 3.0 | 3.0 |
| BOTTOM REFLECTOR (CM) | 5.0 | 5.0 |
| RADIAL REFLECTOR (CM) | 10.0 | 10.0 |
| MASS OF $UC_2$ (KG) | 30.0 | 30.0 |
| MULTIPLICATION FACTOR, $K_{eff}$ | 1.08 | 1.10 |

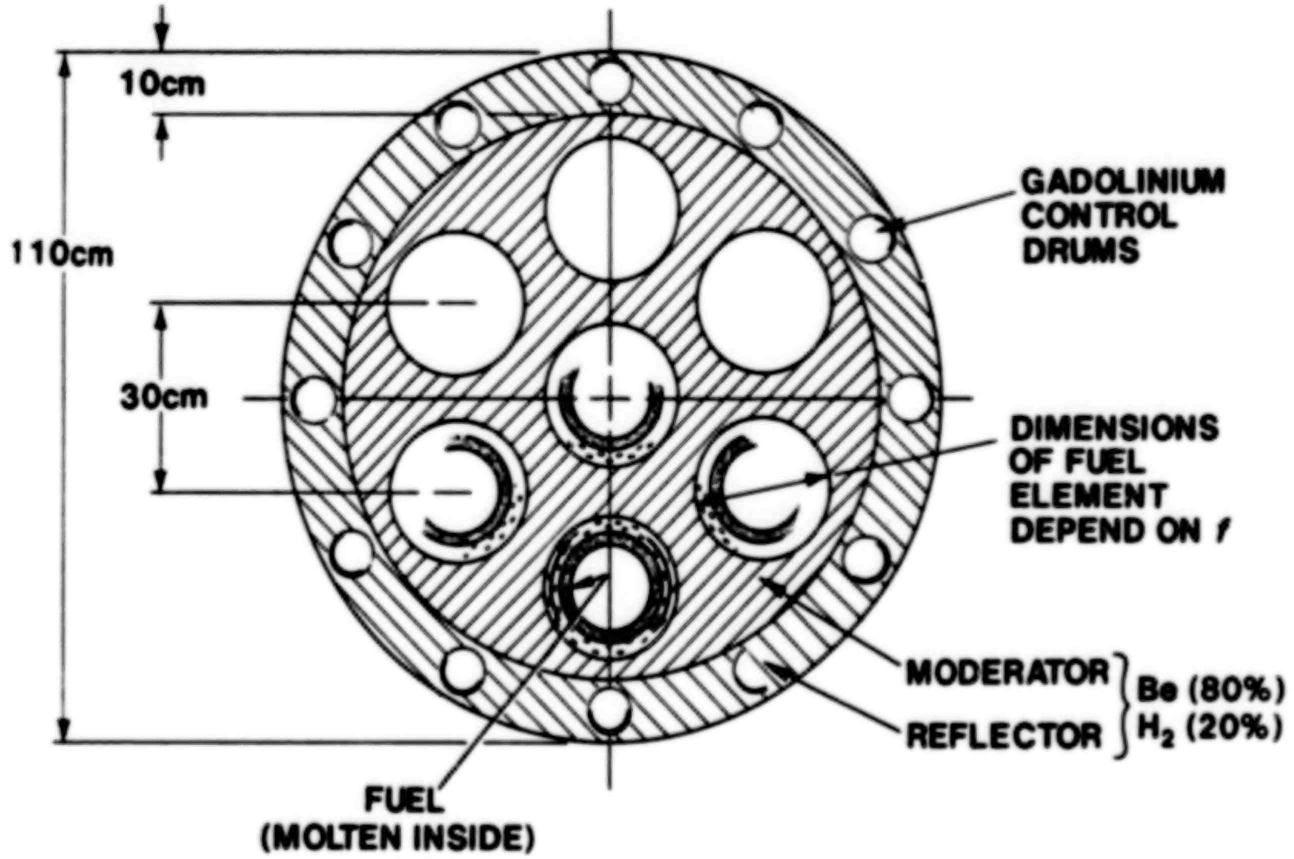

10cm
110cm
30cm
GADOLINIUM CONTROL DRUMS
DIMENSIONS OF FUEL ELEMENT DEPEND ON f
MODERATOR
REFLECTOR
Be (80%)
$H_2$ (20%)
FUEL (MOLTEN INSIDE)

Figure 5 LARS design cross section.

2. **Heat Transfer**

    a. Combined radiation and convection
    b. Seeding of $H_2$ to increase radiation
    c. Moderator cooling
    d. Nozzle cooling

3. **Physics**

    a. High temperature cross sections
    b. Heterogeneous core - will require critical experiment
    c. Reactor startup

4. **Materials**

    a. Radiative properties of liquid fuel
    b. Compatibility with $H_2$
    c. Evaporative loss of fuel
    d. Resistance to radiation damage

## CONCLUSIONS

1. The LARS molten core concept permits operation with hydrogen coolant at high enough temperatures to dissociate.

2. Specific impulse can be doubled relative to "conventional" nuclear rockets.

3. Development of LARS allows high $\Delta V$ missions (outer planets, fast trip times, etc.) using low weight, high thrust, high $I_{sp}$ propulsion.

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N91-22146

# NUCLEAR THERMAL ROCKETS USING INDIGENOUS EXTRATERRESTRIAL PROPELLANTS

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## ABSTRACT

This paper presents a preliminary examination of a novel concept for a Mars and outer solar system exploratory vehicle. Propulsion is provided by utilizing a nuclear thermal reactor to heat a propellant volatile indigenous to the destination world to form a high thrust rocket exhaust. Candidate propellants whose performance, materials compatibility, and ease of acquisition are examined include carbon dioxide, water, methane, nitrogen, carbon monoxide, and argon. Ballistic and winged supersonic configurations are discussed. It is shown that the use of this method of propulsion potentially offers high payoff to a manned Mars mission, both by sharply reducing the initial mission mass required in low Earth orbit, and by providing Mars explorers with greatly enhanced mobility in traveling about the planet through the use of a vehicle that can refuel itself each time it lands. Utilizing the nuclear landing craft in combination with a hydrogen fueled nuclear thermal interplanetary vehicle and a heavy lift booster, it is possible to achieve a manned Mars mission in one launch. Utilizing such a system in the outer solar system, it is found that a low level aerial reconnaissance of Titan combined with a multiple sample return from nearly every satellite of Saturn can be accomplished in a single launch of a Titan IV or STS. Similarly a multiple sample return from Callisto, Ganymede, and Europa can also be accomplished in one launch of a Titan IV or STS.

## INTRODUCTION

Interplanetary travel and colonization can be greatly facilitated if indigenous propellants can be used in place of those transported from Earth. Nuclear thermal rockets, which use a solid core fission reactor to heat a gaseous propellant, offer significant promise in this regard, since, in principle, any gas at all can be made to perform to some extent. In this paper we present a preliminary examination of the potential implementation of such a concept in the context of manned Mars missions. The vehicle in question we hereby christen the NIMF, for Nuclear rocket using Indigenous Martian Fuel.

## CANDIDATE MARTIAN PROPELLANTS

The atmosphere of Mars consists of 95.0% carbon dioxide, 2.7% nitrogen, and 1.6% argon, all of which are candidate fuels for a NIMF. Water could also be used after harvesting ice or permafrost. Carbon monoxide could be manufactured by stripping $CO_2$, and could either be used as a propellant directly, or reacted with water to produce methane propellant.The following chart shows the ideal specific impulse obtainable with each of the above propellants at various temperatures.

Table 1

Ideal Specific Impulse of Martian Propellants

| Temperature | $CO_2$ | Water | Methane | CO or $N_2$ | Argon |
|---|---|---|---|---|---|
| 2800 K | 283 | 370 | 606 | 253 | 165 |
| 3000 K | 310 | 393 | 625 | 264 | 172 |
| 3200 K | 337 | 418 | 644 | 274 | 178 |
| 3500 K | 381 | 458 | 671 | 289 | 187 |

In the table above, 2800 K may be regarded as a safe operating temperature, as NERVA carbide (Koenig, 1986)[3], uranium-thorium oxide, and cermet (Cowan, et al., 1988)[1] fuel elements have been extensively and successfully tested in this range. Some of the final NERVA tests and cermet data both indicate that 3200 K may eventually be attainable. The final temperature of 3500 K can be taken as a ultimate upper limit to what a solid core nuclear rocket may be expected to achieve.

We now examine the characteristics of each of the candidate propellants.

## Carbon Dioxide

Carbon Dioxide is the most readily accessible of all the candidate martian propellants. Composing 95% of the atmosphere, it can be obtained by pumping the martian air into a tank. At a typical martian temperature of 233 K, carbon dioxide liquifies under a pressure of 10 bars. Under these conditions, assuming an isothermal compression process, liquid $CO_2$ can be manufactured for an energy cost of just 84 kW-hrs per metric ton. The NIMF engine produces over a thousand MW (thermal). If an electrical capacity of 1 MWe is built in as well, then the (2800 K, 40 MT) NIMF would be able to fuel itself for a flight into a high orbit in less than 14 hours! Liquid $CO_2$ has a density 1.16 times that of water and is eminently storable under martian conditions.

A 40 MT $CO_2$ propelled NIMF vehicle with a specific impulse of 280 seconds would require a mass ratio of 3.8 to ascend to Low Mars Orbit, or 8.3 to fly directly from the Mars surface into a Hohmann Transfer orbit to Earth, both of which are attainable due to the high mass density of liquid $CO_2$. Reactor power levels of about 1100 MWth would be required to generate sufficient thrust for the ascent to orbit, while 2400 MWth would be required for the direct Trans-Earth Injection mission.

A $CO_2$ NIMF operating in a variety of modes is depicted in fig. 1.

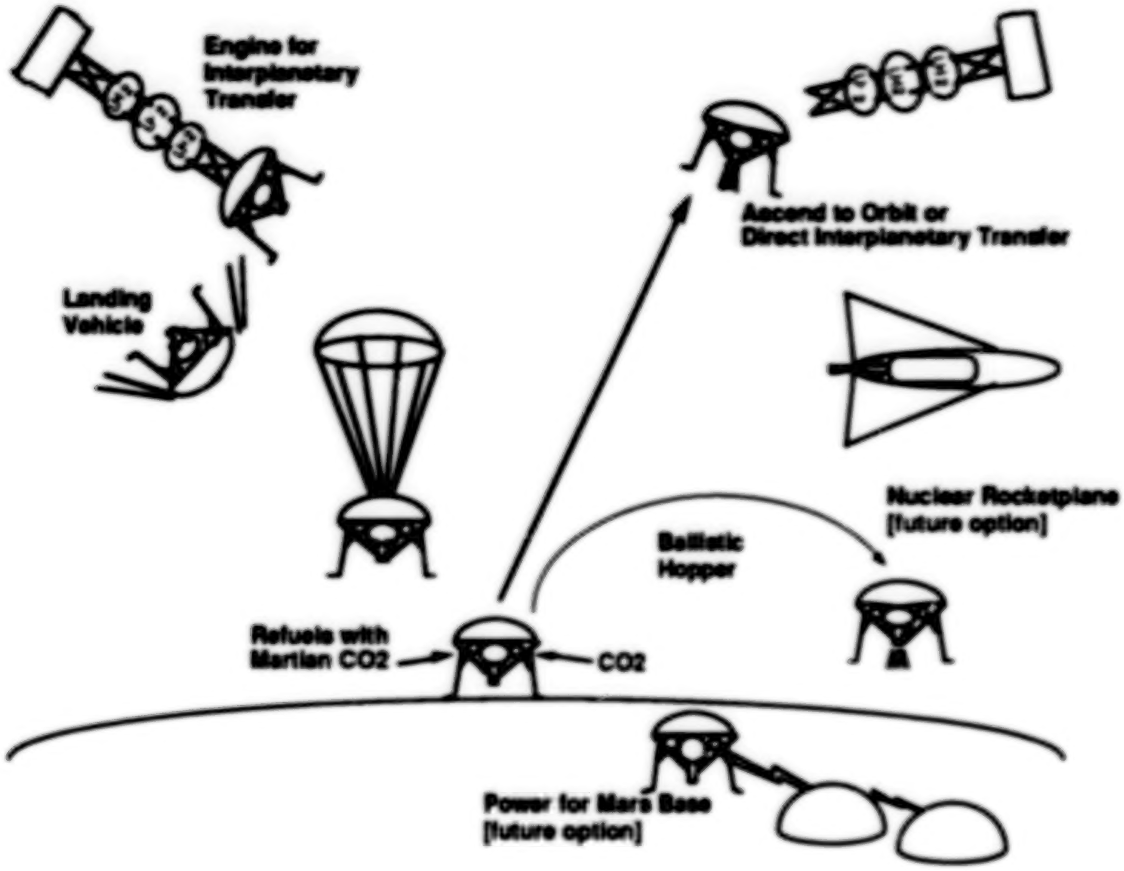
Engine for Interplanetary Transfer
Ascend to Orbit or Direct Interplanetary Transfer
Landing Vehicle
Nuclear Rocketplane [future option]
Ballistic Hopper
Refuels with Martian CO2
CO2
Power for Mars Base [future option]

Figure 1 The NIMF (Nuclear rocket using Indigenous Martin Propellants)

Since $CO_2$ is so readily acquired, it is quite convenient for use for multiple suborbital hops, allowing a Mars exploration mission to visit many sites. The vehicle for such an application (which would also function as the surface to orbit ascent vehicle) could take the form of either a ballistic hopper or a supersonic (Mach 4-5) winged aircraft. Such an aircraft could function either as a pure rocketplane, or use airborne jet intake of $CO_2$ to extend its range. Because of the high speeds and power levels available from NIMF propulsion, wing sizes could be quite modest, similar to those on the Space Shuttle orbiter. Landing and takeoff would require VTOL ability, and be accomplished either in the manner of the Harrier or the X-13.

At high temperatures, both carbon dioxide and water become oxidizers, making it unlikely that a $CO_2$ or water NIMF could utilize the same carbide fuel elements developed for the NERVA hydrogen propelled nuclear engine program. Instead, oxide fuel elements would have to be used. Fuel pellets composed of a combination of uranium-thorium oxide have been made with melting points above 3300 K. If coated with another oxide to prevent fission product migration into the propellant exhaust, such pellets should be able to sustain $CO_2$ or water driven NIMF engines with propellant temperatures of about 3000 K. The disadvantage of such oxide fuel pellets is that they would probably not be compatible with hydrogen propellant, in which case a high Isp interplanetary transfer vehicle would have to employ a separate NERVA type engine.

### Water

While involving greater uncertainty and complexity in its acquisition, the use of water propellant allows for remarkable performance. For example, a 3000 K water propelled NIMF taking off from the martian surface for Low Earth Orbit would have a mass ratio of 5.4 and require about 2000 MWth for liftoff. If a base on Phobos is used as a point of departure, a water propelled NIMF would be able to fly to Earth, aerobrake into a loosely bound orbit, and return to Mars without refueling.

The main problem with the use of water is finding it, and the second is harvesting it. It is believed by many planetary scientists that vast quantities of water may exist on Mars in the form of permafrost covered over by a few feet of sand. After all, the planet once had flowing rivers. The existence of such quantities of water on Mars may be verified by the unmanned probes planned by the U.S. and the Soviets for the 1990s. At the present, the only large sources of water known for certain to exist on Mars is in the north polar cap, which however is a very inconvenient place from which to launch into an orbit useful for Earth return, as the required inclination change is large. If permafrost is discovered, water will become more generally available, but it will require an operation of some complexity to harvest it. It is therefore difficult to see how an initial manned mission could be planned based on the assumption of securing water fuel for the return trip. However, once a martian base is established, locally mined water could function as a near ideal fuel for both Earth return, near Mars, and beyond Mars operations.

### Methane

If water is acquired on Mars, then methane can be produced (along with oxygen) by using heat from the nuclear reactor to strip CO from martian $CO_2$, and then reacting the CO with $H_2O$ in the water gas shift reaction to produce hydrogen and $CO_2$. Some of the $CO_2$ is recycled to be stripped, and the remainder is then catalytically reacted with the hydrogen to produce methane and water. As can be seen from Table 1., methane is an excellent candidate propellent for a NIMF vehicle, yielding specific impulses well in excess of 600 seconds. Furthermore, since it does not contain oxygen, the use of methane eliminates one of the major problems associated with either $CO_2$ or $H_2O$, namely oxygen attack, and would be compatible with conventional NERVA carbide fuel elements. Methane, however, fully dissociates at temperatures of interest for nuclear propulsion, and the free carbons thus created may cause coking problems. This is a question that must be resolved experimentally.

Liquid methane would have to kept refrigerated on Mars, but this is not expected to present significant difficulties. Methane liquifies at 135 or 166 K, at 5 or 20 atmospheres pressure, respectively.

### Other NIMF Propellants

Nitrogen, carbon monoxide, and argon are also potential NIMF propellants. However, as can be seen from table 1., their performance is inferior to that of the much more accessible $CO_2$. Compared to the 84 kWe/MT cost of liquifying $CO_2$, these propellants require about 5 to 10 MWe/MT to produce (Meyer and McKay, 1984)[5]. In addition, these three propellants are all moderately cryogenic, requiring storage temperatures in the 100 K range. The primary advantage of these fuels over $CO_2$ is their lack of chemical reactivity with fuel or cladding materials that are also compatible with hydrogen. Thus the same reactor which uses carbon monoxide for propellant for ascent to orbit could also use hydrogen propellant, taking advantage of its 950 second Isp for interplanetary orbital transfers.

Finally it should be noted that if water is available, it is possible to combine hydrogen from the water with nitrogen to form ammonia, a non cryogenic, oxygen free propellant capable of yielding an Isp of around 450 seconds at reactor temperatures of 3000 K. The processes involved may be excessively complex and energy expensive, however.

## NIMF CONFIGURATIONS

Artist's conceptions of two alternative NIMF configurations are depicted in figs. 2 and 3. The first, (fig. 2)by Martin Marietta artist Robert Murray, shows a NIMF ballistic vehicle standing on the martian surface. The nuclear engine is at the bottom, and is surrounded by a coaxial fuel tank, which when filled, augments the solid lithium/tungsten shadow shield with a several meter thick four-pi shield of liquid $CO_2$. This liquid coaxial shield facilitates human operation on the surface in the immediate vicinity of the NIMF by protecting against gamma rays released by fission decay products. Positioned above the shadow shield is the main spherical fuel tank, above which is the machine deck incorporating the $CO_2$ intake pumps. Above this are the habitation and command decks, and at the very top is a storage dome. Thus, when the main engine is firing the crew is protected from radiation by the shadow shield, the massive amount of liquid $CO_2$ in the main tank, by the structures in the machine deck and the lower habitation deck, and by distance. The NIMF's fuselage acts as an aerobrake, with a L/D approaching unity.

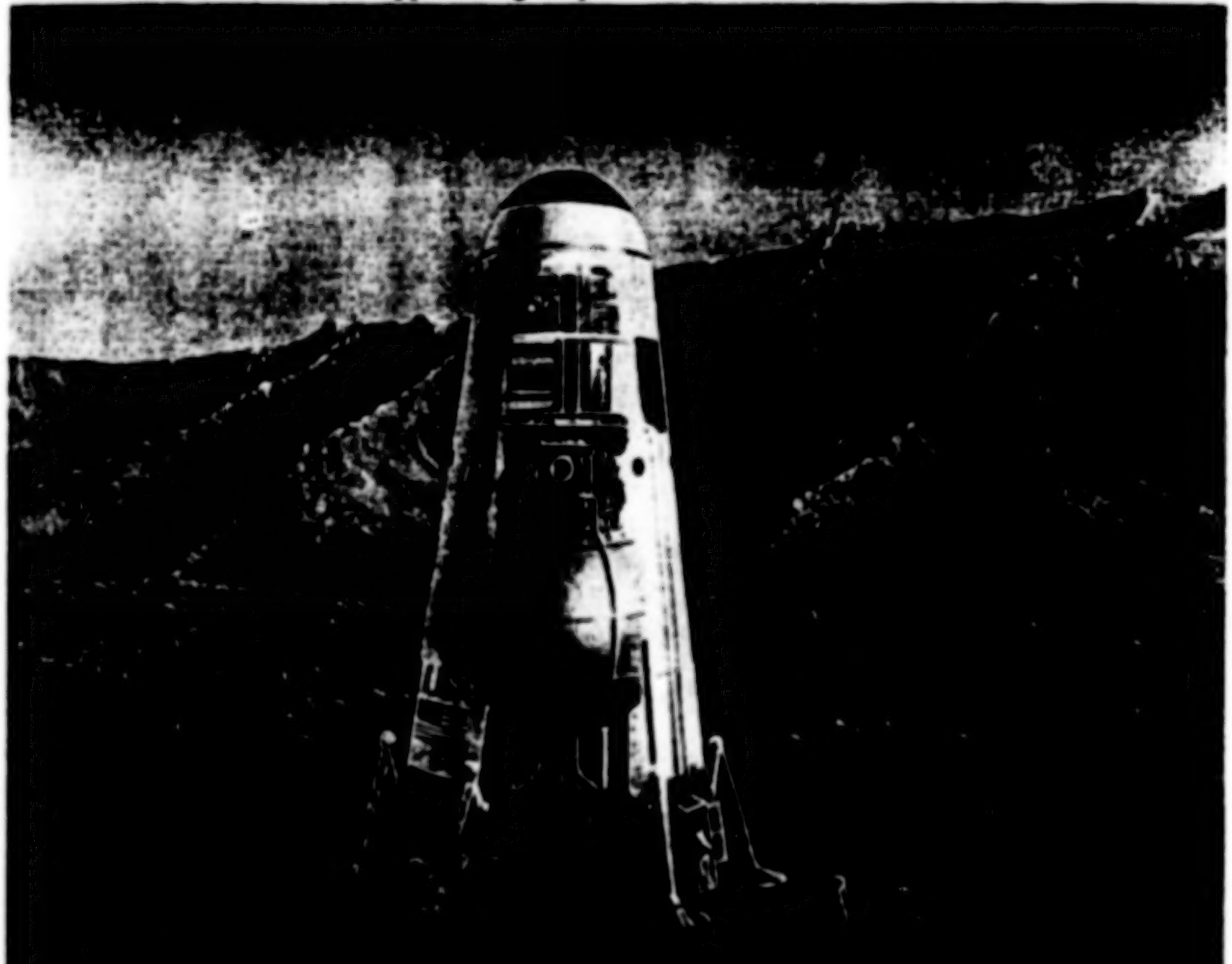

Figure 2 NIMF Ballistic Ascent/Descent Vehicle on the Martian Surface

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The second drawing (fig. 3), also by Murray with the assistance of free lance artist Jeff Danelek, depicts a winged NIMF rocketplane on the surface of Mars. Once again, the reactor shadow shield is supplemented by a coaxial four-pi liquid shield, by the main fuel tank forward of the reactor, and by a machine compartment. Because the craft flies at supersonic speeds (long range level flight at Mach 4), the wings are fairly modest deltas, very unlike the delicate albatross like ultra-high L/D wings proposed for martian aircraft in the past. The winged NIMF's L/D is equal to 4, and the craft, which has orbital capability, functions as its own aerobrake, much in the manner of the Space Shuttle Orbiter. Landing is accomplished by increasing the angle of attack to maintain lift as the glide decelerates, until a speed of about Mach 1 is attained. At this point, the craft flares up and fires its 4 VTOL rockets positioned on its underside to eliminate its remaining forward velocity and accomplish a Harrier like landing. The VTOL rockets generate thrust by releasing hot $CO_2$ gas which is piped to them from the reactor outlet. Cargo, land rovers, and crew can conveniently exit onto the martian surface by means of the landing ramp which lowers from the forward stowage area, just below the control deck and forward of the habitation compartments.

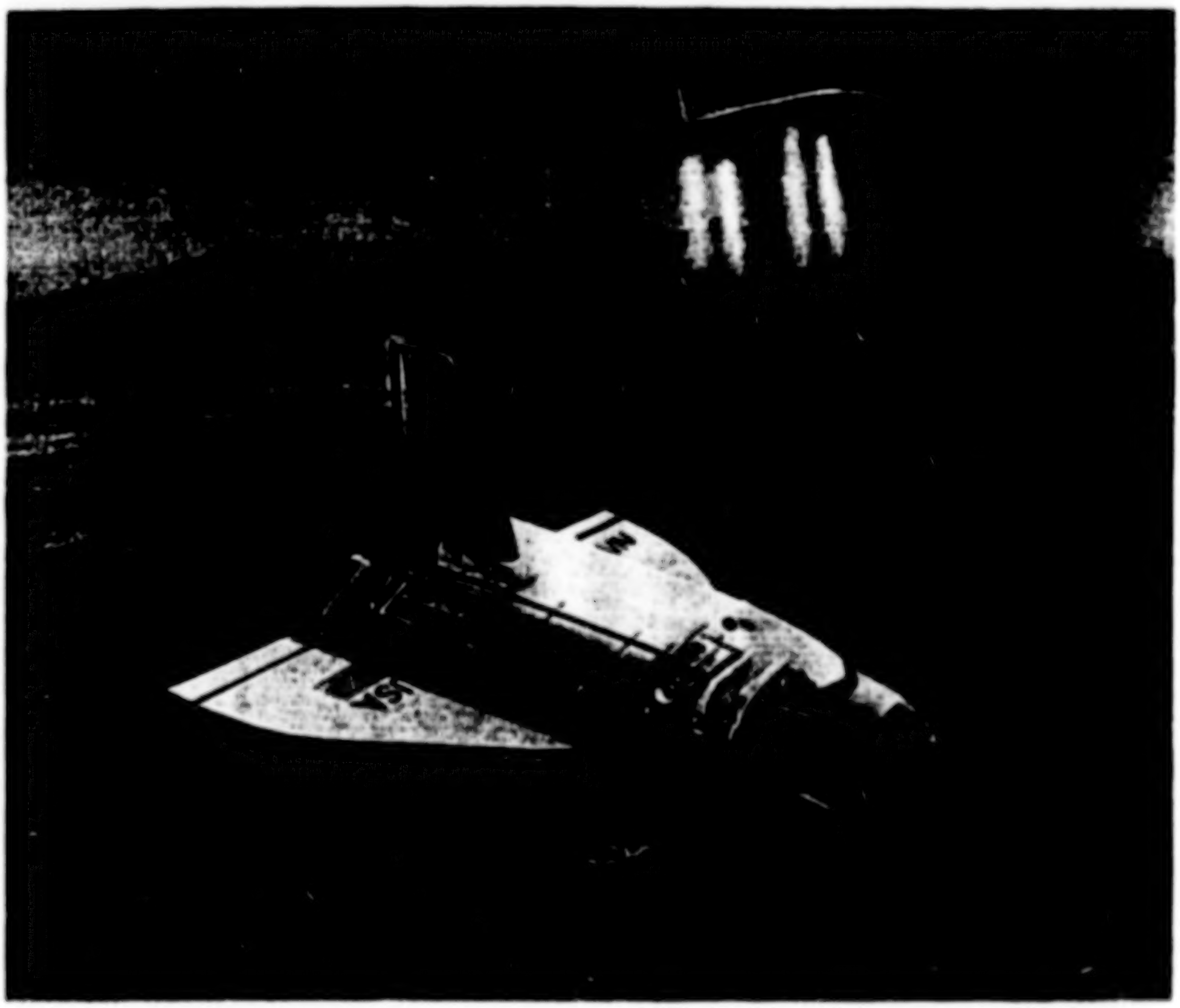

Figure 3 NIMF rocketplane. Aerodynamic flight at Mach 4 allows modest wing area. Takeoff and landing is accomplished using 4 ventral VTOL thrusters.

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# A MANNED MARS MISSION IN A SINGLE LAUNCH

Since the days of the Apollo program, NASA's thinking about manned planetary landings has been dominated by approaches based on a combination of an orbiting mothership containing long term living quarters and a small landing craft, a fraction of which manages to ascend to orbit after a stay on the surface. The reason for such an approach has always been the fact that any mass lowered into a planet's gravity well that requires return to space will require additional fuel mass to accomplish such an ascent. Furthermore, since this fuel mass itself must be transported from Earth, which requires still more fuel, ultimately the entire mass and cost of the mission is multiplied. With the advent of the NIMF, however, such logic is no longer valid. In fact, since any mass landed upon Mars can be lifted back to orbit using readily available indigenous propellant, it becomes advantageous to abandon the concept of the orbiting mothership altogether, and instead land the entire spacecraft living quarters on the planet's surface. In other words, the NIMF and the interplanetary vessel are one and the same. All that is left in orbit is an automated vehicle consisting of either a cryogenic or nuclear thermal orbital transfer propulsion unit with associated fuel and tankage.

The use of the NIMF in this, its proper mission architecture was examined in three alternative scenarios. In each of these scenarios, a 40 metric ton NIMF carrying a three person crew was projected out of a 300 km LEO orbit onto a minimum energy trajectory towards Mars. The NIMF lands on Mars, and hops around visiting various sites, ultimately returning to Earth via a Hohmann transfer orbit. The three scenarios examined are given below.

Scenario 1: An expendable orbital transfer vehicle (OTV) propels the NIMF out of LEO onto a minimum energy transfer orbit to Mars. Upon reaching Mars, the NIMF aerobrakes and lands. The NIMF then explores Mars, hopping around to visit many sites. Finally the NIMF takes off Mars, propelling itself directly into a minimum energy orbit to Earth. Upon reaching Earth, the NIMF aerobrakes into LEO.

Scenario 2: An OTV propels the NIMF out of LEO onto a minimum energy transfer orbit to Mars. Upon reaching Mars, the NIMF and the OTV aerobrake separately, leaving the (automated) OTV in Low Mars Orbit (LMO), while the NIMF lands. After exploring Mars, the NIMF takes off for LMO and rendezvous with the OTV. The OTV then drives itself and the NIMF out of LMO onto a minimum energy orbit towards Earth. Upon reaching Earth, the NIMF and the OTV aerobrake into LEO.

Scenario 3: An OTV drives the NIMF onto a minimum energy transfer orbit to Mars. Upon reaching Mars, the OTV rocket brakes itself 1 km/s into Mars' gravity well, while the NIMF aerobrakes and lands. After exploring Mars, the NIMF ascends to orbit and rendezvous with the OTV. The OTV then drives both onto a minimum energy orbit towards Earth. Upon arrival, the OTV rocket brakes both into LEO.

Scenario 1 would require interplanetary transfer to take place under zero-gravity conditions. In scenarios 2 and 3, various amounts of artificial gravity could be provided by linking the NIMF and the OTV with a tether, separating the two at a distance, and spinning up the assembly.

Three different OTV propulsion systems were considered. The first was a cryogenic LOX/hydrogen engine with a specific impulse of 470 seconds. The second was a NERVA type nuclear thermal engine with an Isp of 950 seconds. The third was a radial flow nuclear thermal rocket (RFNTR) such as that invented by Carl Leyse and his collaborators at the Idaho National Engineering Laboratory, which uses lower chamber pressures and higher temperatures than a conventional NERVA engine to achieve partial hydrogen dissociation and an Isp of 1300 seconds (Leyse, 1988)[4]. The cryogenic propulsion OTV was assumed to consist of several stages, one for each required burn, with each stage having a mass fraction of 0.87. The NERVA and RFNTR were assumed to weigh 5 metric tons each, and additionally require tankage stages filled with hydrogen fuel, with each tankage stage having a mass fraction of 0.83. Aerobrake masses were taken as 0.15 of the maximum masses required to be decelerated.

Given these scenarios and upper stages, the results of the study are given in Table 2. below.

In table 2. all masses are given in metric tons. It can be seen that there are numerous mission architectures where an initial manned Mars mission can be accomplished with a single launch of the STS-Z (125 MT to LEO capacity) or ALS (100 MT to LEO), and even several where an initial mission can be accomplished with a single STS-C (80 MT to LEO) flight. Furthermore, repeat missions (whose requirement is given in the "expended mass" lines in table. 2) in many scenarios can be accomplished with a single refueling flight by an STS, a Titan IV Upgrade, or an STS-C. This is in marked contrast with current NASA Code-Z/Office of Exploration mission plans, which are based on orbiting motherships, and cryogenic propulsion for interplanetary transfer and landing vehicles (NASA Office of Exploration, 1988)[6]. Such plans involve from 700 to well over a thousand metric tons of propellant per mission, requiring 6 or more STS-Z launches per mission! Furthermore, despite their enormous cost and complexity, such mission plans leave the astronaut-explorers relatively impotent to accomplish much in the way of either exploration or development, as their cryogenic landing vehicle will necessarily restrict their visit to one site, and they lack a substantial source of electric or thermal power.

If an unmanned Mars Rover Sample Return (MRSR) is contemplated in place of a manned Mars mission, then the NIMF can be scaled down from 40 MT to 8 MT, and the masses given in Table 2 scaled down by a factor of 5 accordingly. It can thus be seen that there are numerous scenarios where a MRSR mission can be accomplished with a single launch of the current STS (25 MT to LEO capacity) or Titan IV (20 MT to LEO). Such a NIMF MRSR mission would be far superior to the conventional MRSR concept, as it would be able to deliver 2 MT of scientific payload to Mars, collecting samples and leaving behind roving instrument packages at numerous sites all over the planet. In a single one-launch mission, the NIMF MRSR would thus accomplish exploration work equivalent to that which would otherwise require perhaps a dozen conventional MRSR missions, and simultaneously prove in active field service the technology for full scale manned NIMF vehicles to follow.

Table 2

ALTERNATIVE SCENARIOS FOR NIMF MANNED MARS MISSIONS

| | Scen. 1 | Scen. 2 | Scen. 3 |
|---|---|---|---|
| Cryo OTV | | | |
| Mission Mass | 106 | 199 | 495 |
| Expended Mass | 66 | 154 | 445 |
| NERVA | | | |
| Mission Mass | 73 | 100 | 145 |
| Expended Mass | 33 | 53 | 100 |
| RFNTR | | | |
| Mission Mass | 64 | 80 | 104 |
| Expended Mas | 24 | 33 | 59 |
| NIMF Mass Ratio | | | |
| $CO_2$ Propellant | 8.3 | 3.8 | 5.6 |
| $H_2O$ Propellant | 5.4 | 2.9 | 4.0 |
| $CH_4$ Propellant | 2.9 | 1.9 | 2.3 |

The conventional mission plans Code-Z is currently examination offer little potential for human exploration of the Red Planet, and none at all for sustaining a human presence there. By contrast, the one-launch mission architectures made possible by combining the NIMF with a hydrogen fueled nuclear thermal orbital transfer vehicle (either NERVA or, better yet, the RFNTR), will allow ready, repeated, and inexpensive access to Mars, and will open up a new world to human colonization.

## THE NIMF AND THE ISSUE OF GLOBAL ACCESS FOR MARS EXPLORATION

A key requirement for any space transportation architecture designed for the exploration of Mars is that it be able to achieve "global access," which means planetwide mobility for scientific exploration and for long distance transportation of indigenous materials.

In the past, it has been frequently suggested[2] that the mission of global access could be performed by a chemical rocket ballistic hopper burning CO and O2. It is useful, therefore, to draw up a list comparing the merit of such as system to the NIMF in performing this mission.

1. Both the NIMF utilizing CO2 propellant as a working fluid and the chemical vehicle burning CO and O2 obtain a specific impulse in the neighborhood of 280-290 seconds. Neither engine is developed technology today, but the physical principles underlying both are well understood, and there is every reason to believe that either could be developed if appropriate amounts of development funds were available. In these respects the NIMF and the chemical vehicle are equals.

2. The NIMF can acquire propellant by compressing it out of the martian atmosphere at an energy cost of about 84 kWe-hrs per ton. The CO/O2 fuel for the chemical vehicle must be

produced by a chemical processing facility on the surface of the planet at energy cost of about 10,000 kWe-hrs per ton. In this respect, then, the NIMF is over a hundred times superior to the chemical vehicle. Indeed, opting for the chemical vehicle might be compared to buying a car which can only use gasoline costing $100.00 per gallon. Actually however, the situation is worse than that, because in this case you also have to buy not only the gas, but the gas station, and the oil company too. That is to say, the chemical vehicle will not be able to operate until there is a manned base with a nuclear reactor and a significant chemical engineering capability. In other words, no long distance exploration will be possible until after the infrastructure is built. Furthermore, even after the infrastructure is built, the production of fuel for the chemical exploratory vehicle will be an overhead on the base power supply that will be in competition with other demands that may frequently shove it aside.

3. The chemical vehicle must be fueled at a base (THE base) while the NIMF can fuel itself. This means that when the chemical vehicle takes off it must carry sufficient fuel for both the outbound and return trips, whereas the NIMF need only carry sufficient propellant for the hop one way. In effect, this difference in operating cuts the real specific impulse of the chemical vehicle in half relative to the NIMF, which in turn severely limits its operating range.

In the table below we give the mass ratios for both a NIMF and a chemical ballistic hopping vehicle, assuming that both use parachute assisted landing leaving a terminal rocket deceleration requirement of 500 m/s. (If parachutes are rejected in favor of pure rocket deceleration, then the NIMF performance degrades to levels somewhat superior to those given in the table for the chemical vehicle, while the chemical vehicle performance degrades to the point where it is completely unusable for hops beyond 300 km.)

Table 3

COMPARISON OF CAPABILITY OF NIMF AND CHEMICAL MARS HOPPERS

| Hop Range (km) | NIMF Mass Ratio | Chemical Veh. Mass Ratio |
|---|---|---|
| 281 | 1.72 | 2.98 |
| 676 | 2.07 | 4.28 |
| 1266 | 2.48 | 6.16 |
| 2240 | 2.98 | 8.86 |
| 3911 | 3.57 | 12.75 |
| 8000 | 4.28 | 18.34 |
| Orbital | 4.61 | 21.21 |

Now that mass ratio of 8.86 given for the chemical vehicle for a 2240 km hop is pretty sporty, it is slightly higher than the mass ratio of a Centaur upper stage vehicle (an aluminum balloon) carrying ZERO payload. A chemical hopper with this mass ratio might be able to exist and carry a tiny payload (because CO is denser than the Centaur's hydrogen fuel) but that is the absolute limit, and for practical purposes we may take the chemical vehicle's effective range at about 1300 km (810 miles). That is hardly global access. The NIMF, on the other hand, can easily reach any point on the planet in a single hop. Thus we see that in this respect the NIMF is infinitely superior to the chemical vehicle in that it can satisfy the essential mission performance requirements, whereas the chemical vehicle cannot.

4. Because of its lower performance, the chemical vehicle will have to be built much lighter and carry much less payload than the NIMF. This means that it will be structurally less safe, carry less scientific instruments, less supplies, and have less endurance for an extended visitation to an exploratory site than the NIMF. It also means that the chemical vehicle is completely incapable of performing any role in global transport of indigenous materials (such as transporting water from the martian polar cap to a base at the equator, or bringing a useful high grade ore from a distant mining site to the base), while the NIMF can do the job.

5. The NIMF carries its own source of electrical power, whereas the chemical vehicle does not. This means that the NIMF can recharge the hydrogen/oxygen fuel cells for electric land roving vehicles used locally by the exploration party, while the chemical vehicle cannot recharge its land rovers. The poverty of electric power faced by a group of explorers utilizing the chemical vehicle my also limit the use of their instruments, and together with their small supply capability, may put them in peril if a minor malfunction should delay their intended return to base.

6. The CO2 carried by the NIMF is a storable monopropellant under martian conditions, while both the CO and the O2 carried by the chemical vehicle are cryogens, and would boil off over time. The boiloff of these cryogenic propellants would itself limit the stay time of the chemical vehicle at an exploratory site. If the boiloff outgassing or other leakage were to occur in any enclosed space (for example that created by an attempt to vacuum jacket the tanks to reduce heat leak or an enclosing fuselage to reduce aerodynamic drag) a flammable (possibly explosive) and toxic mixture would result.

7. When the chemical vehicle returns to base it must land in the immediate vicinity of the fueling station or it will become useless, as there will be no way to haul it overland through the rough martian terrain if it lands a kilometer or two away. As the vehicle must use a parachute to assist in landing, and martian winds can be high, the chemical vehicle's requirement for precision landings may prove difficult or impossible to meet. The NIMF, on the other hand, can land anywhere. If it is only off by a few kilometers the astronauts can walk or return to bases by rover, if it is hundreds of kilometers off, it can just pump itself some more fuel out of the atmosphere and make an additional hop to get home.

8. Highly versatile non-ballistic supersonic winged aircraft configurations are possible for the NIMF. Because of weight limitations, such configurations are not viable for the chemical vehicle. Because the NIMF propellant is the atmosphere itself, in flight propellant acquisition systems are also possible. Such systems are out of the question for a chemical vehicle.

9. Because it refuels after it lands, the NIMF can land empty of fuel. The chemical vehicle, on the other hand, must land filled with enough fuel to return home. This means that it is much heavier than the NIMF when it is landing, putting increased demands on the engineering of its parachute deceleration system. If it hits the ground hard enough to crack its fuel tanks, it may explode.

10. Set against all these advantages for the NIMF is the fact that the NIMF carries a nuclear reactor. However the NIMF reactor carries a radioactive inventory about 6 orders of magnitude less than a power reactor, which will not only be a relief to the Martian EPA, but eliminates the central engineering headache of nuclear reactors, to wit the possibility of meltdown caused by radioactive decay heat if cooling is lost. This small radioactive inventory represents a small hazard compared to that presented by the chemical alternative

to the NIMF, which will be virtually a flying bomb, a lightly built structure filled to the gills with toxic gas and chemical high explosive.

To summarize, if you want to explore, you have to have an exploratory vehicle, a self contained world that is free to roam at will. The great voyages of exploration of the 15th through 19th centuries were only possible because of the long range capability, independence, endurance and versatility of the full rigged sailing ship. If Columbus had had a coal fired steam paddle wheeler he never would have made it to America. The NIMF like the Santa Maria, the Endeavour, and the Beagle before it, derives her motive power from the air about her, and thus must it ever be with true explorers.

## MISSION TO TITAN

Titan, Saturns's largest moon, possesses an abundance of all the elements necessary to support life. It is believed by many scientists that its chemistry may resemble that of the Earth during the period of the origin of life, frozen in time by the slow rate of chemical reactions in a low temperature environment. The abundant prebiotic organic compounds comprising Titan's surface, atmosphere and oceans may one day provide the resource base for extensive human settlement. However, because of its thick cloudy atmosphere, the surface of Titan is not visible from space, and many basic facts about this world remain a mystery. Thus a mission that could bring back samples from various locations on Titan and also perform a low level aerial reconnaissance would be of immense scientific benefit. As we shall see, the NIMF can accomplish such a mission.

Titan's atmosphere is composed of 90% nitrogen, 6% methane, and 4% argon. The atmospheric pressure is 1.5 times that of Earth at sea level, but because of the surface temperature of 100 K the density is 4.5 Earth sea level. The surface gravity is 0.14 that of Earth,.and the wind conditions are believed to be light. However, the great unknown is the Composition of the surface. It may be rock or water ice, it may have methane lakes and rivers, or the entire world may be covered by a methane ocean. The presence of higher hydrocarbons and other organic compounds within the bodies of liquid methane is highly probable, but the precise chemical nature of the mixture is unknown. Hydrocarbon and ammonia ice may also exist. These facts help determine the strategy that the NIMF Titan Explorer (NIFTE) mission will adopt.

The NIFTE mission is initiated by lifting a 8 MT unmanned automated NIMF fueled with 10 MT of liquid hydrogen to LEO. Such a launch can be accomplished by either an STS , a Titan IV, or an upgraded Titan III. The NIMF uses the hydrogen propellant to generate a delta-V of 7.6 km/s, driving itself onto a 4 year trajectory to Titan. Arriving at Titan, the NIMF aerobrakes itself into the atmosphere, with an entry velocity of 6.2 km/s. Of the 8 MT arriving at Titan, 2 MT are scientific payload, 3 MT comprise a 300 MW engine and its shield, and 3 MT are devoted to vehicle structure and machinery. The NIMF Titan Explorer vehicle is depicted in fig. 4.

Because surface conditions are unknown, the NIMF will not land. Rather it will use atmospheric intake of propellant and aerodynamic lift to remain airborne. Such a mission strategy is uniquely appropriate to Titan, as, with its thick atmosphere and light gravity, this world is the aviation paradise of the solar system. In fact, a human being standing on the surface of Titan would be able to fly by strapping wings onto to arms in the manner of Daedalus and Icarus (and this will no doubt be the preferred mode of transportation of the human settlers of Titan). More to the point, a 8 MT NIMF moving at 50 m/s (112 mph) would require a wing area of only 4 square meters to remain aloft. (i.e. no wings at all)!

- **Engine plus Internal Shield** **2 MT** **(300 MWth)**
- **External Shield** **1 MT**
- **Structure and Machinery** **3 MT**
- **Scientific Payload** **2 MT**

- **Earth Departure Propellant** **10 MT Hydrogen**
- **Titan Departure Propellant** **64 MT Methane**

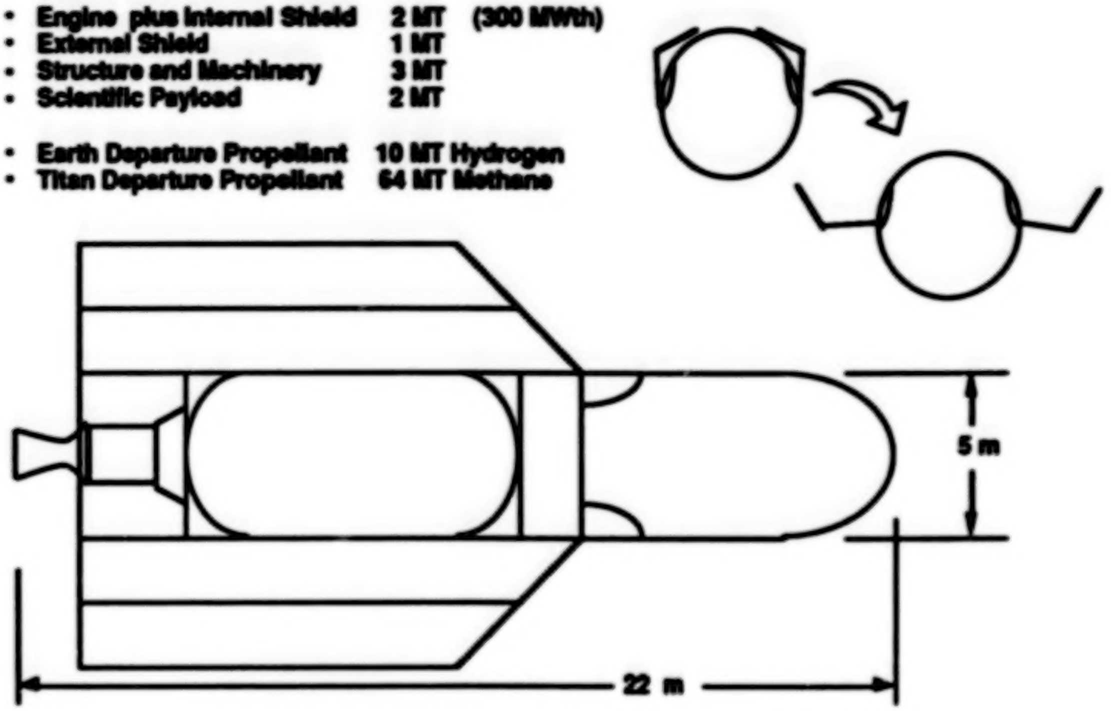

Figure 4 The NIMF Titan Explorer (NIFTE) vehicle

Since fold-out wings of about 100 square meters can be easily accommodated within the payload fairing of the Titan IV, the NIMF Titan Explorer will be able to cruise as slowly as 20 mph, performing a leisurely low-level aerial reconnaissance of the entire satellite.

As it cruises along, the NIMF Titan Explorer will use small electric (battery or RTG) powered aircraft to collect samples from the atmosphere, surface,and submarine regions. These electric aircraft, which we cal TERNs (for Titan Explorers and Retrievers to NIMF) would mass about 10 kg each and could take the form of helicopters, fixed wing tilt rotor seaplanes, dirigibles, or even diving submarines, capable of both aerial flight and subsurface travel in the methane ocean (the low gravity, thick atmosphere, and low density of liquid methane all contribute to making such a vehicle possible). As the NIMF titan Explorer carries a scientific payload of 2 MT, a large variety of TERNs could be carried (fig. 5), anticipating a variety of surface and subsurface conditions, so as to ensure the success of at least several of these probes.

After Titan has been adequately explored and samples collected, the NIMF Titan Explorer will then address itself to investigating the remaining satellites in Saturn's system. Flying in Titan's atmosphere, the vehicle can acquire and liquify methane, which, as we have already noted, is an excellent nuclear thermal rocket propellant, yielding a specific impulse between 560 and 620 seconds. By filling its propellant tank with methane, the NIMF Titan Explorer can provide itself with sufficient propellant to generate a delta-V of 12 km/s. This is sufficient not only for a high energy return to Earth, but also for serial excursions for multiple sample collections from Saturn's other satellites.

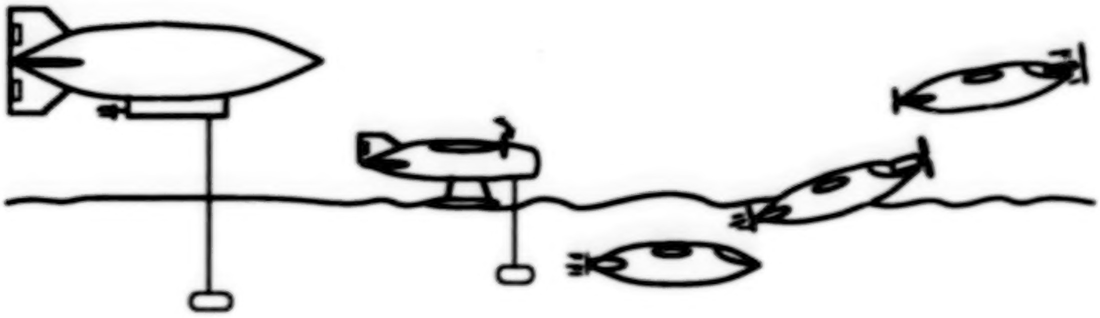

Figure 5 TERN Configurations. TERNs (Titan Explorers & Retrievers to NIMF) are battery or RTG powered miniature aircraft used to collect samples and data from the land, ocean, and submarine regions of Titan.

In Table 4. we show the delta-V's required for excursions from Titan to Saturn's other moons. Each excursion involves landing on the destination moon twice, collecting samples from two locations separated by up to 40 degrees of latitude or longitude, and then returning to aerobrake and refuel at Titan.

Table 4.

NIFTE VISITS TO SATURNS'S OTHER SATELLITES

| Destination | Distance from Saturn (km) | Radius | Delta V Required (km/s) |
|---|---|---|---|
| Mimas | 185,600 | 195 | 13.17 |
| Enceladus | 238,100 | 255 | 11.25 |
| Tethys | 294,700 | 525 | 10.05 |
| Dione | 377,500 | 560 | 8.60 |
| Rhea | 527,200 | 765 | 6.91 |
| Titan | 1,221,600 | 2575 | 0.00 |
| Hyperion | 1,483,000 | 143 | 3.84 |
| Iapetus | 3,560,100 | 720 | 6.90 |
| Phoebe | 12,950,000 | 100 | 8.33 |

It can be seen that with its delta-V of 12 km/s, the NIMF Titan Explorer can collect multiple samples from all of Saturn's moons except Mimas and bring them back to Earth.

## MISSION TO JUPITER

Ganymede, Callisto, and Europa, three of the four Gallilean satellites of Jupiter, are all known to possess large amounts of water ice on their surface. Water can thus be used as a propellant for a NIMF vehicle intending to obtain multiple soil samples from each of these worlds. The NIMF Gallilean Explorer (NIFGE) mission we shall presently describe can accomplish this, and also perform a low altitude orbital reconaisance of all four of the Gallilean satellites (including Io), and perform flyby close inspections of all of the remaining moons of Jupiter.

Because of the absence of an atmosphere on the Gallilean satellites, this mission is in many ways more challenging than the NIFTE mission described above. In this case, a 4 MT NIMF Gallilean Explorer spacecraft (fig. 6) with 11.5 MT of hydrogen propellant and 1 MT of expendable tankage will be lifted to LEO by either an STS, a Titan IV, or an upgraded Titan III. The NIMF will use the hydrogen contained in the expendable tank to generate a delta-V of 6.5 km/s, driving it onto a 2.7 year Hohmann transfer orbit to Jupiter. The NIMF uses oxide pellets coated with ZrC to protect them from hydrogen attack. After the Trans-Jupiter Injection burn, the expendable tank is discarded.

| | | |
|---|---|---|
| **Engine plus Internal Shield** | **1.0 MT** | **(150 MWth)** |
| **External Shield** | **0.5 MT** | |
| **Structure and Machinery** | **1.5 MT** | |
| **Scientific Payload** | **1.0 MT** | |
| | | |
| **Expendable Tank** | **1.0 MT** | |
| | | |
| **Earth Departure Propellant** | **8.3 MT** | **Hydrogen** |
| **Callisto Landing Propellant** | **3.2 MT** | **Hydrogen** |
| **Callisto Departure Propellant** | **45.0 MT** | **Water** |

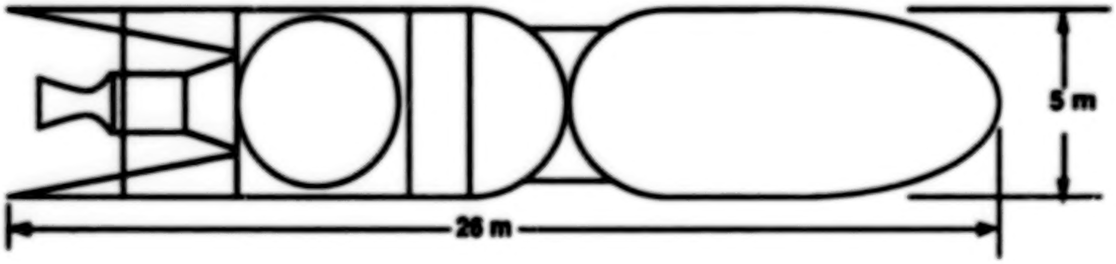

Figure 6 The NIMF Gallilean Explorer Vehicle.

Arriving at Jupiter, the NIMF uses the 3.22 MT of hydrogen contained in its internal tank to generate a delta-V of 5.3 km/s to go into low orbit around and then land on Callisto. The landing spot is chosen from orbit as one near a deposit of ice.

The NIMF then deploys treaded robots to go and collect soil samples. Other robots deploy a long double hose from the NIMF and insert it in an ice deposit. Steam generated by the NIMF reactor is then piped out of of hose to melt a subsurface pocket of ice, while the other hose pipes the resulting water back to fill the NIMF's internal fuel tank.

The internal tank can hold up to 45 MT of water propellant, which provides the NIMF with a delta-V capability of 8.6 km/s. This is sufficient to allow the NIMF to take samples from all over Callisto, to leave Callisto and land on Europa or Ganymede, or to take off Callisto for as medium energy orbit to Earth.

The NIMF Gallilean Explorer carries 1 MT of scientific payload, which is quite sufficient since only land robots are needed. However, during the visit to the ice-world of Europa, one additional instrument whose employment might be of great interest would be a small RTG powered miniature submarine, capable of melting its way through the ice to the water ocean that is believed to exist below. Data on what it finds there can be transmitted back to the NIMF by means of sound.

In Table 5. we show the delta-V's required to take off of either Europa or Callisto and land at a destination satellite of Jupiter.

## Table 5

## JUPITER SYSTEM TRANSPORTATION DELTA-V'S

### A. Departing from Europa

| Destination | Distance from Jupiter (km) | Radius (km) | Delta-V Required (km/S) |
|---|---|---|---|
| Amalthea | 181,000 | 103 | 11.95 |
| Io | 422,000 | 1826 | 5.81 |
| Europa | 671,000 | 1560 | 0.00 |
| Ganymede | 1,071,000 | 2500 | 5.65 |
| Callisto | 1,884,000 | 2450 | 6.88 |

### B. Departing from Callisto

| Destination | Distance from Jupiter (km) | Radius (km) | Delta-V Required (km/S) |
|---|---|---|---|
| Europa | 671,000 | 1560 | 6.88 |
| Ganymede | 1,071,000 | 2500 | 5.86 |
| Callisto | 1,884,000 | 2450 | 0.00 |
| Himalia | 11,480,000 | 85 | 5.07 |
| Elara | 11,740,000 | 40 | 5.08 |
| Lisithea | 11,860,000 | 10 | 5.09 |
| Leda | 11,100,000 | 4 | 5.05 |
| Ananke | 21,200,000 | 10 | 5.24 |
| Carme | 22,600,000 | 10 | 5.24 |
| Pasiphae | 23,500,000 | 10 | 5.25 |
| Sinope | 23,700,000 | 10 | 5.25 |

It can be seen that with its delta-V capability of 8.6 km/s, the NIMF Gallilean Explorer can land on and collect samples from any Jovian moon found to possess ice, except Amalthea.

## EXOTIC MISSIONS MADE POSSIBLE BY NIMF PROPULSION

In addition to its primary purpose as a facilitating technology for manned and large scale unmanned Mars missions, and unmanned sample return missions to the moons of Jupiter and Saturn, the NIMF concept is also an enabling technology for a number of exotic missions whose impossibility without the NIMF has caused them to be largely ignored by mission planners. For example, a winged automated NIMF utilizing atmospheric acquisition of $CO_2$ propellant could accomplish a Venus surface sample return, collecting ground samples and low level aerial reconaisance from every part of the planet before returning to orbit. Water ice exists on Uranus' moons Ariel, Umbriel, Oberon, and Titania, allowing NIMF sample return missions to target these destinations. Neptune's moon Triton could provide a ready source of methane propellant for NIMF exploration of the outer solar system. The asteroid Ceres has ice deposits on its surface, and it is believed that many other asteroids especially in the outermost belt and Trojan regions may also contain large amounts of water ice, thus giving water fueled NIMFs multiple bases from which to carry out the prospecting of the asteroid belt. NIMFs can extract propellant from the icy cores of comets, and could use comets as staging bases for missions to Pluto, the Oort Cloud, and beyond. If equatorial rotation is taken advantage of, the velocity required to attain Saturn orbit is 14.9 km/s, while that for Uranus or Neptune is 12.2 km/s. A winged hydrogen fueled NIMF with an Isp of 950 s could descend into the atmospheres of these planets and collect gas samples (or ground samples, if ground exists) refuel itself out of the hydrogen atmosphere, and reascend to orbit. A pure rocket (i.e. not jet augmented) NIMF would require a mass ratio of about 5.0 to accomplish this Saturn atmosphere return mission, while the mass ratio required for Uranus or Neptune atmosphere return would be about 3.7. It jet intake augmentation is used during thrust, these numbers could be substantially reduced.

## CONCLUSION

We conclude that the NIMF concept offers great potential benefit for human exploration and colonization of the solar system. The NIMF opens up an enormous vista of possibilities, including the ability to launch a manned Mars mission in one launch, and economically sustain a permanent and large scale human presence on Mars. The NIMF vehicle further affords unlimited mobility for exploration not only of Mars, but the asteroid belt, and the satellite systems of the major planets as well. We recommend that the NIMF be made the subject of an in depth study and a substantial research and development effort.

## Acknowledgments

The author wishes to acknowledge many useful discussions of nuclear systems with Carl Leyse and Jack Ramsthaler of the Idaho National Engineering Laboratory.

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N91-22147

# AN APPROACH TO SPACE POWER

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## ABSTRACT

Fusion offers the potential for a very high specific power, providing a large specific impulse that can be traded-off with thrust for mission optimization. Thus fusion is a leading candidate for missions beyond the moon. Here we discuss a new approach for space fusion power, namely Inertial Electrostatic Confinement (IEC). This method offers a high power density in a relatively small, simple device. It appears capable of burning aneutronic fuels which are most desirable for space applications and is well suited for direct conversion. An experimental device to test the concept is described.

## INTRODUCTION

The potential advantages of using a fusion device for space power, both station and propulsion, are well known (See Refs. 1-2 and Fig. 1). These include a very high power density and a relatively "clean" power source. However several concerns have been voiced. The first is the feasibility of developing a suitable fusion device in a near term time frame. The second is the need for a device capable of burning advanced fuels, providing a very low neutron and radioactivity involvement. The work presented here describes an experiment designed as a first step towards answering both issues: namely, an IEC device that can, in principle, be developed rapidly and which can burn fuels like D-$^3$He and p-$^{11}$B.

Recent conceptual studies have concentrated on D-$^3$He fueled devices using magnetic or inertial concepts as candidates for future missions such as Mars or deep space travel. Based upon our review of issues involved, however, IEC emerges as one of the most attractive fusion concepts. This approach offers several significant advantages, including a highly non-Maxwellian plasma which is capable of burning advanced fuels, a relatively simple structure capable of high power density in a relatively small device, and a "natural" coupling to direct energy conversion.

The IEC concept was originally proposed by Farnsworth [3]. Several years after that Hirsch proposed a variation of that design [4]. It is this design of Hirsch's that we base our experiment upon. Figure 2 illustrates our experimental set-up. Electrons are emitted into the vessel from two thoriated tungsten hoops. These electrons then travel about the anode, ionizing the background gas. The created ions will be accelerated towards the cathode where they will form a radial ion current. This current creates a positive space charge in the very center of the IEC device which draws electrons in, creating a very high density central reaction "core" region (Fig. 3). However, energetic charged particle fusion products (MeV protons, alphas) will have sufficient energy to escape the well, allowing coupling to a direct-collector type energy conversion device. This approach, in principle, can lead to stable inner particle reaction rates which are significantly greater than those possible in magnetic confinement and avoid the need for pulsed operation used in inertial confinement.

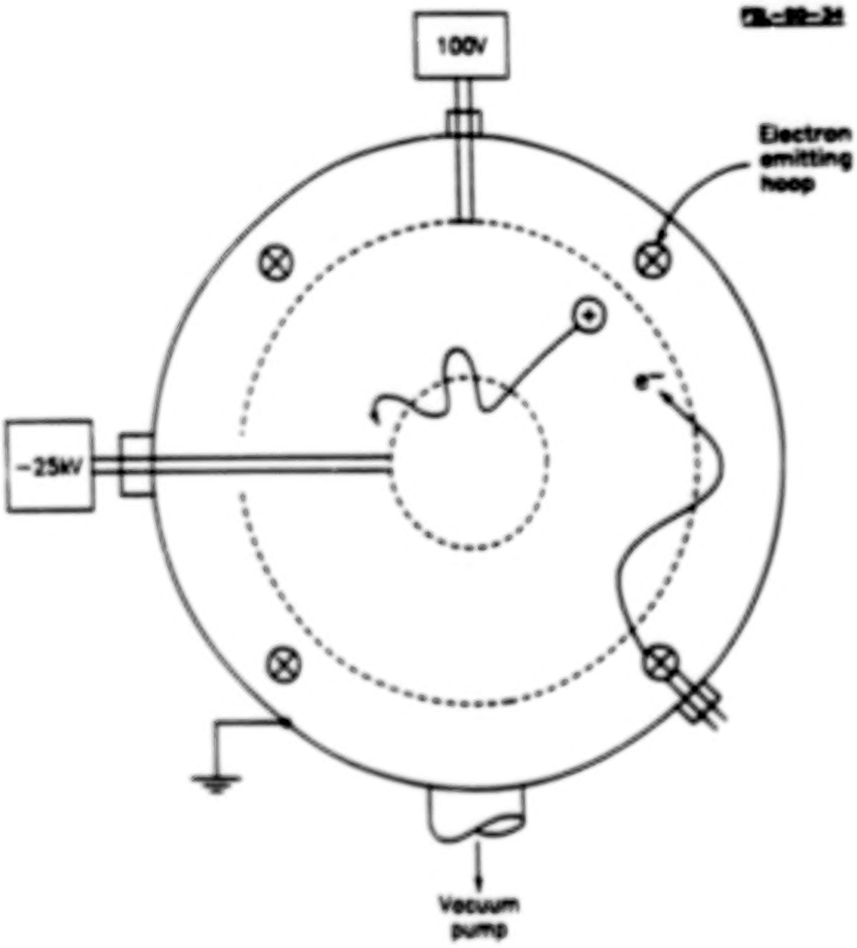
- **VERY HIGH SPECIFIC POWER**
- **VERY HIGH SPECIFIC IMPULSE THAT CAN BE VARIED FOR MISSION OPTIMIZATIONS**
- **HIGHER THRUST THAN ION ENGINES**
- **DEUTERIUM - HELIUM 3: OFFERS BASIC INHERENT SAFETY**
    - **NON TOXIC**
    - **NON HYPERGOLIC/EXPLOSIVE MIXTURE UNDER NORMAL PHYSICAL ENVIRONMENTS**
    - **NON RADIOACTIVE ISOTOPES**
    - **LOWEST NEUTRON FLUX OF REACTIVE ADVANCED FUELS**
- **MISSION ADVANTAGES**
    - **POTENTIAL FOR LONG LIFE**
    - **RESERVICABLE**
    - **ALLOWS CONSIDERATION OF DUAL MODE OPERATION, PROPULSION AND POWER.**

Fig. 1 Advantages of Fusion Energy for Space (From Ref. 1)

The plasma exhaust from the IEC could be used in a direct propulsion concept. However, a very attractive alternative is to couple the IEC to an electrically driven ion propulsion unit. The latter represents a well developed propulsion concept [5,6]. The high specific power offered by the IEC plus the use of direct conversion make its use with an ion thruster unit most attractive.

## PHYSICS OF INERTIAL-ELECTROSTATIC CONFINEMENT

The confinement of plasma in an IEC device has been demonstrated in two different geometries, spherical and cylindrical [7,8]. Spherical geometry offers stronger "convergence", i.e. a higher power density. Thus we are focusing on this geometry. The potential wells, that are needed to confine the plasma, are created by the formation of "virtual" anodes and cathodes, which are in turn created by the accumulation of space charge at the center of the vacuum vessel. This is brought about by injecting ions into a vessel which contains a highly transparent cathode placed concentric with the outer vessel wall. The ion current will travel about the cathode creating a virtual anode, which will then attract electrons. The electrons will travel about this anode in the same manner as the ions, and hence another virtual cathode will be created. In this manner, multiple virtual wells can be created.



- Electrons are emitted from hoop
- 100V biased grid accelerates the electrons
- Ions are created by e⁻ collisions with background gas
- inner HV cathode accelerates ions
- SFID A vessel diameter 30cm
- SFID B vessel diameter 81cm

Fig. 2 Hoop Arrangement for Electron Injection

The wells created in this fashion confine the plasma where the fusion occurs. But there are several ways in which to bring all this about. Mentioned above is ion injection; other methods would include electron injection, with a real anode in addition to the cathode. In this manner, the electrons would travel about the anode ionizing background gas, creating the ions that will then fall into the cathode.

A variety of fusion fuels can be used: D-D, D-T, D-$^3$He, p-$^7$Li and p-$^{11}$B to name a few. The first two involve the use of radioactive tritium and the creation of high energy neutrons. The last three are virtually aneutronic and create minimal problems with radioactivity. Their products are high energy charged particles which are ideal for a direct energy conversion system.

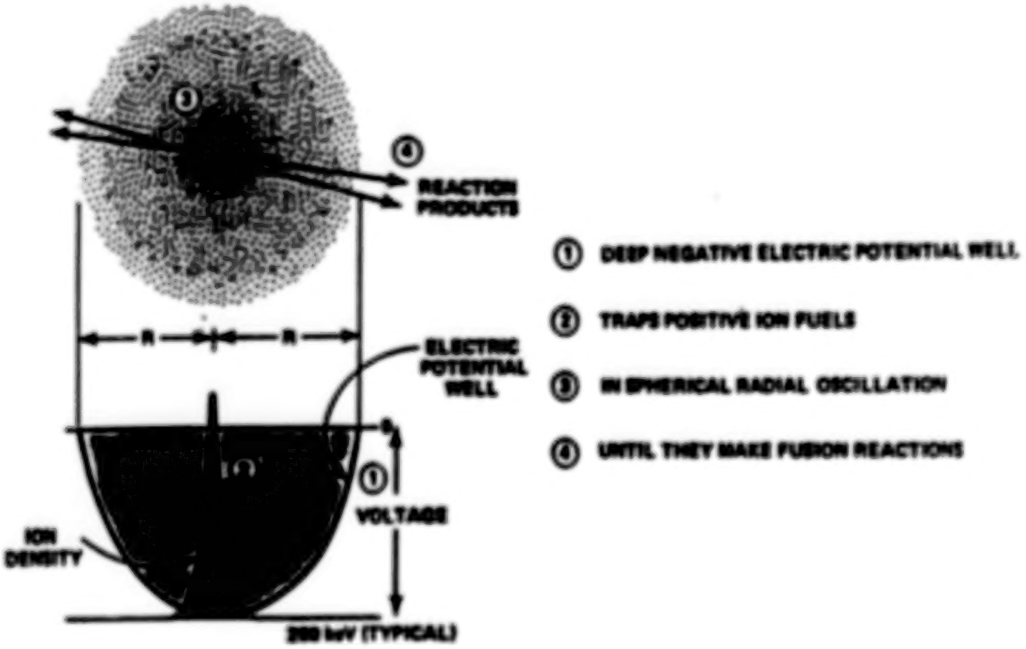

③
④
REACTION PRODUCTS
R
R
ELECTRIC POTENTIAL WELL
①
VOLTAGE
ION DENSITY
200 keV (TYPICAL)
① DEEP NEGATIVE ELECTRIC POTENTIAL WELL
② TRAPS POSITIVE ION FUELS
③ IN SPHERICAL RADIAL OSCILLATION
④ UNTIL THEY MAKE FUSION REACTIONS

Fig. 3 Principles of Operation-1

## U OF I EXPERIMENT

The experiment ongoing at the University of Illinois incorporates two different approaches: electron injection to cause ionization with background gas, and direct ion injection. The former method was outlined earlier; the electrons oscillate about an anode, creating ions which are accelerated by the inner cathode. This set-up is depicted in Fig. 2. Later experiments will employ ion guns and will do away with the electron injection and the outer anode. Two ion guns will be mounted at right angles to each other, with just the cathode remaining in the center of the vessel. The injected ions will start travel about the cathode, trapping negative space charge inside the cathode, and forming a virtual anode. This in turn can lead to the formation of multiple wells, analogous to the skins on an onion.

Two different sized devices will be used (Fig. 4). The smaller device, SFID-A (Spherical Fusion Illinois Device) employs a vessel 29.5 cm in diameter, the larger device, SFID-B, is 61.0 cm in diameter. The cathode and anode in SFID-A will be 7.5 cm and 22.5 cm, respectively. The cathode and anode in SFID-B is 15.2 cm and 45.7 cm, respectively. Both cathodes will be powered by a 100 kV, 25 mA(max) power supply.

A variety of diagnostics will be used. The first to be employed will be a $BF_3$ proportional counter placed inside a neutron moderator outside the vacuum vessels. The expected neutron count with these parameters is expected to be in the $10^7$ neutrons/second region for SFID-B. Additional instrumentation which is presently under development includes a biased probe and an ion beam probe. The biased probe will provide measurements of the plasma density profile during operation of the device with only the anode present. The ion beam probe will be used for potential well measurements. It will be similar to the electron probe employed on an earlier IEC device [9]. However, that device used an anode in the vessel instead of the cathode. Thus, in the present case it is necessary to substitute an ion beam to carry out potential well mapping.

SFID-A (30 cm):

1) Demonstrate operation of a small Hirsch device.
2) Demonstrate scaling parameters.
3) Test the Illinois diagnostics.
4) Measure electrostatic wells.

SFID-B (61 cm):

1) Demonstrate scaling to a large device.
2) Refine diagnostic analysis of center core physics.
3) Study of center core physics.
4) Preliminary study of ion gun injection.
5) Study effects of asymmetries of ion source.

Fig. 4 Physics Objectives

## POWER GENERATION

An IEC fusion device can, in principle, be operated using advanced fuels [10] that produce little or no neutrons during the fusion process. The basic advantage that IEC offers in this respect is that the interacting ions are not thermalized, i.e., remain highly non-maxwellian (Fig. 5). This results in a more beam-beam type reaction which is most favorable for burning advanced fuels with their higher temperature requirements. Three reactions of this type that appear most appropriate for this device are D-$^{3}$He, p-$^{7}$Li, and p-$^{11}$B. These reactions proceed as:

$$\begin{aligned}
&D + {}^{3}He \ \text{-->}\ {}^{4}He\ (3.6\ \text{MeV}) + p\ (14.7\text{MeV})\\
&p + {}^{11}B \ \text{-->}\ 3\ {}^{4}He\ (8.7\ \text{MeV})\\
&p + {}^{7}Li \ \text{-->}\ 2\ {}^{4}He\ (17.5\ \text{MeV})
\end{aligned}$$

Since the products of these reactions consist entirely of charged particles, the energy from the reaction can be extracted by making the fusion products climb a potential barrier [10,11]. If the core of the fusion device is surrounded by a spherical ion collector at high potential, the energy of the radially escaping ions will be converted directly to electrical current at the high voltage. For example, for the p-$^{11}$B reaction, each reaction product carries ~2.9 MeV of energy and the particles are doubly charged. This implies that an ion collector maintained at ~2.9 MV would convert nearly all of the energy of the escaping ions into electricity.

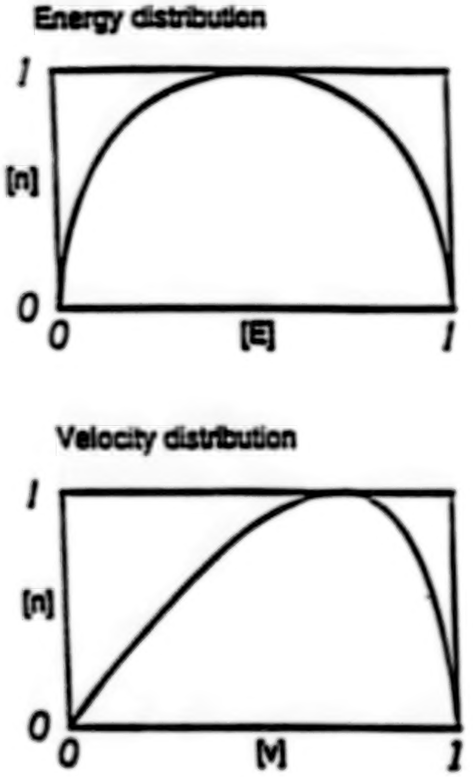


Fig. 5. Energy and velocity distributions for central collisions in radially-nonenergetic, converging ion flow. Center-of-mass and center-of-geometry are coincident in the system.

To illustrate the potential for such operation, results from preliminary calculations are presented in Tables I and II. In Table I we see that a well depth as large as -200 kV can be obtained in a 200-cm sphere with only a modest imbalance in the electron density ($\delta n^-$) in the plasma core. The well depth should be set to match the energy at the peak in the cross section (in the c-m system) for the fusion fuel employed. This means a well of only 40 kV is required for DT, 160 kV for D-$^3$He, and 560 kV for p-$^{11}$B (Table II). Thus, in principle, the 200-cm sphere in the earlier example could burn D-$^3$He with, as shown in Table II, an attractive effective energy gain of over 100.

Such a power generation system has several advantages for space power. It is relatively lightweight, compact, and creates virtually no radioactivity or radiation. It requires only small quantities of fuel, and since it has no moving parts it would be vibration free and durable.

## PROPULSION

Two space propulsion systems based on the inertial-electrostatic fusion systems will be mentioned here. The first is very straightforward: the electric power produced by the direct energy convertor described above would be used to power ion thrusters. Such a system would be simple, rugged, and would have a very high specific impulse. The ion thruster that this would be connected to "has evolved to the point of flight readiness..." [6].

## Table I

## NEGATIVE ELECTRIC POTENTIAL WELLS WITH UNIFORM CHARGE DENSITY

| Well Depth Ew (keV) | Sphere Radius R (cm) | Charge Density $\delta n^-$ (1/cm+3 ) | Charge Neutrality Deviation, for no-1E15/cm$^3$ |
|---|---|---|---|
| 200 | 200 | 1.66E7 | 1.66E-8 |
| 100 | 100 | 3.32E7 | 3.32E-8 |
| 100 | 31.6 | 3.32E* | 3.32E-7 |
| 10 | 3.16 | 3.32E9 | 3.32E-6 |

$\delta n^-$: differential electron density required to create well
no: central ion density

## Table II

## REACTION ENERGY AND EFFECTIVE ENERGY GAIN AT PEAK CROSS-SECTION FOR FUSION FUELS

| Fusion Fuels | Fusion Energy, Released (MeV) | Peak Cross Section $\sigma$ (b) | Energy at Peak | Gain at Peak |
|---|---|---|---|---|
| DT | 17.6 | 5 | 40 | 440 |
| D-$^3$He | 18.3 | 0.7 | 160 | 114 |
| p-$^6$Li | 4.0 | 0.2 | 1250 | 3.2 |
| p-$^{11}$B | 8.7 | 0.8 | 560 | 15.5 |

The second system is one that was proposed by R. W. Bussard [12]. This concept is a high thrust, high specific impulse system referred to as QED. Here, high voltages from the IEC accelerate a relativistic electron beam (REB) which would be directed into a magnetically confined plasma. The REB couples strongly with the plasma, heating it to high temperatures. The plasma is allowed to exhaust through a magnetically insulated nozzle providing thrust while fresh propellant is continuously injected into the system radially. R. W. Bussard [12] calculates that such a system could provide a specific impulse as high as 3000 seconds, with accelerations as high as 0.5 g.

## FUEL REQUIREMENTS AND LUNAR HELIUM-3

The D-$^3$He reaction represents a very attractive approach to fusion for space applications by combining relative ease of operation (e.g., relatively modest plasma temperature and confinement time requirements) with nearly aneutronic operation (neutrons from D-D reactions are reduced in the present beam-beam approach compared to a Maxwellian D-$^3$He plasma). The main difficulty with this approach is the lack of natural sources of $^3$He on the earth. While $^3$He can be bred using accelerator-like techniques [13], the other route which is well suited to space applications is lunar mining [13,14]. Apollo lunar samples indicate that ~$10^9$ kg could be obtained from the lunar surface which has been impregnated with $^3$He and other gases by long bombardment of the solar wind [14]. This amount of $^3$He could fuel fusion plants yielding over $10^7$ Gw-yr, representing a sufficient resource for both terrestrial applications and an active space program. Subsequent needs could eventually move on to extract $^3$He from other sources (such as the $10^{23}$ kg estimated on Jupiter).

In conclusion, it should be stressed that lunar $^3$He is not absolutely essential for the development of advanced fuel fusion power sources. Indeed $^3$He can be bred; or p-$^{11}$B, though more difficult to burn, could be developed. Still, in terms of a power source for space, lunar $^3$He appears to fit in so well that this approach deserves serious study.

## SUMMARY

Fusion offers a most attractive way to power deep space travel. IEC offers a different approach to fusion confinement which combines a high power density with the ability to burn aneutronic fuels and employ direct energy conversion. An experiment to explore this approach has been described. While considerable R & D would be required to scale this experiment up to the power levels required for space applications, the small size and relative simplicity of IEC imply that such development could be done rapidly compared to other fusion devices.

## ACKNOWLEDGEMENTS

This work is supported by DARPA under contract DTI-S394-006. Helpful discussions with R. W. Bussard are gratefully acknowledged.

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# THE NASA-LEWIS PROGRAM ON FUSION ENERGY FOR SPACE POWER AND PROPULSION, 1958-1978*

N91-22148

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## ABSTRACT

This paper will provide an historical synopsis of the NASA-Lewis research program on fusion energy for space power and propulsion systems. This research program in fusion energy was initiated at the NASA Lewis Research Center in 1958 to explore the potential applications of fusion energy to space power and propulsion systems. This program involved several hundred man-years of effort, and included all aspects of fusion research such as theory, experiment, technology development, and advanced mission analysis. It ended in 1978. This effort was carried out within the NASA-Lewis Electromagnetic Propulsion Division, where the mission analysis and basic research on high temperature plasma physics was done by the Advanced Concepts Branch, and technology development, including the development of three pioneering high-field superconducting magnetic confinement facilities, was conducted in the Magnetics and Cryophysics Branch.

Some of the fusion-related accomplishments and program areas covered by the NASA-Lewis effort during this period include: basic research on the Electric Field Bumpy Torus (EFBT) magnetoelectric fusion containment concept, including identification of its radial transport mechanism and confinement time scaling; operation of the Pilot Rig mirror machine, the first superconducting magnet facility to be used (in 1964) in plasma physics or fusion research; operation of the Superconducting Bumpy Torus magnet facility in 1972, the first such facility used in fusion research to generate a toroidal magnetic field; steady-state production of neutrons from DD reactions, starting in 1967; studies of the direct conversion of plasma enthalpy to thrust by a direct fusion rocket via propellant addition and magnetic nozzles; power and propulsion system studies, including D$^3$He power balance, neutron shielding, and refrigeration requirements; development of large volume, high field superconducting and cryogenic magnet technology; studies of the direct conversion of plasma enthalpy to electrical power; advancing the state of the art in cryogenic and liquid helium technology; development of ferrofluids and studies of ferrofluidics technology; ion cyclotron resonance heating of plasmas at high power and in the steady state; development of the Hipp coil, a class of ICRH antenna which is now universally used in tokamak RF heating applications; development of heat pipe technology; development of the bundle divertor as the basis of a direct fusion rocket using a toroidal plasma; advancing the state of the art in plasma diagnostics, including fluctuation-induced transport, heavy ion beam probes, and Thomson scattering; and mission analyses and systems studies of fusion propulsion systems for interplanetary missions. Many of these accomplishments resulted in patents, were the first of their kind, and have been incorporated in the world fusion program.

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*A full-length version of this paper, containing an extensive bibliography of NASA-Lewis Publications, will appear in Fusion Technology in Fall, 1990.

- **Duration of NASA Fusion Effort: 1958-1978**
- **NASA Centers Involved**

  1. **Lewis Research Center (98%)**
  2. **JPL (2% - mission studies only**

- **Other Organizations Active in the 1960's**

  1. **AFOSR - D. George Samaras**
  2. **Aerojet-General (San Ramon) - John Luce**

- **Individuals Supervising NASA-Lewis Fusion Effort**

  1. **Gerald V. Brown**
  2. **Gerald W. Englert**
  3. **John Evvard**
  4. **Wolfgang (Wolf) E. Moeckel**
  5. **Warren D. Rayle**
  6. **John J. Reinmann**
  7. **Eli Reshotko**
  8. **J. Reece Roth**
  9. **Abe Silverstein**

- **Contractors to NASA-Lewis Fusion Space Power and Propulsion effort**

  1. **Prof. Edward J. Powers, University of Texas, Austin**
  2. **Prof. Andrew L. Gardner, Brigham Young University**
  3. **Profs. M. Kristiansen and M. O. Hagler, Texas Tech University**
  4. **Prof. George H. Miley, University of Illinois**
  5. **Prof. Robert Hickok, R.P.I.**
  6. **Prof. Benjamin Lax, M.I.T.**

- **Areas Within the NASA-Lewis Fusion Space Power and Propulsion Effort**
  - **Direct Relevance**
    1. **Mission Studies**
    2. **Basic Research in High Temperature Plasma Physics**
    3. **Engineering Development of Superconducting Magnet Facilities for Plasma Research**
  - **Collateral Relevance**
    1. **Superconducting and Cryogenic Magnet Technology**
    2. **Direct Convertor Technology**
    3. **Cryogenic/Liquid Helium Technology**
    4. **Ferrofluidic Technology**
    5. **Ion Cyclotron Resonance Heating Technology**
    6. **Heat Pipe Technology**
    7. **Space Nuclear Power Systems Testing Facility at Plum Brook, Ohio**
    8. **Plasma Diagnostic Development**

## Assumptions Underlying NASA-Lewis Fusion Space Power and Propulsion Effort

1. **DT reactors out of question because of shielding requirements**

2. **Program focussed on propulsion rather than power supply**

3. **Direct fusion rockets preferred**

4. **Fusion reactions with ~ 100% energy released in charged particles most desirable**

5. **High beta confinement concepts required**

6. **Steady-state operation required**

7. **Power source on board, not remote to spacecraft**

## Screening Criteria Used at NASA-Lewis for Fusion Fuels and Confinement Concepts

1. **Will it make a good fusion power or propulsion system?**
2. **Is it technologically feasible?**
3. **Is the physics of the concept known?**

- **If the answer to #3 was no, then basic research was initiated on otherwise promising concepts.**

- **This approach lead to basic research in the following areas:**
  1. **Magnetoelectric confinement**
  2. **Steady-state ICRH for start-up**
  3. **Design studies of advanced fuel reactors**

## PERSONNEL ASSOCIATED WITH NASA-LEWIS BUMPY TORUS PROJECT

### NASA Personnel

| Physics Investigations | Superconducting Magnet Facility |
|---|---|
| J. Reece Roth, P. I. | Willard Coles |
| Walter M. Krawczonek | A. David Holmes |
| | Thomas A. Keller |

### External Collaborators

**Andrew L. Gardner, B.Y.U**

**Edward J. Powers, U. of Texas**
**Young Kim, U. of Texas**

### NASA-NRC Postdoctoral Associates

**Hans Persson**
**George X. Kambic**
**Richard W. Richardson**
**Glenn A. Gerdin**
**Chandra M. Singh**
**Chitra Sen**
**Raghuveer Mallavarpu**

## Bumpy Torus Facility Data to Final Shutdown at Lewis

| | |
|---|---|
| **Days of Operation with Coils Charged** | **436** |
| **Working Days Since First Plasma** | **1337** |
| **Utilization Fraction** | **33%** |
| **Total Hours Experimental Operation** | **2620 Hours** |
| **Number of Coil Normalicies** | **189** |

## RESULTS OF NASA LEWIS EFBT RESEARCH

1. **Magnet facility operated satisfactorily until final shut-down**
2. **$D^+$ Ion kinetic temperatures up to 2500 eV, 3500 eV in He.**
3. **Up to 45% of power into plasma appears in ion population**
4. **Electric field has major effect on plasma density and confinement time-opposite polarities make a factor of 20 difference**
5. **Fluctuation-induced transport identified as the dominant radial transport mechanism**
6. **Geometric mean plasma emission discovered experimentally**
7. **Scaling laws of ion kinetic temperatures, densities, and containment times with B, T, $V_A$ are favorable**
8. **No asymptotic limits on performance**

- **Accomplishments of the NASA-Lewis Fusion Space Power and Propulsion Effort:**

  1. **World's first superconducting magnet facility for plasma physics and fusion research (1964)**
  2. **World's first superconducting magnet facility generating a toroidal magnetic field (1972)**
  3. **Steady-state neutron production from DD reactions in an electric field dominated modified Penning discharge (1967)**
  4. **Steady-state operation of the NASA-Lewis Electric Field Bumpy Torus (EFBT) at plasma parameters ($T_i$, $N_e$, $\tau_p$) never equaled or exceeded by the EFBT experiment at ORNL**
  5. **Steady-state DD neutrons from the electric field bumpy torus**
  6. **Very encouraging mission studies vis-a-vis fission-electric and mission solar-electric systems**
  7. **Development of steady-state ICRH technology**
  8. **Development and demonstration of short, compact ICRH antenna (HIPP coil)**
  9. **Discovery of continuity-equation plasma oscillation**
  10. **Demonstration of fluctuation-induced transport as the dominant radial transport mechanism in EFBT plasma**
  11. **Demonstration of radially-inward fluctuation-induced ion transport in EFBT Plasma**
  12. **Operation of SUMMA magnet facility, 50 cm-Bore, 8.0-5.0-8.0 Tesla mirror configuration.**

- **Low Visibility of NASA-Lewis Fusion Power and Propulsion Effort**

  1. **Mostly bootlegged and funded within NASA Lewis**

  2. **Subject too long-term for many tastes**

  3. **Subject too "far out" for public discussion in the midwest**

  4. **Desire to avoid competition within NASA-Lewis, with the Space Nuclear Auxiliary Propulsion (SNAP) Office**

- **Results of our Magnet Technology Engineering Development and Basic Research on High Temperature Plasmas was Freely and Extensively Published as NASA Reports and in the Archival Literature**

- **Reasons for Termination of NASA-Lewis Fusion Space Power and Propulsion Effort**

  1. **Retirement of administrators with vision**
  2. **Lack of headquarters support**
  3. **Ascendency of accounting over technical management**
  4. **Budget pressures generated by the Shuttle program**
  5. **Termination of the SNAP nuclear-electric program for lack of a mission**
  6. **No approved mission for space fusion systems**
  7. **Long term nature of goal**

N91-22149

# SAFETY AND ENVIRONMENTAL CONSTRAINTS ON SPACE APPLICATIONS OF FUSION ENERGY

by

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## Abstract

This paper examines some of the constraints on fusion reactions, plasma confinement systems, and fusion reactors that are intended for such space-related missions as manned or unmanned operations in near earth orbit, interplanetary missions, or requirements of the SDI program. Of the many constraints on space power and propulsion systems, those arising from safety and environmental considerations are emphasized in this paper since these considerations place severe constraints on some fusion systems and have not been adequately treated in previous studies.

## Introduction

It is very probable that only nuclear fission or nuclear fusion energy will be capable of satisfying space-related requirements for more than a few hundred kilowatts of steady state electrical power. Early work at the NASA Lewis Research Center on fusion propulsion systems between 1958 and 1978 [refs. 1 and 2] examined the application of steady state fusion reactors, generating several hundred megawatts of thermal power, to direct fusion rockets for manned interplanetary missions. More recent design studies of fission and fusion space electrical power systems [refs. 3 to 6] have addressed the long-term needs of the strategic defense initiative (SDI) program, for which a requirement of 1 to 10 megawatts of steady state electrical power is anticipated, with a further possible requirement of up to several hundred megawatts of "burst" electrical power for periods of hours. A preliminary report on advanced fusion power for space applications of interest to the Department of Defense has recently been published by the National Academy of Sciences, under the sponsorship of the Air Force Studies Board [ref. 7]. The data in this report indicate that fusion power and propulsion systems may have a lower specific mass (kilograms per kilowatt of electrical power) than that anticipated for fission-electric systems.

Of the many constraints on fusion space power and propulsion systems, those arising from safety and environmental considerations will be emphasized in this paper, since these considerations place servere constraints on some fusion systems, and they have not been adequately treated in previous studies. This paper first discusses the safety and environmental factors which affect the selection of fusionable fuels, followed by a consideration of factors which affect the choice of confinement concept. Finally, some conclusions are drawn, on the basis of safety and environmental considerations, about the choice of magnetic vs inertial fusion; the best apparent choice of fusion reaction; and the constraints which magnetic containment concepts must satisfy for space applications.

## Safety and Environmental Factors Affecting Fusion Fuel Selection

### Tritium in Space

Whatever the merits of the DT reaction for electric utility powerplant reactors [ref. 8], the constraints of the space environment, to be discussed below, make it desirable to consider other fusion reactions. Utilization of the DT, DD, or catalyzed DD reactions would necessitate the use of radiologically significant amounts of tritium in space, and one must ask whether the risks of doing so can be reduced to acceptable levels.

A benchmark for large-scale radiological accidents was established by the Chernobyl nuclear accident of April, 1986 [refs. 9, 10]. The approximate radioactive source terms and inventories associated with this accident are listed in the first column of Table I. As a consequence of this accident, approximately 50 megacuries of biologically inert noble gases, mostly krypton and xenon, were released into the atmosphere. An additional 50 megacuries of biologically active fission products were released and spread over a large portion of the Eurasian continent. It is instructive to compare these inventories with the radioactive inventories associated with the Starfire DT tokamak reactor [ref. 11], a gigawatt level powerplant fusion reactor. The Starfire reactor had a total tritium inventory of 11.6 kilograms, which is approximately 110 megacuries of volatile radioactive material. This is more than twice as great as the biologically significant (non noble gas) radioactivity released during the Chernobyl accident.

TABLE I

**Fission and DT Fusion Radiological Hazard Comparison**

**1 GWE Operation for One Year**

| Reactor Characteristics | Chernobyl Accident | Typical LWR | Starfire DT Tokamak |
|---|---|---|---|
| Biologically Inert (noble) Gas Release, (MC$_i$) | ~50 | ---- | ---- |
| Biologically Active Radiation Release, (MC$_i$) | ~50 | ---- | ---- |
| Tritium Inventory (MC$_i$) | ---- | ---- | 111 |
| Nonvolatile Core/Blanket Inventory (MC$_i$) | 1500 | ---- | 6140 |
| Annual Radioactive Waste Production, Tonnes/Year | ---- | 30-60 | 69 |

By comparing inventories and source terms in Curies, it is intended to provide an indication of relative public acceptability, public perception of relative risk, and relative immediate consequences of an accident. Such immediate consequences include exposure of operating staff and emergency crews, the necessity of evacuating large areas, and other emergency measures taken for public safety. Long-term consequences such as genetic or somatic damage to individuals or populations would require, in addition, consideration of the relative biological effectiveness (RBE) of each species released, along with its environmental pathway and source term. While the relative biological effectiveness may be useful for assessing the long-term consequences of a particular accident, it probably has little impact on the social acceptance of a nuclear technology prior to its introduction.

If a DT reactor were used in space, the penalties associated with a lithium breeding blanket for the tritium would probably be so great that tritium fuel would be supplied from ground-based sources. In a direct fusion rocket, or in a fusion-electric system based on a direct converter, it will be very difficult to recover the unburned tritium, and it is therefore prudent to assume that all unburned tritium is lost to space and unavailable for reinjection into the reactor. Since the burnup fraction [ref. 12] of tritium will likely be in the range from 5 to 30% for DT reactors, much more tritium will be required to fuel a reactor than is actually burned to produce electrical power or propulsion. Depending on power level, from 10 MWT to 1GWT, a space power or propulsion system might use from approximately 0.03 kilogram to 3 kilograms of tritium per day. Safety considerations make it necessary to be concerned both about lifting this tritium fuel into orbit in the first place, and then assuring that it does not re-enter the atmosphere.

The particles trapped in the earth's magnetosphere are supplied by the solar wind, and consist mostly of hydrogen with a small admixture of helium and other elements. If all of the hydrogen trapped in the magnetosphere were liquified, the liquid hydrogen would approximately fill an olympic-sized swimming pool. Thus, the total amount of matter in the magnetosphere is not very great, and one must be seriously concerned about the effects of adding charged particles or significant amounts of additional matter to that already present in the magnetosphere. The amount of matter in the magnetosphere is comparable to the propellent exhausted by many propulsion systems as they move through it. Since most particles trapped in the magnetosphere eventually find their way into the earth's atmosphere, one must be concerned about the possible effects of injecting tritium ions in the magnetosphere, which are later precipitated into the atmosphere by MHD instabilities.

If tritium or any other radioactive nuclear fuel is used in space, one must address the following accident scenarios: a) a Challenger-type accident in which the space shuttle ferrying the tritium into orbit blows up in the atmosphere and releases the tritium inventory; b) re-entry of the fuel inventory into the atmosphere as a result of atmospheric drag or an unintended change in the orbital elements of a spacecraft with a fusion reactor on board; or c) leakage of unburned fusionable fuel into the atmosphere by such routes as trapping of its ions in the magnetosphere, followed by auroral precipitation in the earth's atmosphere.

## Neutron Fluxes in Space

Probably the single most important factor in optimizing a space power or propulsion system is to minimize the initial mass that must be placed in earth orbit, since transportation into orbit is very expensive. In consequence, there is a strong temptation to omit shielding to the maximum extent possible, in order to conserve mass. The 14 MeV neutrons produced by the DT reaction require at least a meter of shielding to be slowed down. A spectrum of blanket designs is possible, ranging from full shielding to a bare reactor. In very unusual circumstances, a partially or fully shielded neutronic reactor might be lighter than a bare aneutronic reactor, but this appears unlikely in view of the mass penalty of radiators and energy handling equipment required to deal with thermal energy deposited in the shield. Here, we assess the environmental consequences of the limiting case, a bare reactor.

It has not always been realized that unshielded fluxes of neutrons, charged particles, and x-rays can pose a serious environmental hazard over surprisingly large distances from an unshielded source. Let us consider a bare, unshielded fusion reactor, and focus on the consequences of an unshielded flux of neutrons from such a reactor. The flux of neutrons, $\phi_L$, from a point source of S neutrons per second at a distance $R_L$ is given by

$$\phi_L = \frac{S}{4\pi R_L^2} \text{ neutrons/m}^2 - \text{sec.} \qquad (1)$$

The relationship between the power lost in the form of neutrons, $P_N$, the total fusion power, $p_F$, and the fraction of the power in neutrons, $f_N$, is given by

$$P_N = f_N P_F, \qquad (2)$$

while the power in charged particles, $P_C$, is given in terms of the total fusion power produced by

$$P_C = (1 - f_N) P_F. \qquad (3)$$

Combining Equations 2 and 3, the relation between the neutron power, the power in charged particles, and the fraction of the power in the form of neutrons is given by

$$P_N = \frac{f_N P_C}{1 - f_N} . \tag{4}$$

The source term from an unshielded fusion reactor generating neutrons is given in terms of the total neutron power, $P_N$, in megawatts, the electronic charge, $e = 1.60 \times 10^{-19}$ Coulomb, and the energy $E_N$ of the individual neutron in MeV as follows,

$$S = \frac{P_N(\mathrm{MW})}{e\,E_N(\mathrm{MeV})} = \frac{f_N P_c(\mathrm{MW})}{(1 - f_N) e E_N(\mathrm{MeV})} \text{ neutrons/sec,} \tag{5}$$

where Equation 4 has been substituted for the neutron power in Equation 5. Substituting Equation 5 into Equation 1, and solving for the standoff distance $R_L$, one obtains

$$R_L = \left| \frac{f_N P_c(\mathrm{MW})}{4\pi e\,\phi_L (1 - f_N) E_N(\mathrm{MeV})} \right|^{1/2} \text{meters} \tag{6}$$

It is of interest to calculate the standoff distance based on radiological safety considerations for a typical fusion propulsion system. The design studies of references 1 and 2 indicate that a propulsion system utilizing a direct fusion rocket might require about 200 megawatts of charged particle power in the exhaust jet. For pure DT fusion, the fraction of the energy released in the form of neutrons is $f_N = 0.80$, and the neutron energy $E_N$ is 14.1 MeV. The occupationally acceptable safe dose for 40 hour per week exposure to MeV neutrons is approximately 10 neutrons per square centimeter per second, or $10^5$ neutrons per square meter per second. Since such a fusion propulsion system would start out orbiting the earth, the natural unit of length used to measure the standoff distance is the earth radius, $R = 6378$ kilometers.

The neutron fraction, neutron energy, and safe standoff distance under the above assumptions is shown for an unshielded fusion reactor with isotropic neutron production in Table II for 5 fusion reactions. The safe distance, beyond which the neutron fluxes are below the occupational standard, are also given in earth radii. Clearly, an unshielded fusion reactor will generate such a large neutron flux that no unshielded person or thing can approach safely within a very large distance of it. If one were to use a more demanding standard, like 10% of the background radiation level, for example, these standoff distances would be still larger. It should be noted that the same consideration applies to both unshielded magnetic and inertial fusion reactors, since the average of 200 megawatts for an unshielded propulsion system is based on the average power required for interplanetary missions.

## TABLE II

## SAFE DISTANCE FROM UNSHIELDED FUSION REACTOR WITH ISOTROPIC NEUTRON PRODUCTION

| REACTION | NEUTRON FRACTION, $f_n$ | NEUTRON ENERGY, $E_n$ MeV | SAFE DISTANCE, R,KM | $R/R_0$ |
|---|---|---|---|---|
| DT | 0.80 | 14.07 | 16,800 | 2.6 |
| DD | 0.336 | 2.45 | 14,300 | 2.2 |
| CAT DD | 0.38 | 8.26 | 8,600 | 1.4 |
| $D^3He$ | 0.02 | 2.45 | 2,900 | 0.45 |
| $p^6Li$ | 0.05 | 1.75 | 5,500 | 0.86 |

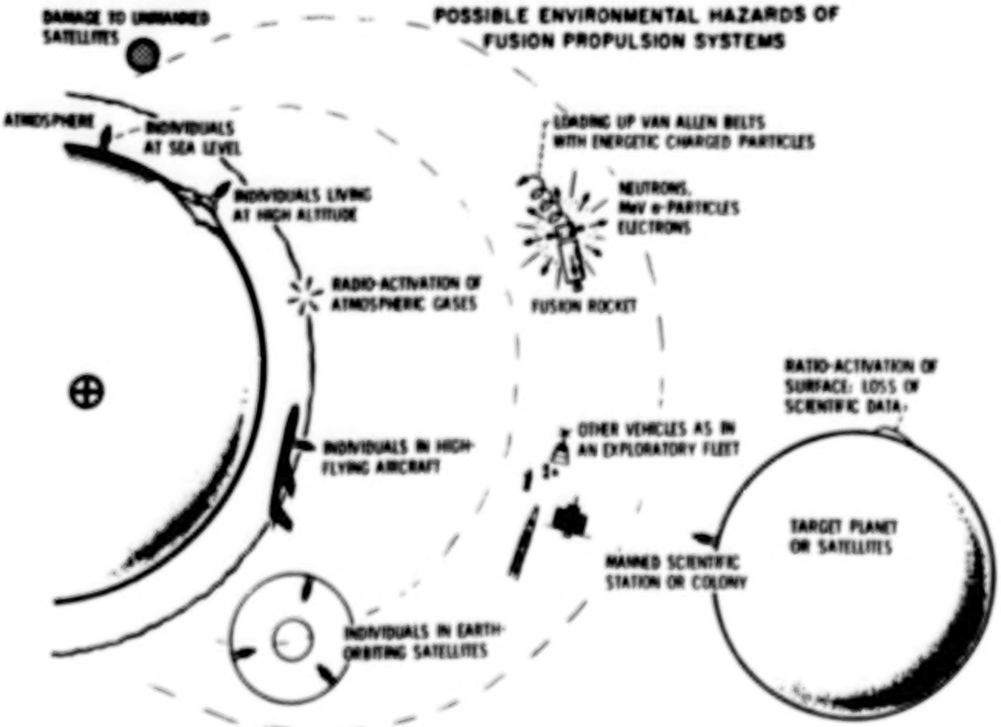
Assumptions:

a) 200 MW of charged particle power
b) Radiologically safe dose for continuous exposure to MeV neutrons: 10 neutrons/cm$^2$-sec
c) Earth radius $R_0$ = 6378 km

Some of the environmental hazards posed by an unshielded fusion reactor in space are summarized in Figure 1. These hazards could include the effects of neutrons, energetic reaction products, unburned fuel ions, electrons, and X-ray radiation. At least some of these hazards would result from the use of a bare reactor, regardless of the fusion reaction used. These forms of radiation could load up the magnetosphere with energetic charged particles; cause damage to unmanned satellites in earth orbit; damage other spacecraft, as in an exploratory fleet or a space shuttle; it could affect individuals in earth-orbiting satellites or in high flying aircraft; the radiation could activate atmospheric gases by direct interaction; and at optical frequencies, radiation could affect the work of individuals such as astronomers at high altitudes or even at sea level.

The detrimental environmental effects of an unshielded fusion reactor are not limited to low earth orbit. If such an unshielded fusion reactor or propulsion system were to approach another planet or satellite, the unshielded radiation could lead to activation of the surface and loss of scientific data. It could affect a manned scientific station or colony. Unshielded radiation on a target planet or satellite surface could alter or erase the cumulative effect of eons of integrated information from the solar wind, cosmic rays, or other long-term surface interaction processes.

The above considerations make it clear that neutrons should be shielded and not allowed to escape directly into space. If this is so, then their thermal energy must be disposed of. In space, the only way in which an isolated spacecraft can dispose of waste heat is by radiation, and the necessary radiators then represent a significant mass penalty which must be paid to accommodate the presence of neutrons. These considerations suggest strongly that the best way of avoiding what must be either a safety hazard or a mass penalty is to use fusion reactions which generate the minimum possible amount of neutron, radiant, or thermal energy.


POSSIBLE ENVIRONMENTAL HAZARDS OF FUSION PROPULSION SYSTEMS
DAMAGE TO UNMANNED SATELLITES
ATMOSPHERE
INDIVIDUALS AT SEA LEVEL
INDIVIDUALS LIVING AT HIGH ALTITUDE
RADIO-ACTIVATION OF ATMOSPHERIC GASES
INDIVIDUALS IN HIGH-FLYING AIRCRAFT
INDIVIDUALS IN EARTH-ORBITING SATELLITES
LOADING UP VAN ALLEN BELTS WITH ENERGETIC CHARGED PARTICLES
NEUTRONS, MeV α-PARTICLES ELECTRONS
FUSION ROCKET
OTHER VEHICLES AS IN AN EXPLORATORY FLEET
MANNED SCIENTIFIC STATION OR COLONY
RATIO-ACTIVATION OF SURFACE; LOSS OF SCIENTIFIC DATA;
TARGET PLANET OR SATELLITES

## Neutronic Activation of Structure

Both fission and fusion reactors will activate their shielding and structure to some extent. In fission reactors, activation arises from fission products and the interaction of low energy (below 1 MeV) neutrons with the core and shielding materials; in fusion reactors, the activation arises from 14.1 or 2.45 MeV neutrons which activate the material of the first wall and blanket. The magnitude of this activation is evident in the last line of Table I, where the nonvolatile core or blanket inventory is listed for the Chernobyl reactor at the time of the accident on April 26, 1986 [ref. 9, 10] and for the Starfire DT Tokamak after one year of operation [ref. 11]. These inventories were the result of about a year of full power operation at 1 gigawatt of electrical power output in each case. These inventories are, respectively, about 30 times and about 120 times the radioactive release of the Chernobyl accident, and are clearly much too large to dump into the atmosphere. Thus, any fission or DT fusion reactor, once operated in space, may become a serious radiological safety hazard upon re-entry into the atmosphere.

In low earth orbit, there is a narrow band of orbital altitudes within which manned operations are possible. Below approximately 300 kilometers, atmospheric drag is so large that the orbit of a space station would decay in a relatively short period of time; above about 500 kilometers, radiation fluxes from particles trapped in the magnetosphere are sufficiently high that sustained manned operations are not possible. Parenthetically, it is not generally realized that the Apollo astronauts acquired a whole body radiation dose of 50 rads during one round trip through the earth's magnetosphere. This is approximately 1/10 of the L-50 fatal dose.

Because of the hazard of the radioactive inventory of fission or fusion reactors, these reactors should be parked, after use, in a "nuclear safe orbit", that is, an orbit that is sufficiently high above the earth's surface that atmospheric drag will not cause the reactor to re-enter the atmosphere until the longest-lived radionuclide of any significance decays. For fission reactors, the lowest nuclear safe orbit is about 700 kilometers, thus placing the parking orbit for nuclear fission reactors beyond the 500 mile limit where manned operations are possible. The radionuclides in activated DT tokamak fusion reactors are, in most blanket designs, not as long-lived as those of fission reactors, and their nuclear safe orbit may be somewhat lower than the 700 kilometers appropriate for fission reactors.

The fact that the nuclear safe orbit is likely to be above the altitude band where manned operations are possible leaves open the possibility that a fission or fusion reactor associated with a manned space station might reenter the atmosphere. If it did so, the last line of Table I implies that an amount, in Curies, of radioactive material could be released into the atmosphere by the reentry process which would be about 30 times the Chernobyl release for a gigawatt fission reactor, and about 120 times the Chernobyl release for a DT fusion reactor comparable to the Starfire tokamak.

## Availability of Fusionable Fuels

Most fusionable fuels are available for space applications in unlimited quantities. The only two fusionable fuels which may be in short supply are tritium and $^3$He. Tritium has a half-life for decay into $^3$He of 12.3 years,

$$T \xrightarrow{12.3\ \mathrm{YR}} {}^{3}\mathrm{He} + e \qquad (7)$$

and is not found to any significant extent in nature. $^3$He is a stable isotope of helium, but is found with an isotopic abundance of only about one part in $10^6$ on the surface of the earth.

Only a very limited number of fusion reactors will ever be required for space applications, and their fueling requirements will be far smaller than, for example, a ground-based fusion economy for the electric utilities. Thus, for space applications, it becomes possible to consider sources of tritium and $^3$He which would not be feasible for ground-based electric utility applications.

In space, severe mass penalties will probably be associated with the recovery of unburned tritium fuel for reinjection into the plasma, or with any attempt to breed tritium on a space vehicle by the neutron-lithium breeding reactions [ref. 12, page 202] which have been proposed for ground-based electric utility DT powerplant reactors. The relatively small amount of tritium required for space missions, compared to the much larger amounts which might be needed in ground-based electric utility powerplants, should make it possible to increase the tritium breeding ratio of ground-based powerplants to an extent which will produce enough additional tritium for space missions.

Because of its extremely low isotopic abundance, very little $^3$He will be available from natural sources on the earth's surface. Another source of $^3$He is the decay of tritium, according to Equation 7 above, which is used for weapons and other purposes. It is likely that the total amount of $^3$He which will be available early in the 21st century will be no more than a few hundred kilograms, an amount barely adequate for one or two space missions.

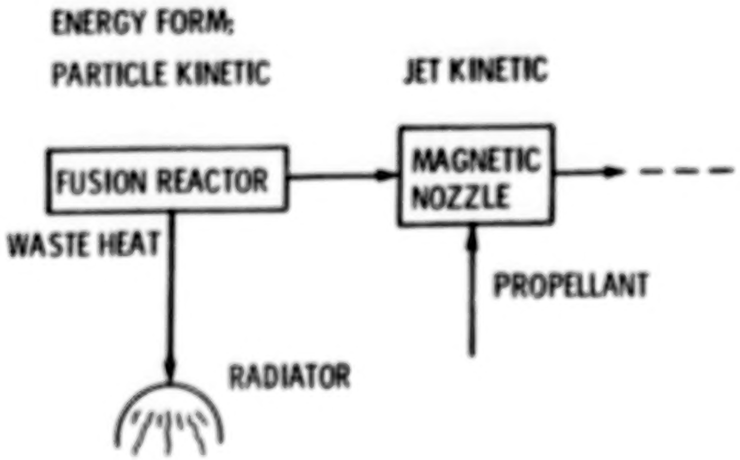
Other potential sources of $^3$He include the decay of tritium specially produced by fission reactors for space missions; and semi-catalyzed DD reactors, following the suggestion of Miley, et al. [ref. 13], which are part of a ground-based fusion economy using DD reactions. The most promising source of large amounts of $^3$He appears to be heating regolith on the lunar surface which has been implanted with $^3$He over geological ages by the solar wind [refs. 7, 14]. It appears that essentially unlimited amounts of $^3$He would be available for space missions from sources on the lunar surface, or the atmospheres of the outer planets or their satellites which have retained light elements in their atmospheres.

## Safety and Environmental Factors Affecting Confinement Concept

In this section some of the factors which affect the choice of confinement concept will be examined. This includes both the choice between inertial and magnetic fusion energy, and also the choice among magnetic containment concepts.

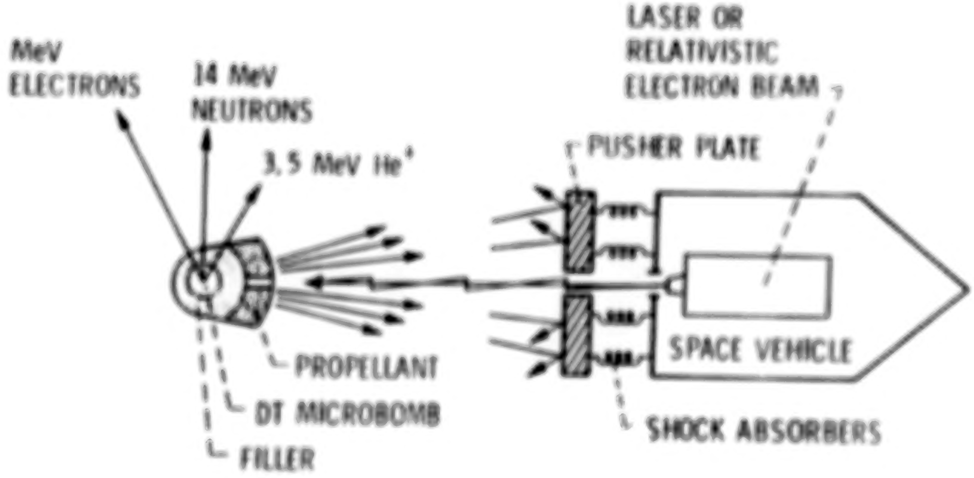
The most effective fusion propulsion system, which minimizes the size of the radiator required and the total mass, is the direct fusion rocket shown on Figure 2 [refs. 1,2]. In this propulsion system, the escaping unreacted fuel and reaction products are expanded in a magnetic nozzle, where they are mixed with cold propellant to achieve a unidirectional plasma jet with a spread of velocities, but an optimum mean exhaust velocity [Ref. 15]. If an aneutronic fusion reaction (one which produces few neutrons from all sources, including side reactions) is used, and if all the unburned reaction products appear in the exhaust jet, relatively little heat energy remains on board the spacecraft to be disposed of by massive radiators. This direct propulsion system may either require a very high burn-up fraction, or waste what might be a scarce or expensive fuel.

DIRECT FUSION ROCKET


ENERGY FORM:
PARTICLE KINETIC
JET KINETIC
FUSION REACTOR
MAGNETIC NOZZLE
WASTE HEAT
PROPELLANT
RADIATOR

On Figure 3 is shown one of many schemes put forward for space propulsion using inertial fusion. In this concept, a laser or charged particle beam is fired at a DT pellet which explodes, after which the propellent and some of the filler material bounces off a pusher plate, located at a sufficient distance that significant ablation does not occur. This pusher plate is connected to the vehicle by springs and dashpots, which absorb and transmit momentum to the spacecraft. Sometimes a magnetic field, which "catches" the charged reaction products, replaces the pusher plate in this concept. The repetitive explosion of these fusion microbombs can yield high accelerations, and short interplanetary round trip times. In most inertial fusion schemes, 14 MeV neutrons, 3.5 MeV helium-4 ions, radiation, and other materials are emitted isotropically (except those that intercept the pusher plate) into space.

NUCLEAR PULSE PROPULSION USING FUSION MICROBOMBS


MeV ELECTRONS
14 MeV NEUTRONS
3.5 MeV He$^4$
LASER OR RELATIVISTIC ELECTRON BEAM
PUSHER PLATE
SPACE VEHICLE
PROPELLANT
DT MICROBOMB
FILLER
SHOCK ABSORBERS

### Inertial Fusion in Space

Over the past 20 years there have been numerous design studies of space propulsion systems using inertial fusion. Most of these studies are of direct fusion rockets of the type indicated schematically on Figure 3, in which the initiating energy pulse is provided by lasers or particle beams. There have been few if any studies of inertial fusion systems for the primary purpose of generating electrical power in space. The large recirculating power flows usually required for inertial fusion, and the resulting mass penalties, may have discouraged detailed studies of inertial fusion for such applications.

Another characteristic of most engineering design studies of inertial fusion propulsion systems is that, as indicated in Figure 3, the neutrons and much of the radiant energy are unshielded and escape freely into the space environment. There appear to be few, if any, engineering design studies of inertial fusion space propulsion systems which fully shield the neutron, charged particle, and radiant energy fluxes produced by the explosion of the pellets.

Another problem with inertial fusion space propulsion systems is that investigations of burn dynamics with classified computer codes indicate that the energy gains of advanced fuels are insufficent to marginal for a pellet burn [Ref. 7]. Two unclassified papers [Refs. 16, 17] on advanced fuel inertial confinement show that a special pellet design (AFLINT) may be capable of burning the DD reaction, although with very high recirculating power flows [Ref. 17]. For such reasons as these, published design studies assume the DT reaction, the very high reactivity of which assures an adequate pellet burn. If inertial fusion systems in space are limited to the DT reaction, the implications of this are rather serious for the overall propulsion system. One must be concerned about the risk inherent in the tritium fuel, as described previously, and one must avoid contaminating either the atmosphere or the magnetosphere with radioactive tritium in the event of an accident, or escape of the tritium as unburned propellent from the reactor.

## Conclusions

### Magnetic vs Inertial Fusion in Space

For environmental and safety reasons touched upon in the above discussion, it appears that inertial fusion is at a disadvantage with respect magnetic fusion for application to space power and propulsion systems. Inertial fusion systems may be restricted to the DT reaction, raising the possibility of contamination of the atmosphere and/or the magnetosphere with radioactive tritium; and it appears difficult to shield the neutron, the charged particle, and the radiant energy fluxes that result from the explosion of the pellets without paying a large mass penalty for DT inertial fusion systems. It appears difficult to burn advanced fuels with reduced neutron production in inertial fusion systems because of the relatively low reactivity of advanced fuels relative to the DT reaction. Unclassified studies of advanced fuel inertial confinement [Refs. 16, 17] have demonstrated the feasibility of a fusion burn, although with high recirculating powers [Ref. 17]. Further research is needed to demonstrate really attractive ICF performance with advanced fuels. On present evidence, it appears that safety and environmental considerations make inertial fusion systems relatively a more difficult prospect for space applications than magnetic fusion reactors which, if they have low recirculating power flows, can burn advanced fuels.

## Choice of Fusion Reaction for Space Applications

Considerations discussed in Ref. 12, Chapters 8 and 9, indicate that at kinetic temperatures below 100 keV, only the DT, catalyzed DD, and D$^3$He reactions are capable of producing power densities in the range of 1 to 10 megawatts per cubic meter at number densities, confinement times, and kinetic temperatures which are modest extensions of current DT tokamak research. If it is desired to minimize the transportation, handling, and leakage of tritium into the environment, that leaves only the catalyzed DD and D$^3$He reactions. If it is further desired to minimize the radioactivation and shielding mass associated with high levels of neutron production, that leaves only the D$^3$He reaction. On present evidence, it appears that the D$^3$He reaction is the all-around best choice for space applications of magnetic fusion energy.

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# Attitude Control Requirements for Various Solar Sail Missions

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## Summary

This paper summarizes the differences between the attitude control requirements for various types of proposed solar sail missions (Earth-orbiting; heliocentric; asteroid rendezvous). In particular, it is pointed out that the most demanding type of mission is the Earth-orbiting one, with the solar orbit case quite benign and asteroid station-keeping only slightly more difficult.

It is then shown, using numerical results derived for the British Solar Sail Group Earth-orbiting design, that the disturbance torques acting on a realistic sail can completely dominate the torques required for nominal maneuvering of an 'ideal' sail. This is obviously an important consideration when sizing control actuators; not so obvious is the fact that it makes the 'standard' rotating vane actuator quite unsatisfactory in practice. The reason for this is given here, and a set of new actuators described which avoids the difficulty.

## Solar Sailing History:

- *Concept:* originally described by Tsiolkovsky.

- *In-flight experience:* (all for attitude torque generation, not propulsion).

  Mariner 4: limited use of 'fans' on solar array tips.

  Mariner 10: significant use of differential solar array rotation to balance roll disturbance torques. Use of this technique allowed the full mission to be flown, despite a gyro resonance problem that wasted enough propellant to threaten it.

  OTS-2: European Space Agency (ESA) communications spacecraft test article in GEO.

- *Proposed propulsion demonstrations:*

  JPL Halley's Comet rendezvous: rejected in favor of electric propulsion (later itself dropped).

  ESA Halley's Comet rendezvous: essentially a scaled-down version of the JPL sail, proposed for launch on an Ariane test vehicle.

  Amateur Earth-orbiting sails: for instance, the French U3P group's proposal for 2 or 3 sails to be launched on the Ariane 4 test vehicle and then race to the Moon's orbit. This stimulated research in various countries (e.g. Japan; Czechoslovakia; Great Britain [British Solar Sail Group]). A similar race has recently also been proposed by the AIAA to commemorate Columbus' mission in 1492. Another group very active in amateur sail design is the Pasadena-based World Space Foundation, which proposed a sub-scale version of the JPL sail in low Earth orbit.

## Various Solar Sail Missions:

- *Heliocentric:* for example, the proposed rendezvous missions with Halley's Comet.

  Such missions are the least demanding from the point of view of attitude control. Orbit-raising requires a constant angle between sail normal and orbital radius : this leads to slow maneuvering, as well as simple sensor requirements.

- *Earth-orbiting:* e.g., the various proposed amateur sails.

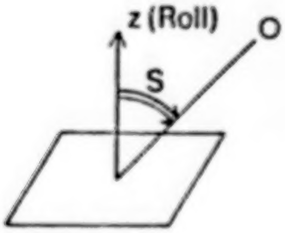
  More demanding, as the required sail attitude now changes throughout the spacecraft orbit. The required maneuver rates are thus much higher than in the previous case (although still very low by "conventional" standards). Sensor requirements are also more complicated, as the desired sail attitude is not now fixed relative to the Sun (or Earth).

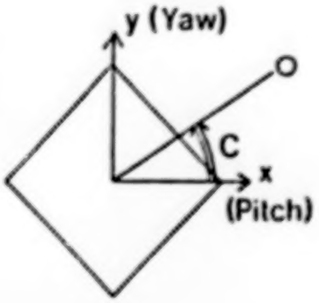
- *Asteroid reconnaisance:* There is currently great interest in studying the minor bodies in the Solar System. A result of this is the decision to target all NASA outer planet missions to at least one asteroid fly-by, with Galileo being the first spacecraft to do this. Considerably greater information could be obtained by long-term study of one or more asteroids from a spacecraft station-keeping with it.

  Such a mission appears to be well suited to solar sailing. Once boosted to Earth escape (by conventional chemical propulsion), the flight would decompose into: a heliocentric portion (Earth to sphere of influence of first asteroid target), with properties as outlined above; a phase involving maneuvering into orbit about the asteroid, with properties comparable to high-altitude Earth-orbit flight. The work of [3] for ion propulsion indicates that a modest sail would be adequate. Reboost and rendezvous with subsequent targets would be done in an entirely similar fashion.

## Real vs. Ideal Sail Properties:

- *Ideal sail dynamics:* quite simple. If the sail is assumed to be **perfectly reflective** and **flat**, the solar force is always perpendicular to the sail plane, with magnitude $F = 2pA\cos^2 S$, where $p = 4.65 \times 10^{-6}$ N/m$^2$ at 1 A.U.:





Note that F is independent of C; furthermore, as the solar force acts along the roll axis, no roll torque can be generated by e.g. center of mass offsets.

- *Real sail dynamics:* the main difference is that any real sail has non-zero absorptivity $a_s$, leading to a more complicated solar force with a down-Sun term. In the coordinates above,

$$\mathbf{F} = -pA|\cos S|\,(a_s \sin S \cos C,\ a_s \sin S \sin C,\ (2-a_s)\cos S)^T.$$

Not only is this more complicated than the ideal sail force, but roll torques can now be generated by a shift in the center of pressure relative to the center of mass. If this shift is $(x, y, z)^T$, then the resultant torque is

$$\mathbf{g} = pA|\cos S| \begin{pmatrix} z a_s \sin S \sin C - y(2-a_s)\cos S \\ x(2-a_s)\cos S - z a_s \sin S \cos C \\ a_s \sin S(y \cos C - x \sin C) \end{pmatrix}. \quad \leftarrow [\text{Small, but} \neq 0]$$

# Disturbance Torque Sources:

- *Typical disturbance mechanisms:* (Details depend on the orbital parameters and design of the sail considered.)

  Center of pressure shift: this would result, for instance, if different parts of the sail degrade (increase in absorptivity) at different rates. LDEF results on the effects of exposing aluminized Kapton to the space environment should help quantify this.

  Center of mass shift: a typical way this can come about is as a result of thermal bending of the booms which support the sail. Such bending can be considerable, even for small thermal gradients across the booms, because of their great lengths. This results in a solar angle-dependent C.M. vs. C.P. shift.

  Gravity gradient torques: can be significant for Earth-orbiting sails.

  Spacecraft initial asymmetries: e.g. variability in the mass properties of sail and boom material and in the reflectivity of sail material; imperfect control of the deployment angles of booms, leading to a slightly unsymmetrical deployed sail.

  Negligible effects:

  - Boom bending caused by solar *radiation pressure* rather than solar *heating.*

  - Force due to the solar *wind* rather than photon pressure. (The solar wind pressure is about 4 orders of magnitude weaker than that of the photons.)

  - Atmospheric drag and magnetic torques: negligible at the high altitudes required for any Earth-orbiting sail.

## Disturbance Torque Numerical Results: The BSSG Sail Design.

- *Outline sail design:* aluminized Kapton sail of area 2400 $m^2$, supported on 4 GFRP booms and deployed using a simplified 'wrap-rib' technique. Total spacecraft mass of 200 kg gives a modest sail acceleration of about $10^{-4}$ $m/s^2$, sufficient for demonstration purposes. (For more information of the design philosophy and details of the British Solar Sail Group design, see [1] and [2].)

- *Predicted worst-case disturbance torques:* (from [2])

| Cause of torque | Max. roll (Nm) | Max. pitch/yaw (Nm) |
|---|---|---|
| Sail degradation | $3.76 \times 10^{-5}$ | $6.77 \times 10^{-4}$ |
| Boom thermal bend | $5.44 \times 10^{-6}$ | $2.44 \times 10^{-4}$ |
| Gravity gradient | 0 | $1.11 \times 10^{-4}$ |
| Initial asymmetry | $3.35 \times 10^{-5}$ | $6.97 \times 10^{-4}$ |
| TOTAL: | $7.65 \times 10^{-5}$ | $1.73 \times 10^{-3}$ |

- *Observations and implications:*

  **(1)** These disturbance torques are considerably greater than the nominal steering torques required for an ideal sail, even for the relatively demanding Earth-orbiting BSSG mission.

  **(2)** The gravity gradient torque is predictable; that due to thermal bending is calculable if the booms are instrumented, e.g. with strain gauges. The remaining torques, which make up the bulk of the total, result from a nearly **constant** center of pressure/center of mass shift.

  **(3)** The roll disturbances are much lower than those in pitch and yaw. It is therefore inefficient to have actuators which can provide roll torques as large as those in pitch/yaw.

## "Traditional" Radiation Pressure Actuators:

- *Variable-angle vanes:* the only extensive in-flight radiation pressure attitude control experience to date, i.e. Mariner 10 and OTS-2, was carried out as an 'add-on', using existing spacecraft hardware. These spacecraft used tiltable solar panels as solar pressure vanes, and this type of actuator has been used extensively in many solar sail designs (e.g. the JPL and ESA square sails both had rotating vanes at the boom tips). However, the preceding disturbance analysis points up some severe practical limitations of this type of actuator:

  Roll sensitivity: such vanes must be sized for the required pitch/yaw torques, but produce roll torques of the same magnitude. A misalignment of the vanes of just $1^{\circ}$ can thus be shown [2] to give rise to a roll torque as large as all other disturbance sources combined.

  Duty cycle: as already noted, most sail disturbances result from a slowly-varying center of pressure/center of mass shift. They thus vary with solar angles S and C as given by the equation for **g** on page 4 with x, y and z roughly constant. But the torque produced by a rotating vane varies with **its** solar angles, not those of the sail. Thus, using a set of rotating vanes to counteract even a constant C.P./C.M. shift will require frequent vane rotations, complicating the control problem and reducing motor lifetimes.

- *In-plane ballast masses:* this technique, incorporated into the WSF sail for pitch/yaw, avoids the above problems. In particular, a constant C.M./C.P. offset is now easily compensated for by a constant offset of the ballast mass. It is important to note though that, as the disturbance torques dominate the nominal maneuver torques, the ballast mass must be sized with the disturbances in mind. This will typically result in a requirement that the ballast mass be allowed to move along the entire length of the booms.

## Novel Radiation Pressure Actuators:

- *Variable-area vanes:* these avoid some of the problems of variable-angle vanes. A pair of 'roller-blind' vanes mounted on the tips of two adjacent booms and parallel to the sail plane would allow a constant C.P./C.M. shift to be compensated for by a constant vane offset. This greatly simplifies the problem of sequencing actuator commands. Furthermore, no large roll torque errors are produced by this arrangement; the undesirable coupling of the rotating vanes is avoided. A third small vane normal to the sail plane would now suffice for counteracting the low roll disturbances acting on the sail.

- *Product of inertia modulation:* this makes use of a mass on a variable-length boom mounted at the end of a sail boom and moving normal to the sail plane. This allows the spacecraft to be made controllably unbalanced: e.g. the product of inertia $I_{xz}$ can be altered as required, so coupling the pitch and roll axes. The result of this is that a commanded pitch torque gives rise to an 'effective' roll torque of specified size. This technique may have applications to 'standard' spacecraft; in the BSSG design, the CCD camera was used as the movable ballast mass.

- *Phased roll control:* for a real sail, two actuators are actually adequate for pitch, yaw and limited roll control, which is all that is required. From the expression for **g** with $z = 0$, it can be seen that a (small) roll torque is produced by altering x and y. The ratio of roll torque to pitch/yaw torque is proportional to tanS, and so is small for small S and large for S approaching $90^o$; furthermore, it can be of either sign. As a result of this, modulating e.g. y about the average value needed for pure pitch control can produce pitch plus roll control; for instance, if a larger net roll torque is required, setting y low for small S and high for S in the range $50^o$-$70^o$ or so would achieve this.

## Summary:

This paper summarized the differences between the attitude control requirements for various types of proposed solar sail missions (Earth-orbiting; heliocentric; asteroid rendezvous). Numerical results derived for the British Solar Sail Group Earth-orbiting design were then used to show that the disturbance torques acting on a realistic sail can dominate the torques required for nominal maneuvering of an 'ideal' sail. This is obviously an important consideration when sizing actuators; it also makes the 'standard' rotating vane quite a poor choice in practice. The reason for this was given in the paper, and a set of new actuators described which avoids the difficulty.

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## Acknowledgements

The author wishes to acknowledge the extensive work that Dr. P.Q. Collins of Imperial College, London put into the sail design of the British Solar Sail Group.

N91-22151

# PULSED PLASMOID ELECTRIC PROPULSION

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## ABSTRACT

A method of electric propulsion is explored where plasmoids such as spheromaks and field-reversed configurations (FRC) are formed and then allowed to expand down a diverging conducting shell. The plasmoids contain a toroidal electric current that provides both heating and a confining magnetic field. They are free to translate because there are no externally-supplied magnetic fields that would restrict motion. Image currents in the diverging conducting shell keep the plasmoids from contacting the wall. Because these currents translate relative to the wall[1], losses due to magnetic flux diffusion into the wall are minimized. During the expansion of the plasma in the diverging cone, both the inductive and thermal plasma energy are converted to directed kinetic energy producing thrust. Specific impulses can be in the 4000 to 20000 sec range with thrusts from 0.1 to 1000 Newtons, depending on available power.

## BACKGROUND

While chemical rockets are capable of very high thrust, consumption of propellant is excessive for many space missions such as long-term satellite station keeping and planetary missions. Their nozzle exhaust velocities are limited by heats of reaction to $u_e \simeq 5000$ m/sec, limiting specific impulse to $I_s \simeq 500$ sec. Nuclear thermal rockets using hydrogen propellant can approach 950 sec, still too low for many missions.

For many space missions, specific impulses in the 2000 to 10000 sec range are desirable in order to limit propellant consumption. However, the power in the exhaust stream

$$P_e = \frac{1}{2}\dot{m}u_e^2 \quad ,$$

must be supplied by means other than chemical. Here $\dot{m}$ is the propellant mass flow in kg/sec. The only practical form is electrical power and an electric generator, either solar or nuclear (fission or fusion), must be carried on board. (Batteries, fuel cells, and other chemical energy systems are not suitable because their fuel supply or energy storage mass is comparable to, and subject to the same limitations as, chemical rockets.)

When one trades off power plant mass against propellant mass, it becomes clear that every mission has an optimum specific impulse above which power plant mass becomes excessive and below which propellant mass dominates.

In terms of propulsion, space missions can be characterized by a certain payload mass $m_p$ carried through a velocity increment $\Delta V$. Typical $\Delta V$s range from 9 km/sec to go from low earth to geosynchronous earth orbits (LEO - GEO), to 14 km/sec to go from LEO to Mars orbit and return, to 110 km/sec to go from LEO to Saturn orbit and return (ref. 1). Payload masses can range from 10s of kg for a small experimental satellite to hundreds of tonnes for a manned Mars mission.

Existing electrically-powered thrusters can supply high specific impulse at rather low thrust. There are three kinds: electrothermal (resistojets and arcjets), electrostatic (ion thrusters), and electromagnetic [magnetoplasmadynamic (MPD) thrusters].

Resistojets use resistance wire to heat a flowing gas. Arcjets do the same with arc discharges across the gas. Ion thrusters use perforated accelerator plates at high potential difference to accelerate an ionized gas. MPD thrusters are similar to arcjets except they also exploit the $\mathbf{J} \times \mathbf{B}$ force in the arc to blow the plasma through a nozzle.

Only the MPD thruster is capable of both high thrust and high specific impulse. However, these thrusters have rather short lifetimes due to electrode erosion (ref. 2). The plasmoid propulsion thruster discussed below may fill the need for high-power levels with long life and high specific impulse.

## DESCRIPTION OF THE CONCEPT

The basic idea is shown in Fig. 1. Compact tori (CT) are formed inside an electrically conducting chamber. A compact torus is a toroidal-shaped ionized gas containing an internal electric current traveling mainly along the minor axis of the torus. Formation can be either by a coaxial Marshall gun or by a rapidly pulsed induction coil. Because the internal current can twist helically at locations off the minor axis, the configuration could take two forms: (1) a spheromak which has near-equal magnetic field strength at the plasma edge in directions parallel and perpendicular to minor axis, or (2) a field-reversed configuration (FRC) where the current has no helical twist and all of the magnetic field from that current is perpendicular to the minor axis. Both configurations are being explored for nuclear fusion applications. The difference between the two is that the FRC may be capable of higher plasma pressure but appears to have poorer confinement of the plasma thermal energy than the spheromak.

Once formed, the compact torus or plasmoid is pushed into a diverging conducting shell where it spontaneously accelerates down the divergence so as to reduce its internal

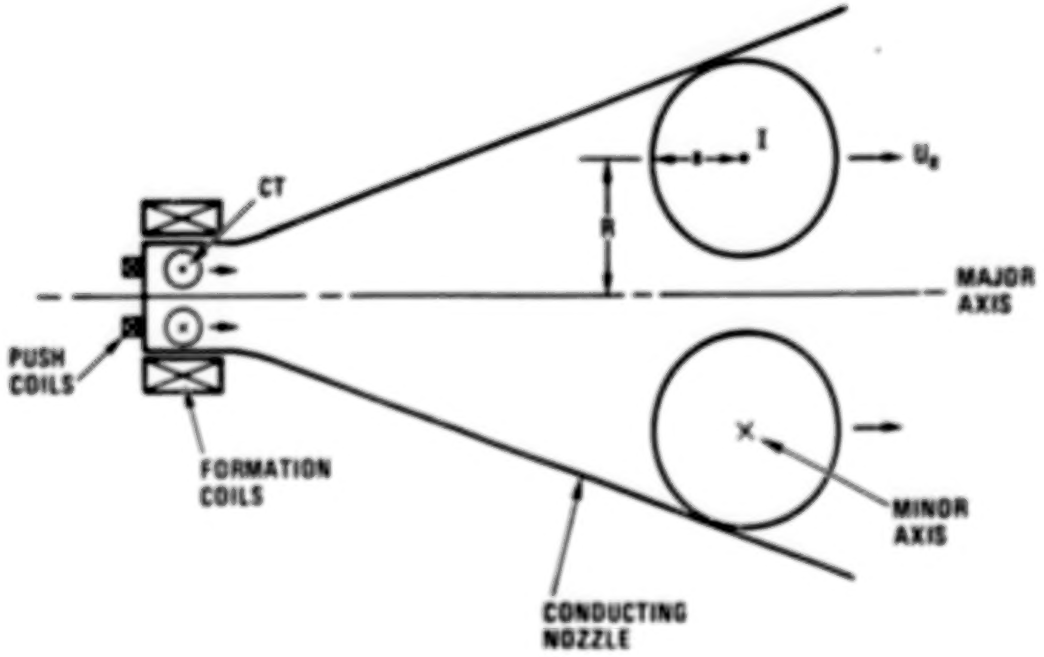

Fig. 1. Plasmoid propulsion concept.

pressure. Image currents in the shell keep the plasma off the wall and provide the physical push that gives the thrust.

This is a pulsed system and thrust variation is accomplished by simply varying the rate at which these plasmoids are formed. We believe that realistic pulse rates can vary from 100 Hz on down.

## ANALYSIS

In this section, we examined the dynamics and energetics of plasmoid expansion down the conducting shell in order to arrive at preliminary estimates of specific impulse, thrust, and losses.

The magnetic energy in the plasma is

$$E_m = \frac{1}{2} L I^2 \quad ,$$

where I is the internal current in the plasma. L is the plasma inductance and is very nearly $\mu_0 R$ for aspect ratios $R/a \simeq 1.5$, typical of spheromaks. Here $\mu_0$ is the magnetic permeability of free space; and $a$ and $R$ are the plasma minor and major radii, respectively. The thermal energy, assuming electron and ion temperatures are equal, is

$$E_t = 3n\,T V_p \quad ,$$

where $n$ and T are the the ion density and temperature, respectively, and $V_p$ the plasma volume ($= 2\pi^2 Ra^2$).

During an adiabatic expansion, total magnetic flux is conserved. That is,

$$B_p\, aR = \text{constant} \quad ,$$

where $B_p$ is the magnetic field due to the plasma current I:

$$B_p = \frac{\mu_0 I}{2\pi a} \quad .$$

Assuming geometric similarity during expansion ($R \sim a$), it can be shown that

$$Ia = \text{constant} \quad ,$$

Therefore, as the plasma expands, the internal total current drops linearly with size.

Over an expansion from $a_1$ to $a_2$, it can be shown from the above that the magnetic energy scales like

$$\frac{E_{m2}}{E_{m1}} = \frac{a_1}{a_2} \quad .$$

A five-fold increase in size will therefore convert 80% of the magnetic energy to kinetic energy of the particles by interaction of the plasma current and the image current in the inclined shell.

The thermal energy in the plasma is also converted to kinetic energy during expansion. With an adiabatic expansion

$$\frac{n_2}{n_1} = \left[\frac{a_1}{a_2}\right]^3 \equiv \chi \quad ,$$

$$\frac{T_2}{T_1} = \chi^{\gamma-1} \quad .$$

Since, for hydrogen, $\gamma = 5/3$, the thermal energy scales like

$$\frac{E_{t2}}{E_{t1}} = \left[\frac{a_1}{a_2}\right]^2 \quad .$$

A five-fold linear expansion will therefore recover 96% of the thermal energy in the plasma and, like the expansion in a regular nozzle, convert it to kinetic energy.

Also of interest is the behavior of $\beta$ during expansion. $\beta$ is the ratio of plasma pressure to magnetic field pressure and is a measure of containment efficiency of the plasma configuration:

$$\beta \sim \frac{n\,T}{(I/a)^2} \quad .$$

It can be shown from the above equations that

$$\frac{\beta_2}{\beta_1} = \frac{a_1}{a_2} \quad .$$

Therefore, $\beta$ is highest at the beginning of expansion. Any disruption of the plasma is likely to occur then rather than later. A disruption is when the plasma structure rapidly breaks down due to pressure-related instabilities. However, since the nozzle presents a free boundary to the plasma, particle motion will be downstream and thrust should be achieved anyway, albeit in a less organized fashion.

If, for simplicity, we assume that all of the plasma energy $E_{tot} = E_m + E_t$ goes to kinetic energy, then the exhaust velocity is

$$u_e = \left[\frac{2\,E_{tot}}{M}\right]^{1/2} \quad ,$$

where M is the total plasma mass:

$$M = (1.67 \cdot 10^{-27}\ \text{kg})\, n\, V_p\, W \quad .$$

Here $n$ is the average ion density and $W$ is the ion atomic weight.

The primary issues which drive the sizing of the system are the time for radiation loss from the hot plasma (typically over 10,000K), and resistive decay time — both relative to the acceleration time of the plasmoid.

Radiation calculations are very complex and only crude estimates can be made here. Hopkins and Rawls (ref. 3) have produced analytical fits to the various radiation loss mechanisms. Figure 2 shows normalized radiation loss $\psi$ for several elements. The total radiation loss, in $W/m^3$, is given by

$$P_r = \psi\, n_e\, n_i \quad ,$$

where $n_e$ and $n_i$ are the electron and ion density of the plasma, respectively. The characteristic radiation time is

$$\tau_r = \frac{E_{tot}}{P_r} \quad .$$

This time gives only a rough measure of the degree of loss through radiation. To do the problem right requires a detailed time-dependent analysis. However, it is useful in assessing feasibility. Note from Fig. 2 that radiation from hydrogen is well under that from the other elements. While it would be nice to use the more massive elements to raise thrust and lower specific impulse, it turns out that radiation loss precludes this option and hydrogen is the element of choice.

Since plasma temperatures are low in order to have high mass through high density for a given plasma pressure (otherwise, specific impulse is too high), only partial ionization occurs. Figure 3 shows ionization of hydrogen as a function of temperature and density (taken from ref. 4). At 1.5 eV and $10^{15}$ cm$^{-3}$ total density, for example, hydrogen is about 50% ionized. Major concerns are the time required for this level of ionization to occur and the interaction of the ions with the remaining neutrals. These issues will be addressed in future work.

The other characteristic time of interest is the resistive decay time, which is the time it takes for the current in the plasma to dissipate due to the internal plasma resistance. If this is short compared to the acceleration time, ohmic dissipation will raise the plasma temperature and dissipate the current. The plasma $\beta$ will increase to the point of disruption. It is not clear if this is a detriment since the plasma will

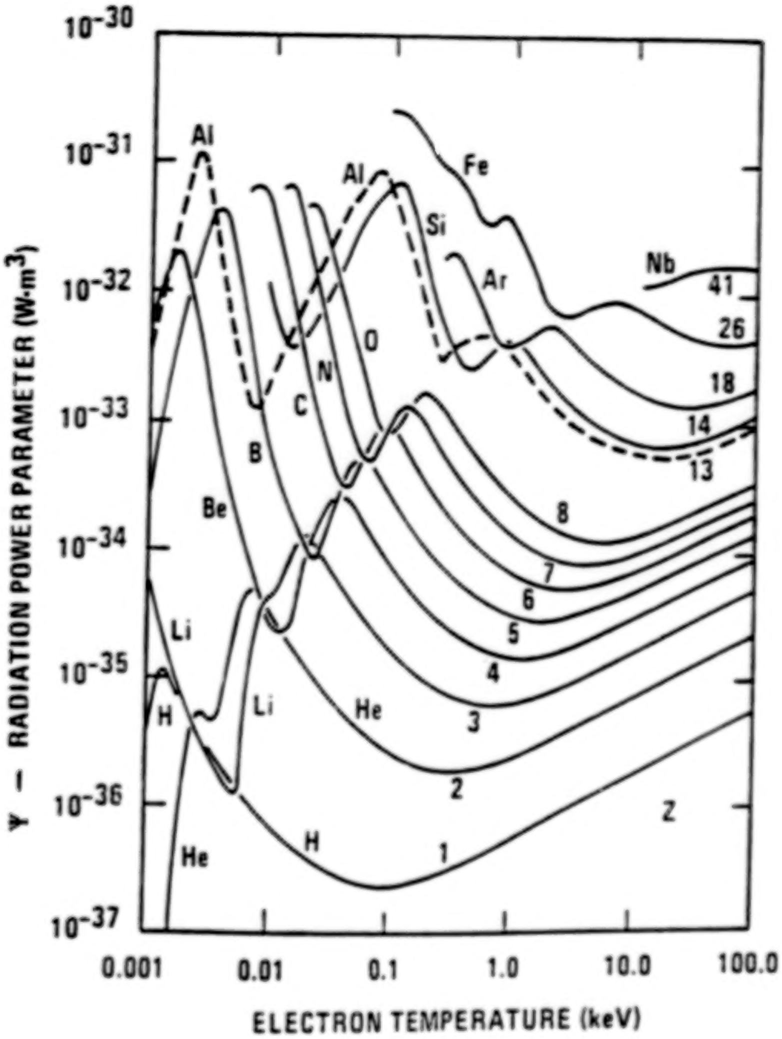

Fig. 2. Total radiation loss parameter as a function of plasma electron temperature. Coronal equilibrium is assumed (from ref. 3).

still expand down the nozzle converting its thermal energy to kinetic. And since, for a given expansion, a greater fraction of thermal energy is converted than magnetic, thrust efficiency could actually be higher. We will also explore this issue in follow-on work.

The full expression for plasma inductance is (ref. 5)

$$\mathrm{L} = \mu_0 R \left[ ln \, \frac{8R}{a} - 2 + \frac{l_i}{2} \right] \quad ,$$

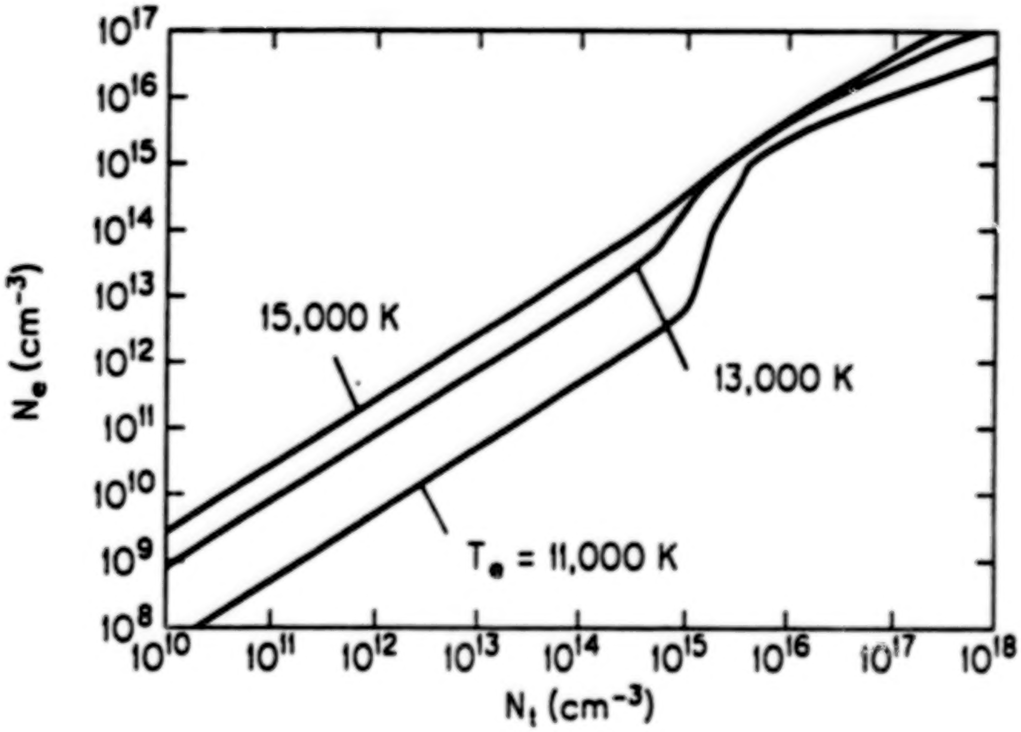

**Fig. 3. Production of electrons in a hydrogen plasma as a function of particle density and electron temperature (from ref. 4).**

where $l_i$ is the plasma internal inductance. For uniform current profiles, which is likely here (due to turbulence), $l_i = 1/2$. With this and an aspect ratio $R/a = 1.5$, the bracket term is very nearly 1.0 and so $L \simeq \mu_0 R$.

The plasma resistivity is

$$\eta = 5.38 \cdot 10^{-5} \, ln \Lambda \, Z \, T^{-1.5} \quad ,$$

where $Z$ is the atomic number of the gas and $ln \Lambda$ is a constant that is typically about 16. The plasma total resistance, assuming flat temperature profiles and no resistivity enhancement due to turbulence or helical pitch of the current, is given by

$$\Omega = \eta \, \frac{2\pi R}{\pi a^2} \quad .$$

The time for the current to decay to 1/e of its initial value is $L/\Omega$.

The characteristic times above must be compared to the time to accelerate the plasmoid down the length of the nozzle. To limit radial kinetic energy the nozzle length $L_n$ should be at least five times the outer radius of the plasmoid. Assuming constant acceleration, the time for acceleration is

$$\tau_{acc} = \frac{2\,L_n}{u_e}$$

The above expressions are sufficient to perform an initial assessment of pulsed plasmoid propulsion. This is done below.

## SCOPING CALCULATIONS

Using the above plasma equations along with some expressions for missions, a short code was written and exercised to determine operating parameter spaces. Two example cases are reported here. If time permits in the future, sensitivity studies could also be performed with this code.

Code output for a small thruster is shown in Table 1. Inputs 1 through 7, 10, and 13 define the plasma while Inputs 8, 9, and 12 refer to the mission. The power plant specific weight of 2 kg/kW is not unreasonable for an advanced nuclear electric plant or an advanced photovoltaic solar array (ref. 6). The radiation parameter of $7 \cdot 10^{-36}$ W-m$^3$ is taken from Fig. 2 with hydrogen at 1.5 eV. Note that this temperature is used to establish the density at a given $\beta$. In actual fact, both temperature and $\beta$ will float depending on the dynamics of the plasma power balance. That analysis is very complex and well beyond the scope of this effort.

With plasma current, $\beta$, temperature, field, and aspect ratio dictated, plasma dimensions, density, total mass and energy are determined from the equations discussed above. Also determined are the radiation loss and resistive decay times, the acceleration times, specific impulse and thrust. Using the mission input parameters, one can readily determine power plant and propellant mass, and a lower bound on acceleration time to the specified $\Delta V$ (it is an underestimate for guidance only because just the payload mass is considered).

The example in Table 1 is for a thruster that could be used to transfer the 1000 kg satellite from LEO to GEO in about 200 days. This is typical of electric propulsion missions and care is taken to protect delicate parts during the long spiral transit through the Van Allen radiation belts. With a specific impulse of 8000 sec and a thrust of 0.5 N, the total mass of propellant used is only 11.4% of the payload mass. Similarly, with an exhaust power of 20 kW, the power plant mass, here likely to be solar cells, is only 5% of the payload.

## TABLE 1
## SMALL HYDROGEN THRUSTER FOR ORBIT RAISING

| | | |
|---|---|---|
| | Input List | |
| 1 | Plasma current, amps | 0.10E+06 |
| 2 | Poloidal beta | 0.20 |
| 3 | Plasma temperature, eV | 1.50 |
| 4 | Poloidal field, tesla | 0.10 |
| 5 | Aspect ratio, $R/a$ | 1.50 |
| 6 | Ion atomic number | 1.0 |
| 7 | Ion atomic weight | 1.00 |
| 8 | Mission delta-V, m/sec | 9000 |
| 9 | Mission payload, kg | 1000 |
| 10 | Radiation parameter, W-m$^3$ | 0.70E-35 |
| 11 | Electron density fraction | 0.30 |
| 12 | Power plant specific weight, kg/kW | 2.0 |
| 13 | Pulse frequency, Hz | 10.0 |
| | Output | |
| | Specific impulse, sec | 8027 |
| | Thrust, N | 0.515 |
| | Ratio of radiation to accel time | 23.3 |
| | Plasma acceleration time, sec | 0.636E-04 |
| | Radiation time, sec | 0.149E-02 |
| | Propellant-to-payload mass ratio | 0.114 |
| | Exhaust power, W | 0.203E+05 |
| | Power plant mass, kg | 0.507E+02 |
| | Ratio of power plant mass to payload | 0.051 |
| | Total plasma energy, J | 0.203E+04 |
| | Plasma thermal energy, J | 0.141E+03 |
| | Plasma magnetic energy, J | 0.188E+04 |
| | Plasma mass, kg | 0.655E-06 |
| | Exhaust mass flow, kg/sec | 0.655E-05 |
| | Ion density, m$^{-3}$ | 0.166E+22 |
| | Electron density, m$^{-3}$ | 0.497E+21 |
| | Plasma volume, m$^3$ | 0.236 |
| | Major radius, m | 0.30 |
| | Plasma radius, m | 0.20 |
| | Plasma o.d., m | 1.0 |
| | Accelerator cone length, m | 2.50 |
| | Plasma current resistive decay time, sec | 0.536E-04 |
| | Spacecraft acceleration time, days | 202 |

Considerable iterations were required to find a configuration that did not radiate excessively. It turned out that magnetic fields must be low, 0.1 T in this case, resulting in fairly large plasmas with low density. This is no surprise since radiation power density goes like $n^2$. In this case, the radiation time is 23 times longer than the acceleration time, providing a comfortable margin to compensate for the crudeness of the calculation. The plasma $L/\Omega$ time of 53.6 $\mu$sec, however, is comparable to the acceleration time of 63.6 $\mu$sec. This suggests that some magnetic energy will convert to thermal, and then to kinetic, during the expansion. Clearly, detailed analysis of all this is needed, but it is very complex.

The overall plasma diameter of 1.0 m is quite large. However, although formation coils would have similar dimensions, their fields should be on the order of the magnetic field (0.1 T) and therefore would have fairly thin windings. The accelerator cone is 2.5 m long with an maximum diameter of perhaps 5 m. The forces on this cone are very small and so it could perhaps be made of very thin aluminum. Such a cone made of 10-mil thick aluminum would weigh about 60 kg (6% of payload) and could double as a heat radiator (more on this later).

A larger thruster, suitable for a manned Mars mission, is shown in Table 2. To increase thrust, plasma current is raised from 100 kA to 1.0 MA. Since thrust goes like $I^3$, it increases from 0.5 to 515 N. Using 20 km/sec as a conservative $\Delta V$ for a Mars mission, a 200 tonne payload requires roughly 90 days worth of acceleration. Actually, it is longer because the propellant and power plant, each 25% as massive as the payload, must be added in as must other components discussed below.

A somewhat fanciful outline of such a spacecraft is shown in Fig. 4. As seen in Table 2, the plasma has a 10 m overall diameter and an accelerator length of 25 m. With a 5:1 expansion, the cone outer diameter is 50 m. This is acceptable only if it can serve multiple purposes. It turns out that it is just about the right size for the reject heat radiator for the nuclear reactor. The reactor power conversion is assumed to be a helium closed cycle gas turbine with a mean heat rejection temperature of 650K (1000K cooling to 300K). Assuming an emissivity of 0.8 and a 30% power conversion efficiency, then the 58 MW(th) reject heat can be radiated at roughly 8 kW/m$^2$. The total radiator area must then be around 7000 m$^2$. The above cone area is about 9000 m$^2$, just a little more than needed, which is fine because it allows for added heat input from plasma radiation (which is then radiated directly to space). If the specific weight of the nozzle can be held to 5 kg/m$^2$, its mass would be about 45,000 kg, which is an acceptable 22% of the payload mass.

There is no reason why the accelerator nozzle must be conical. It is only necessary that the plasma expansion be smooth. If instead it is made parabolic, then it may also be able to serve as a high-gain antenna. A swing-boom signal collector can be placed at the focus when the thruster is off. During the acceleration period from earth, the antenna is pointing properly to receive terrestrial signals.

## TABLE 2
## LARGE HYDROGEN THRUSTER FOR MARS MISSION

| Input List | |
|---|---|
| 1 Plasma current, amps | 0.10E+07 |
| 2 Poloidal beta | 0.20 |
| 3 Plasma temperature, eV | 1.50 |
| 4 Poloidal field, tesla | 0.10 |
| 5 Aspect ratio, $R/a$ | 1.50 |
| 6 Ion atomic number | 1.0 |
| 7 Ion atomic weight | 1.0 |
| 8 Mission delta-V, m/sec | 20,000 |
| 9 Mission payload, kg | 200,000 |
| 10 Radiation parameter, W-$m^3$ | 0.70E-35 |
| 11 Electron density fraction | 0.30 |
| 12 Power plant specific weight, kg/kW | 2.0 |
| 13 Pulse frequency, Hz | 10.0 |

| Output | |
|---|---|
| Specific impulse, sec | 8027 |
| Thrust, N | 515 |
| Ratio of radiation to accel time | 2.34 |
| Plasma acceleration time, sec | 0.636E-03 |
| Radiation time, sec | 0.149E-02 |
| Propellant-to-payload mass ratio | 0.254 |
| Exhaust power, W | 0.203E+08 |
| Power plant mass, kg | 0.507E+05 |
| Ratio of power plant mass to payload | 0.253 |
| Total plasma energy, J | 0.203E+07 |
| Plasma thermal energy, J | 0.141E+06 |
| Plasma magnetic energy, J | 0.188E+07 |
| Plasma mass, kg | 0.655E-03 |
| Exhaust mass flow, kg/sec | 0.655E-02 |
| Ion density, $m^{-3}$ | 0.166E+22 |
| Electron density, $m^{-3}$ | 0.497E+21 |
| Plasma volume, $m^3$ | 236 |
| Major radius, m | 3.0 |
| Plasma radius, m | 2.0 |
| Plasma o.d., m | 10.0 |
| Accelerator cone length, m | 25.00 |
| Plasma current resistive decay time, sec | 0.536E-02 |
| Spacecraft acceleration time, days | 89.9 |

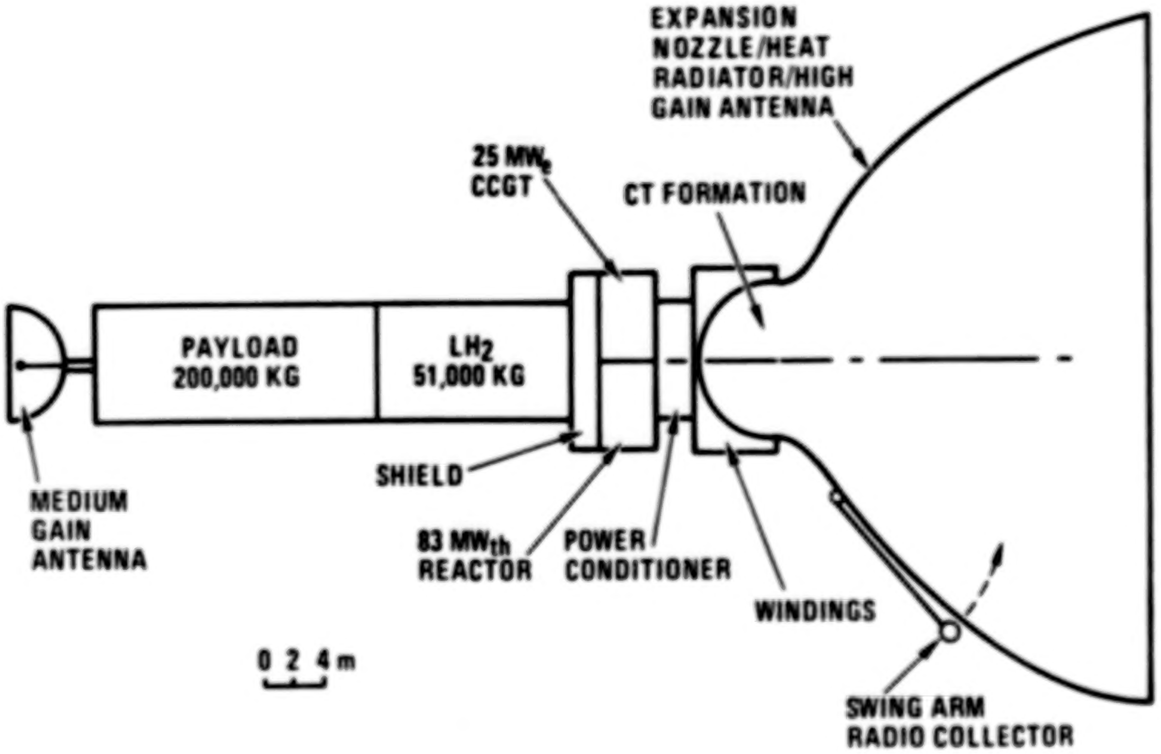
EXPANSION NOZZLE/HEAT RADIATOR/HIGH GAIN ANTENNA
25 $MW_e$ CCGT
CT FORMATION
PAYLOAD 200,000 KG
$LH_2$ 51,000 KG
SHIELD
MEDIUM GAIN ANTENNA
83 $MW_{th}$ REACTOR
POWER CONDITIONER
WINDINGS
0 2 4 m
SWING ARM RADIO COLLECTOR

Fig. 4. Schematic of space station for manned Mars mission. Note that the large rocket nozzle serves multiple purposes.

## DISCUSSION

The major attraction to this thruster is the potential for high thrust and high specific impulse and, if electrodes are not needed for formation, long lifetime. One would also expect the energy efficiency, which is the ratio of exhaust power to power supplied, to be quite high. In these respects, plasmoid propulsion appears to fill a gap in the arsenal of thrusters now available.

The main problems stem from radiation loss and the resulting large plasma dimensions needed to keep it in check and the partial ionization of the gas. The large size is compensated for by the low magnetic fields and low forces on the accelerator nozzle, permitting it to be made of thin-walled material. If the nozzle can in fact double as a heat radiator, then the large size is not a handicap.

There has been no discussion thus far of the power conditioning required. This is a pulsed system and high voltages with fast risetimes are required to form the plasma. In the case of the Mars thruster, total energies around 2 MJ must be supplied. If capacitors were used in this case, their mass would be roughly 40,000 kg, about 20% of the payload mass. This may actually be tolerable, although other pulsed energy sources such as inductors should be explored as well. If the pulse frequency can be increased, the energy per pulse will drop, lowering not only the capacitor mass but the mass of the large accelerator cone. The 10 Hz chosen for the examples seemed reasonable from the standpoint of circuit recharging and chamber clearing. It can probably be much higher.

If one adds up all the major hardware for the Mars vehicle above, including the capacitors, one gets a mass roughly equal to the payload mass; *i.e.*, the payload fraction is 50% of the total initial mass, which would be very appealing for a Mars mission.

## FUTURE WORK

The next step is to perform time-dependent plasma analyses and to scope out the design of the power conditioning system and plasma startup system. All of this is needed before an accurate estimate can be made of thrust efficiency, which is a very important number because it determines the size of the power plant needed.

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N91-22152

# LASER DRIVEN LAUNCH VEHICLES FOR CONTINUOUS ACCESS TO SPACE

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## ABSTRACT

The availability of megawatt laser systems in the next century will make laser launch systems from ground to orbit feasible and useful. Systems studies indicate launch capabilities of 1 ton payload per gigawatt laser power. This corresponds to 20 Kg payloads with 20 megawatt lasers. A launch repetition frequency of one payload every 10 min would allow delivery of parts for assembly of large space structures, and would allow resupply of space stations on a continuous basis. Recent research in ground-to-orbit laser propulsion has emphasized laser supported detonation wave thrusters driven by repetitively pulsed infrared lasers. In this propulsion concept each laser repetition cycle consists of two pulses. A lower energy first pulse is used to vaporize a small amount of solid propellant and then after a brief expansion period of a few microseconds, a second and higher energy laser pulse is used to drive a detonation wave through the expanded vapor. Temperatures of order 10,000 K are achieved. During a several millisecond intercycle delay, expansion of the hot vapor converts thermal energy to directed kinetic energy. High specific impulses of ~600 to 800s are achievable at energy conversion efficiencies of ~20 to 40 percent. The physics of such thrusters has been explored both theoretically and in the laboratory. We report here the results of numerical studies comparing the detonation wave properties of various candidate propellants, and the simulation of thruster performance under realistic conditions. Experimental measurements designed to test the theoretical predictions are also presented. We discuss measurements of radiance and opacity in absorption waves, and mass loss and momentum transfer. These data are interpreted in terms of specific impulse and energy conversion efficiency. Directions for future research are indicated.

The availability of megawatt laser systems in the next century will make laser launch systems from ground to orbit feasible and useful. Systems studies indicate launch capabilities of 1 ton payload per gigawatt laser power. This corresponds to 20 Kg payloads with 20 MW lasers. A launch repetition frequency of one payload every 10 min would allow delivery of parts for assembly of large space structures, and would allow resupply of space stations on a continuous basis (Figure 1).

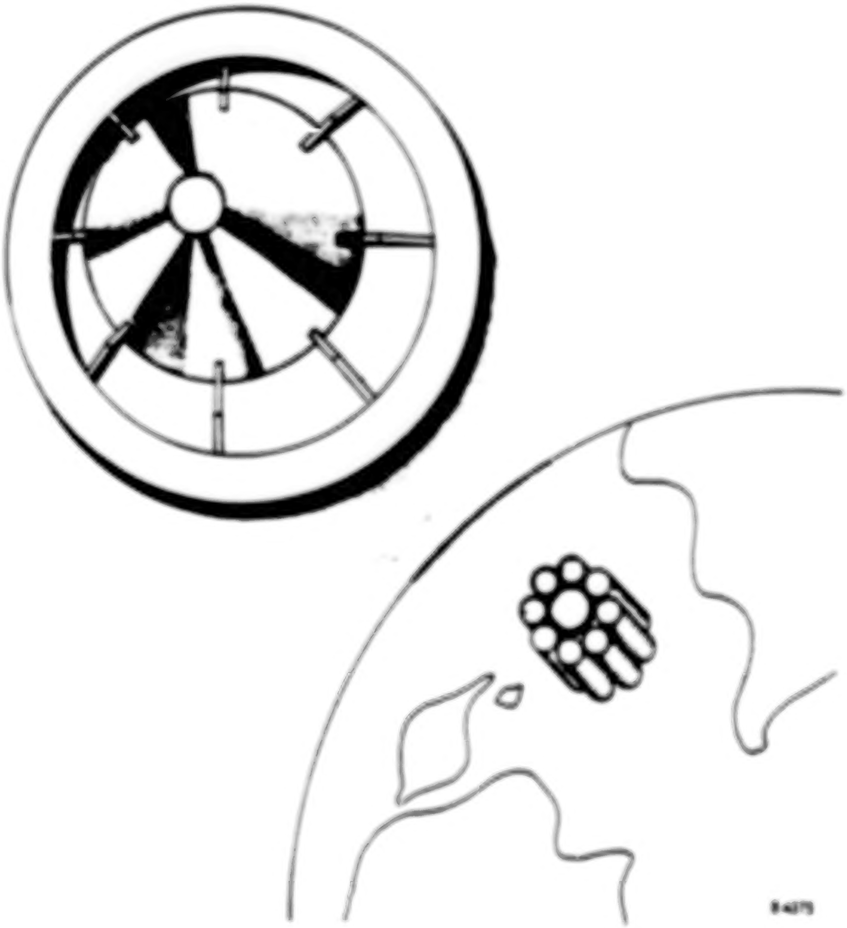

Figure 1. Laser propelled ground-to-orbit launch vehicle

Recent work has been conducted under the SDIO Laser Propulsion Program to demonstrate feasibility of the concept through fundamental research, with an eye toward future applications. In another symposium paper, Jordin Kare (ref. 1) reviews the status of pulsed laser propulsion and identifies future applications. In this paper, we focus on the current basic research which forms the foundation for future advances. We emphasize here a research effort oriented toward the earth to low-earth orbit problem which offers the most immediate challenge, and promises the most immediate rewards, but it should be kept in mind that simple and straightforward extensions of this effort can easily be brought to frustion in support of inter-orbit and inter-planetary mission opportunities.

Recent research in ground-to-orbit laser propulsion has emphasized laser supported detonation wave thrusters driven by repetitively pulsed infrared laser. The detonation wave thruster concept was introduced by Pirri and Weiss (ref. 2). The current work has centered on a double pulse version invented by Reilly (ref. 3), and reintroduced by Kantrowitz (ref. 4) in simplified form for use in ground-to-orbit launches of small payloads. In this propulsion concept each laser repetition cycle consists of two pulses. A lower energy first pulse is used to vaporize a small amount of solid propellant and then after a brief expansion period of a few microseconds, a second and higher energy laser pulse is used to drive a detonation wave through the expanded vapor. Temperatures of order 10,000 K are achieved. During a several millisecond intercycle delay, expansion of the hot vapor converts thermal energy to directed kinetic energy. High specific impulses of ~ 600 to 800s are achievable at energy conversion efficiencies of ~ 20 to 40 percent. Figure 2 illustrates the thruster concept during each of the two laser pulses of a thrust cycle.

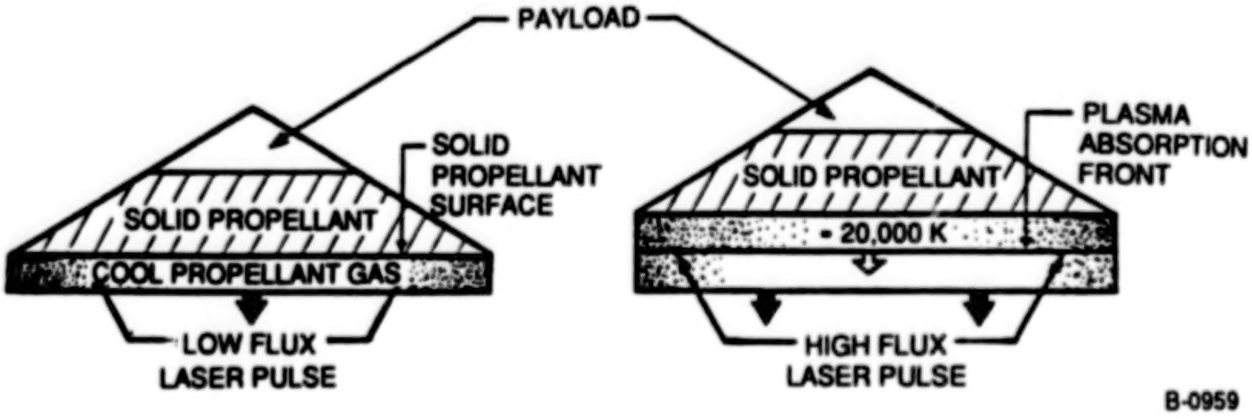

Figure 2. Laser-sustained-detonation wave rocket

The physics of such thrusters has been explored both theoretically and in the laboratory. We report here the results of numerical studies comparing the detonation wave properties of various candidate propellants, and the simulation of thruster performance under realistic conditions. Experimental measurements designed to test the theoretical predictions are also presented.

**Detonation wave properties**

In a detonation wave thruster, several requirements must be made on the propellant properties to insure proper operation. Among the important considerations are: moderate heat of vaporization, short optical absorption depth to the laser beam, low ionization threshold in the gas phase, and low average atomic weight. These requirements combine to ensure that there is a proper amount of mass delivered during laser irradiation, minimal mass flow between pulse pairs, and uniform processing of the gas during the high irradiance detonation wave portion of the thruster cycle. To achieve this last condition, it is necessary that a detonation wave can be formed quickly in the high irradiance portion of the thruster cycle, and that the wave must be thin compared to the gas slug which has been produced by the low irradiance portion of the cycle. Failure on the first point would result in considerable mass being ejected from the thruster at low specific impulse, while failure on the second point causes non-uniform processing of the gas, with consequent low energy conversion efficiency.

In order to address the differences between candidate propellants with respect to detonation wave properties, we have constructed a computer code which simulates such waves under assumed conditions of steady state and local thermodynamic equilibrium. The details of the code have been published elsewhere (ref. 5). For the present paper, we have chosen two specific cases to illustrate the utility of the method. Figures 3a and 3b show the density versus distance profiles of two

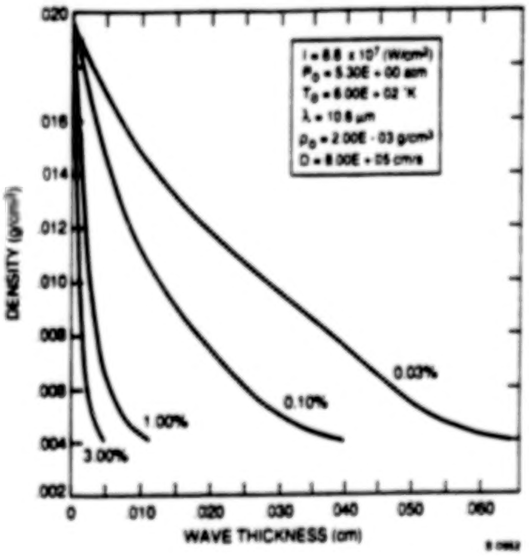

Figure 3a. Absorption wave for water propellant seeded with Li at indicated percentages, density versus position

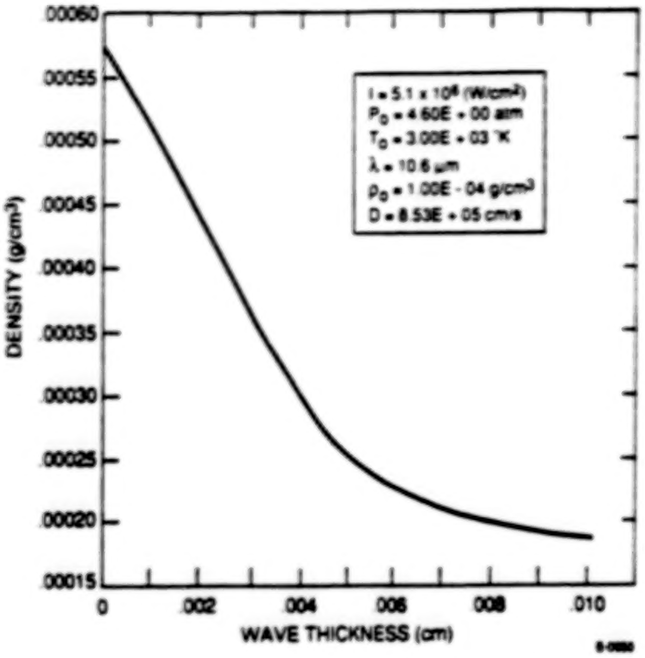

Figure 3b. Absorption wave for lithium hydride, density versus position

candidate propellants, water and lithium hydride, under conditions of comparable detonation wave velocity, which in turn implies comparable specific impulse. In these figures, the laser is incident from the left. Two factors are of particular importance in the comparison. The first is that at comparable specific impulse, considerably higher irradiance is required to operate using water (ice) as the propellant. This is because of the corresponding higher heats of vaporization, dissociation, and ionization of ice, as compared to lithium hydride. The second important feature is that in order to achieve the required thin detonation wave structure ($\lesssim$ 100 μm is desirable for a ~ 1 cm gas slug), it is necessary to add substantial atom fractions of a low ionization potential material to the water. On-going research in this area includes incorporation of non-equilibrium properties, exploration of other possible propellants, and assessment of the unsteady portions of the process.

### Reactive flow modeling

Once the detonation wave processing is complete, the hot gas expands away from the rocket. Both electron-ion and atom-atom recombination take place in this expansion phase of the thrust cycle. In order to adequately describe the non-equilibrium aspects of these recombination processes, we have constructed a hydrocode which includes finite rate kinetics for the important reactions. The details of this code have been presented elsewhere (ref. 6). For the present context, we have chosen to emphasize a dependence on parameter variations for a particular propellant.

A variety of different cases was simulated. To most clearly identify the effects of different parameters, a base case was chosen for which the effect of perturbations were examined. This base case was stoichiometrically $COH_2$, that is one mole of carbon to one mole of oxygen to two moles of hydrogen. A mixture of this molar concentration is produced by the breakdown of a polyacelal resin. A 2 cm thick gas slug with a density of $2 \times 10^{-3}$ grams/cm$^3$ and a temperature of 15,000 K was used, where the slug was initially at rest with respect to the base of the rocket and allowed to expand one dimensionally into vacuum. This case produced an impulse of 730s with a coupling of 36 dyne-s/Joule and an energy efficiency of 33 percent (neglecting radiation losses).

The results (see Table 1) clearly demonstrate the advantages of a dense gas slug over a more diffuse one. At low densities the recombination rate drops precipitously, substantially reducing the impulse. If a lower mass removal rate per pulse is needed, it is preferable to reduce the slug thickness rather than lower its density. If this drives one to an excessively thin slab which deposits its

Table 1. Gas expansion simulation results

| Fuel | Temperature | Mesh Points | Gas Slug (cm) | CO Recombination (%) | Density (g/cm$^3$) | Impulse (s) | Impulse/ Energy (dyne-s/ J) | Efficiency (%) |
|---|---|---|---|---|---|---|---|---|
| $COH_2$ | 15,000 | 40 | 2 | 47 | $2 \times 10^{-3}$ | 730 | 36 | 33 |
| $COH_2$ | 15,000 | 80 | 2 | 54 | $2 \times 10^{-3}$ | 760 | 38 | 36 |
| $COH_2$ | 15,000 | 40 | 0.2 | 33 | $2 \times 10^{-3}$ | 650 | 34 | 29 |
| $COH_2$ | 10,000 | 40 | 2 | $33^* \rightarrow 74$ | $2 \times 10^{-3}$ | 620 | 42 | 32 |
| $COH_2$ | 15,000 | 40 | 2 | 0 (forced) | $2 \times 10^{-3}$ | 550 | 27 | 18 |
| $COH_4$ | 15,000 | 40 | 2 | 60 | $2 \times 10^{-3}$ | 840 | 33 | 34 |
| $COH_2$ | 15,000 | 40 | 2 | 16 | $2 \times 10^{-4}$ | 640 | 32 | 25 |
| $COH_2$ | 15,000 | 40x30 | 2 x 50 | 44 | $2 \times 10^{-3}$ | 610 | 31 | 24 |

impulse over two brief an interval, then propellants for which recombination is unimportant should be considered. Going to lower temperatures improves the impulse per unit energy, but, as expected, the total impulse suffers.

On-going research seeks to model a wide variety of propellants and to incorporate the effects of specific projectile geometries.

### Experimental work

In conjunction with the theory and modeling activities, an experimental effort has been conducted to verify the validity of the conclusions. In particular, measurements have been made of the effective heat of mass removal, and the delivered impulse for various propellant materials, along with various optical and radiometric measurements. To illustrate some of the ongoing work, we offer an example from experiments designed to visualize the detonation wave as it processes a slab of vaporized propellant.

In order to visualize these phenomena in our experiments, we have used a technique to form temporally and spatially resolved images of the plasma radiance, and opacity to its own radiance. The technique is a two-dimensional, temporally resolved generalization of a method developed by Goncharov (ref. 7). The key to the technique involves positioning on the side of the plasma opposite the observation site an imaging lens and a concave reflecting mirror whose surface has been divided into a fine pattern of alternating reflective and non-reflective (opaque) strips. With appropriately chosen and positioned imaging lenses, the radiance distribution recorded by the observation camera consists of a striped pattern with the amplitude of adjacent stripes being proportional to, respectively, the local plasma radiance and the plasma radiance amplified by its self-transmission through the plasma. By comparing the relative intensities of adjacent stripes and knowing the effective transmission of the retro-reflecting optical components, one then finds the spatial profile of plasma transmission.

Detonation waves were observed in soda lime glass vapor (Figure 4). The figure consists, in principal, of eight exposures proceeding raster fashion from the lower left to the upper right. The first two frames precede the introduction of the high intensity laser pulse and are consequently black.

A lower intensity laser pulse was used about 6 μs before the main pulse to vaporize some of the glass target. The vapor from this pulse occupies a zone extending about a centimeter from the target surface at the time when the high intensity pulse is introduced. This second pulse

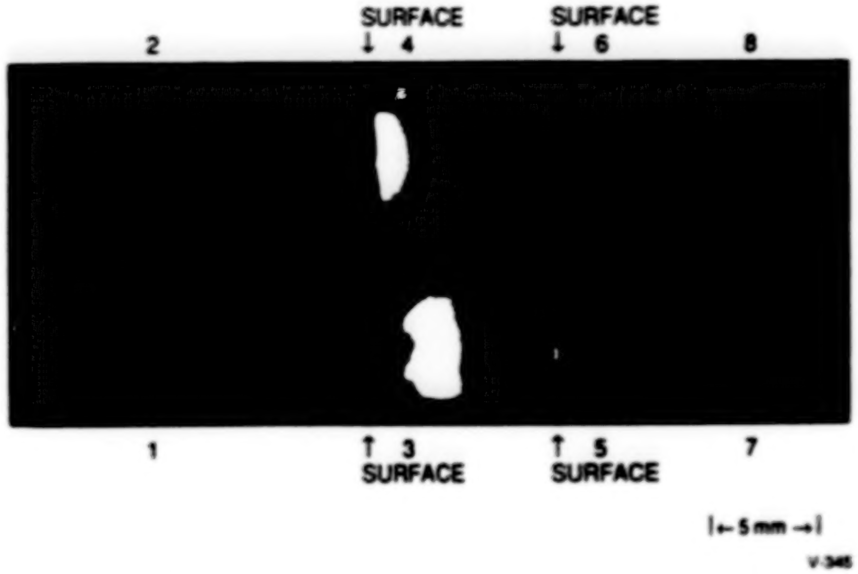

Figure 4. Self-transmission framing photo for double pulse experiments on glass

ignites a laser supported detonation wave wave and dries it at speeds approaching $10^6$ cm/s. The wave is clearly visible in the third frame at a position several millimeters off the target surface. In the intervening microsecond between the third and fourth frames, the wave would be expected to propagate to the edge of the initial pulse vapor distribution, where the rarefied gas density would no longer support an opaque wave. The fourth frame confirms this expectation, showing that laser light is once again being admitted to the surface. By the fifth frame, the TEA laser pulse has considerably weakened and is able to support only a comparatively tenuous plasma near the target surface. No exposure at all is detectable in the subsequent frames.

## Future Research

While the current paper has focused on some particular examples to demonstrate the character of current research, simple extension of the research will lend themselves to a wide variety of future laser propulsion applications. In particular, there may be advantages in exploring lower specific impulse propellants for a multistage ground-to-orbit concept, or to make use of radio frequency free-electron laser waveforms. High specific impulse systems and nozzled thrusters may lend themselves more to in space missions. In any event, transition from the current research phase to 21st century actual systems appears straightforward.

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# USE OF MAGNETIC SAILS FOR ADVANCED EXPLORATION MISSIONS

N91-22153

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## ABSTRACT

The magnetic sail, or magsail, is a field effect device which interacts with the ambient solar wind or interstellar medium over a considerable volume of space to generate drag and lift forces. Two theories describing the method of thrust generation are analyzed and data results presented. The techniques for maintaining superconductor temperatures in interplanetary space are analyzed and low risk options presented. Comparisons are presented showing mission performance differences between currently proposed spacecraft utilizing conventional chemical and electric propulsion systems, and a Magsail propelled Spacecraft capable of generating an average thrust of 250 Newtons at a radius of one A.U. The magsail also provides unique capabilities for interstellar missions, in that at relativistic speeds the magnetic field would ionize and deflect the interstellar medium producing a large drag force. This would make it an ideal brake for decelerating a spacecraft from relativistic speeds and then maneuvering within the target star system.

## INTRODUCTION

The magnetic sail, or Magsail, is a device which can be used to accelerate or decelerate a spacecraft by using a magnetic field to accelerate/deflect the plasma naturally found in the solar wind and interstellar medium. Its principle of operation is as follows: A loop of superconducting cable hundreds of kilometers in diameter is stored on a drum attached to a payload spacecraft. When the time comes for operation the cable is played out into space and a current is initiated in the loop. This current once initiated, will be maintained indefinitely in the superconductor without further power. The magnetic field created by the current will impart a hoop stress to the loop aiding the deployment and eventually forcing it to a rigid circular shape. The loop operates at low field strengths, typically $10^{-6}$ Tesla, so little structural strengthening is required.Two different configurations were examined as shown in figure 1. In the axial configuration (fig. 1a), the axis of the dipole is somewhat aligned with the direction of flight. In the normal configuration (fig. 1b) the axis of the dipole is normal (or perpendicular) to the direction of flight. Three previous papers by the same authors (references 1,2 &3) have discussed the Magsail's principles of operation and its applications for interstellar and planetary missions. This paper will show additional data generated this year and incorporate various technology advancements made since the last paper. A general description of the principles of operation follows:

The Magsail as currently conceived depends on operating the superconducting loop at high current densities at ambient temperatures. In interstellar space ambient is 2.7 degrees Kelvin where current low temperaturesuperconductors NbTi and $Nb_3Sn$ have critical currents of about $1.0\text{x}10^{10}$ and $2.0\text{x}10^{10}$ Amps/$m^2$ respectively. In interplanetary space, where ambient temperatures are above the critical temperatures of low temperature superconductors, these materials would require expensive refrigeration. However, the new high temperature ceramic superconductors such as $YBa_2Cu_3O_7$ have demonstrated enormous critical currents in samples at temperatures maintainable in interplanetary space using simple radiative thermal control concepts (ie. 70-90 degrees K). Assuming this performance will someday be available in bulk cable we have chosen to parameterize the problem by assuming a near term high temperature superconductor with a critical current of $10^{10}$ amp/$m^2$, and an advanced technology superconductor with a critical current of $10^{11}$ amps/$m^2$. Because the magnets are only operating in an ambient environment below their critical temperature no substrate material beyond that required for mechanical support was assumed. Assuming a fixed magnet density of 5000 kg/$m^3$(copper-oxide), our magnets have current to mass density ratios ($j/\rho$) of $2\text{x}10^6$ and $2\text{x}10^7$ amp-m/kg for the near term and advanced cases, respectively.

The equation for superconductor mass as a function of radius, peak field strength,and current density ratio was found to be:

$$M_{sc}=4(\pi/\mu o)B_m R_m^2/(j/\rho) \qquad (1)$$

where $\mu o$ is the permitivity of free space, $B_m$ = maximum field strength in center of loop, $R_m$ = loop radius and $j/\rho$= maximum allowable current density to mass ratio.

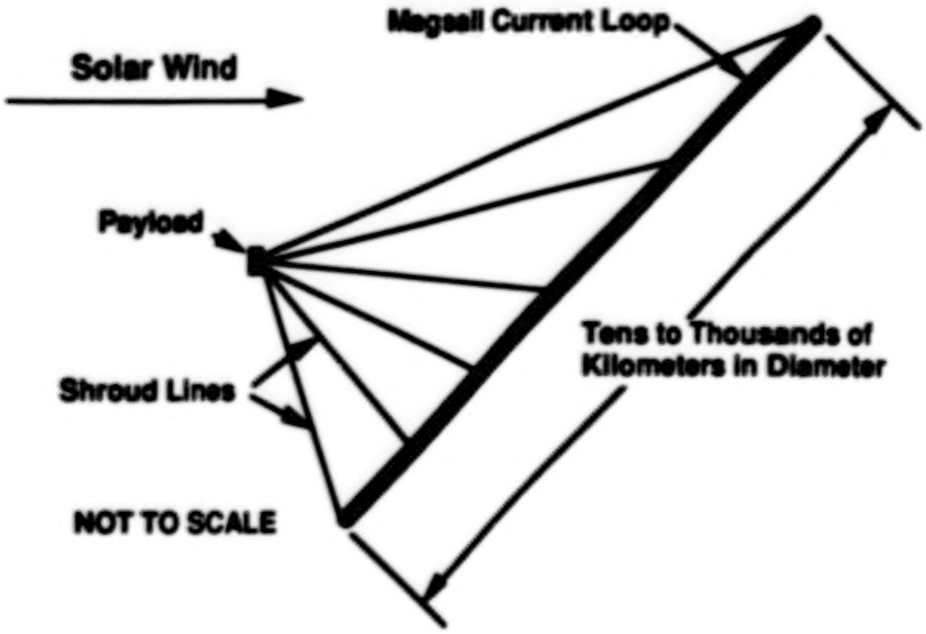

Magsail Current Loop
Solar Wind
Payload
Tens to Thousands of Kilometers in Diameter
Shroud Lines
NOT TO SCALE

**Figure 1a Axial Magsail Configuration**

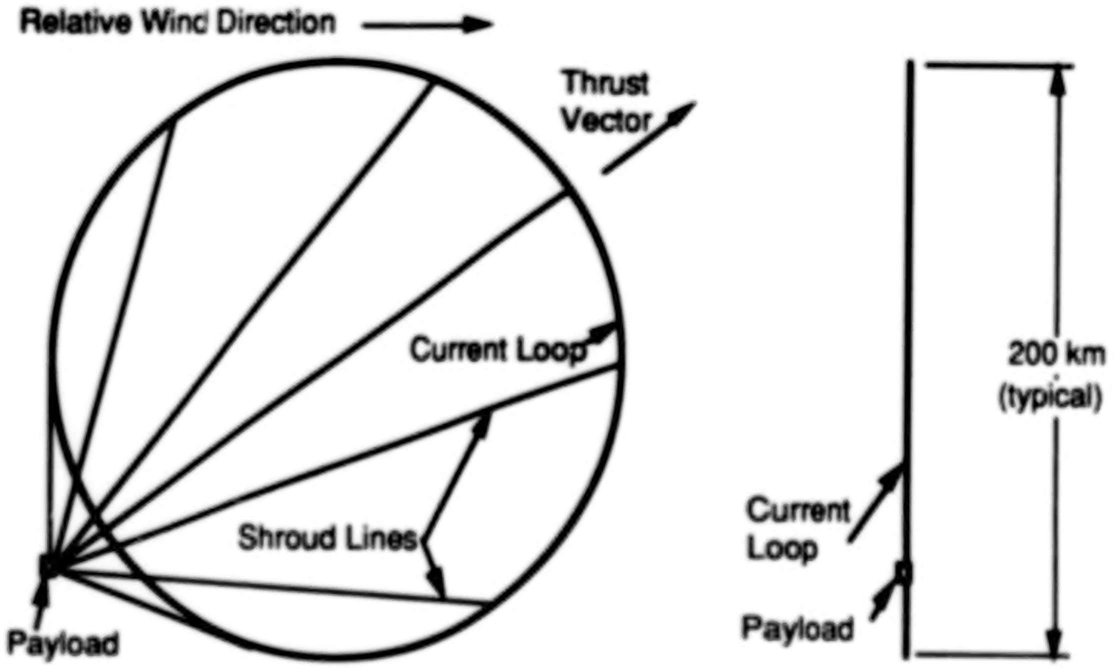

Relative Wind Direction
Thrust Vector
Current Loop
Shroud Lines
Payload
200 km (typical)
Current Loop
Payload

**Figure 1b. Normal Magsail Configuration**

In operation charged particles entering the field are deflected according to the B-field they experience, thus imparting momentum to the loop. If a net plasma wind, such as the solar wind, exsists relative to the spacecraft, the Magsail loop will always create drag, and thus accelerate the spacecraft in the direction of the relative wind. The solar wind in the vicinity of earth is a flux of several million protons and electrons per cubic meter at a velocity of 300 to 600 km/sec. This can be used to accelerate a spacecraft radially away from the sun and the maximum speed available would approximate that of the solar wind itself. While inadequate for interstellar missions these velocities are certainly more than adequate for interplanetary missions.

When the dipole field is inclined to the wind vector the magsail also generates a force perpendicular to the wind (i.e. lift). While not crucial for interstellar applications, lift greatly enhances the usefulness of the Magsail for interplanetary operations. Additional interplanetary maneuvering capability could be attained by using gravitational swingbys of the major planets. The second application, and the one which will receive the majority of our attention in this paper, is as a brake for an interstellar spacecraft travelling at fractions of the speed of light. The rapidly moving magnetic field of the Magsail ionizes the interstellar medium and then deflects the resulting plasma, creating drag which decelerates the spacecraft. The ability to slow down spacecraft from interstellar to interplanetary velocities without the expenditure of rocket propellant results in a dramatic lowering of the total mission mass, as we shall show in the detailed systems performance trades presented below.

There are two ways in which the Magsail physically interacts with the surrounding plasma. At very low particle densities we would expect a particle interaction, where each particle interacts separately with the field of the magnetic dipole. In this case, we calculate the lift and drag generated by the current loop by summing the momentum changes over all incoming particles.

For the particle flows characteristic of the solar wind in Earth's vicinity and for the large current loops where the dipole magnetic field dominates the local magnetic field beyond several proton gyration radii, we would expect a fluid interaction, where the solar wind interacts with the Magsail field as a magnetohydrodynamic fluid, in the same manner as it interacts with the earths' magnetic field. In this case, we can calculate Magsail lift and drag using aerodynamic approximations developed to explain the shape of earth's magnetosphere. We will discuss the mechanism for thrust generation and its impact on interplanetary orbital mechanics only briefly in this paper. For details on the orbital trajectories of a Magsail propelled vehicle using with the solar wind, see references 2 &3.

## Particle Interaction

For the particle interaction, a computer code, TRACE, was written which follows the trajectory of individual charged particles as they interact with the magnetic field generated by the current loop, and a series of computer experiments were conducted testing the final disposition of particles fired into the magnetic field with various wind velocities and starting positions. A random thermal velocity perpendicular to the wind velocity was included to accurately model proton reflection characteristics. Summing the momentum changes of individual protons as they traversed various regions of the magnetic field allows the total impact of the field on the oncoming solar wind to be calculated. The total changes in momentum would be experienced as drag and lift on the current loop.

As a test case, we have integrated the change in proton momentums in the radial and orbital directions for a example magsail loop in the normal configuration (fig. 1b) with a Magnetic Dipole Moment ($M_{DM}$) of $1.63 \times 10^{15}$ amp-turns-m$^2$. The initial integration was over a square intercept area 40 loop radii on a side using the TRACE program. This was then extended to an area more than 200 loop radii on a side using statistical sampling and emperical equations derived through curve fitting, and showed that the average change in radial momentum summed over the entire area, $A_o$, ($A_o = 10^{14} m^2$) was:

$$A_o \frac{\Delta V_r}{V_{ro}} = -0.00237 \, A_o \tag{2}$$

Therefore the radial force, $F_r$, can be represented as:

$$F_r = \rho_o V_{ro}^2 A_o \frac{\Delta V_r}{V_{ro}} \tag{3}$$

Which becomes $F_r = 2q\,A_0 \frac{\Delta V_r}{V_{ro}} = 264.5$ Newtons for a quiet sun solar wind at one AU. For this study, we have assumed that a quiet sun solar wind at one AU has $q = 5\text{x}10^{-10}$ N/m$^2$( $q = \frac{1}{2}\rho_0 V_{ro}^2$ ), that an average solar wind has $q = 1.0\text{x}10^{-9}$ N/m$^2$, and that a high solar wind has $q = 2.0\text{x}10^{-9}$ N/m$^2$. These are approximations based on published Mariner 2 data.

The average change in velocity in the orbital plane(zero average particle velocity before encountering the magnetic field) summed over the same area is:

$$A_0 \frac{\Delta V_z}{V_{ro}} = +72.4\ R_m^2 \qquad (4)$$

Therefore, the minimum tangential force ,$F_t$, can represented as:

$$F_t = 2q\,A_0 \frac{\Delta V_z}{V_{ro}} = 74.1 \text{ Newtons} \qquad (5)$$

and the Magsail produces a lift to drag ratio of 0.28. The generation of significant lift is an important result, and as we show below , is in contrast to the fluid interaction case.

## Fluid Interaction

In the fluid interaction case, the forces on the current loop are modeled from the measured interaction of the solar wind with the earth's magnetic field, with corrections for scale based on known physical principles.

The solar wind is a continuous hydrodynamic expansion of the solar corona out through the solar system. The nature of this expansion is such that the plasma achieves very high velocities ($V_0$ ~ 500 km/sec) within the first one tenth astronomical unit and then the velocity increases very slowly with radius (ie. $V \sim V_0(\ln R)^{1/2}$), where R is the radius from the center of the sun (reference 4). The hot coronal plasma had very high electrical conductivity and, as such a fluid expands, the magnetic field lines are "frozen-in". This "frozen-in" magnetic field causes the solar wind to behave as a fluid even when its density is so low that the mean free length between collisions is several Astronomical Units!

The fluid interaction process between the solar wind and Earths magnetic field is shown schematically in figure 2 from reference 5.

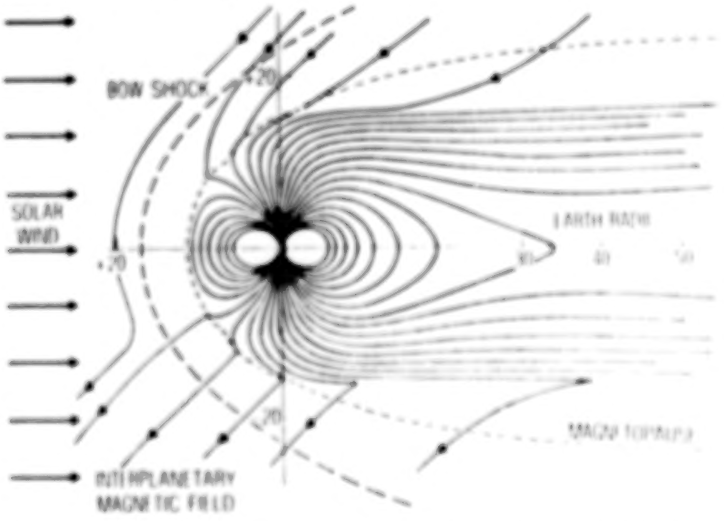


**Figure 2 Schematic of the Earth's Magnetosphere**

The magnetosphere represents the boundary between the perfectly conducting plasma in the solar wind and the compressed dipole magnetic field. The pressure, p, on a unit surface area of the magnetosphere is determined by the balance between the dynamic pressure from Newtonian flow aerodynamics, and the magnetic pressure from the magnetic field. This can be repesented as:

$$2\left(\frac{1}{2}\rho_0 V_0^2 \cos^2\theta\right) = \frac{B^2}{2\mu_0} \qquad (6)$$

where $\theta$ is the angle between the solar wind vector and the normal to the unit surface area of the magnetosphere. Note that $\mu_0$ is the vacuum magnetic permitivity described earlier ($\mu_0 = 1.26\text{x}10^{-6}$ henry/meter). At this point the solar wind has lost its velocity normal to the field lines and only flows tangentially. In front of this region a bow shock forms similar to the shock wave in front of a blunt body in hypersonic flow. The region between the bow shock and the magnetosphere is called the magnetosheath. The magnetosheath is made up of higher density, thermalized solar wind, which is deflected and accelerated as it flows around the magnetopause region (again very similar to hypersonic flow). Assuming the solar wind excludes the magnetic field by generating surface currents to create an equal and opposite magnetic field at the boundary, we get a doubling of the magnetic field inside the boundary from symmetry (reference 2). The magnetic field at any point for a dipole in a vacuum can be written as:

$$B = \frac{\mu_0 M_{DM}}{4\pi r^3}\left(2\cos\phi\,\mathbf{r} + \sin\phi\,\boldsymbol{\phi}\right) \qquad (7)$$

where $M_{DM}$, the magnetic dipole moment, is equal to the number of amp-turns of current times the area of the current loop, r is the radius from the center of the current loop to the point in question, $\phi$ is the angle between r and the dipole axis. $\mathbf{r}$ and $\boldsymbol{\phi}$ are the corresponding spherical coordinate vectors. Since the field strength at the magnetosphere boundary is twice that at the same point in a vacuum, we can write an equality for the pressure at any point on the boundary surface:

$$2q\cos^2\theta = \frac{B^2}{2\mu_0} = \frac{(2\mu_0 M_{DM})^2}{(4\pi)^2}\frac{1}{2\mu_0}\frac{1}{r^6}(1+3\cos^2\phi) \qquad (8)$$

which can be written:

$$\cos\theta = \left(\frac{M}{Q}\frac{G}{r^6}\right)^{0.5} \qquad (9)$$

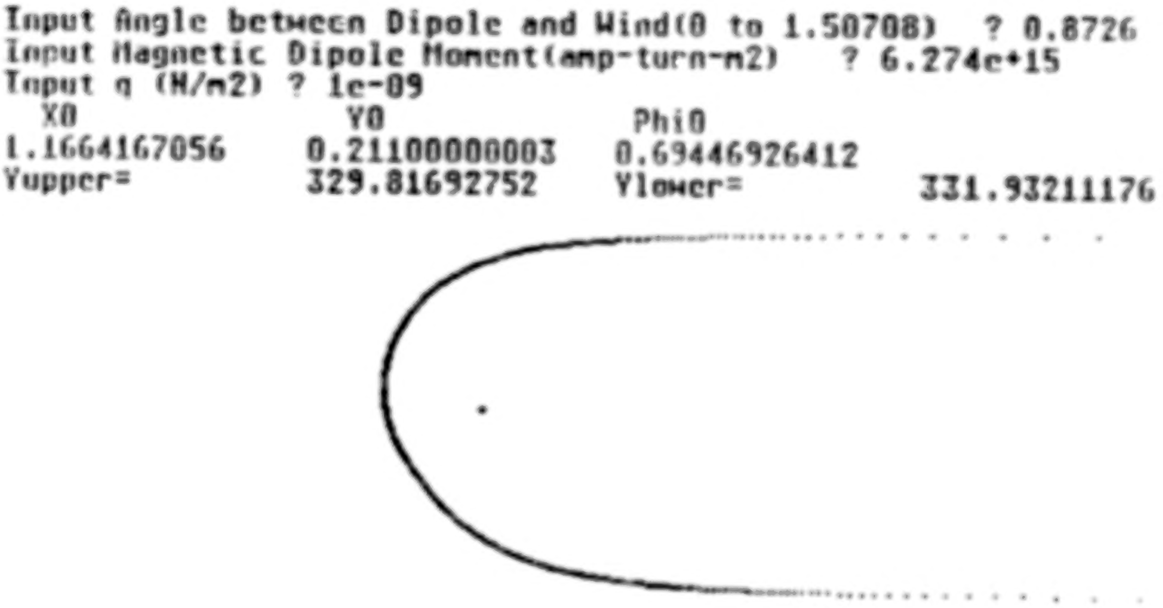
where $M = \frac{\mu_0 M_{DM}^2}{16\pi^2}$ and, $G = 1+3\cos^2\phi$

For calculating purposes, we define a new variable: $R_{MQ} = \left(\frac{M}{Q}\right)^{0.16667}$, so that:

$$r = R_{MQ}\left(G\cos^2\theta\right)^{0.16667} \qquad (10)$$

Note, that $R_{MQ}$ is the characteristic standoff distance between the loop center and the front surface of the magnetosphere when $\phi = 90$ degrees ( dipole axis is normal to the flow).

A computer program , MAGSHAPE, has been written to numerically integrate Equation (10) in order to determine the shape of the magnetosphere. A typical result is shown in figure 3. A variant of MAGSHAPE, called MAGFORCE, which integrates the pressures over the entire magnetosphere body of revolution has recently been completed. This enabled us to calculate the magsail drag, lift, and moment coefficients based on equivalent Newtonian flow aerodynamics. The results, sumarized in figure 4, are somewhat disappointing because the maximum magsail L/D is so low (about 0.05), but this L/D is more than adequate for interstellar applications, and precludes very few interplanetary applications. The only effect seen in Mars mission simulations using the reduced L/D was an increase in flight times of around ten percent. The full impact of magsail aerodynamics on interplanetary missions will be explored in a future paper ( AIAA Paper 90-2367 "Progress in Magnetic Sails").



```
Input Angle between Dipole and Wind(0 to 1.50708)   ? 0.8726
Input Magnetic Dipole Moment(amp-turn-m2)    ? 6.274e+15
Input q (N/m2) ? 1e-09
  X0                  Y0                Phi0
1.1664167056     0.21100000003    0.69446926412
Yupper=          329.81692752     Ylower=           331.93211176
```

Figure 3. Integrated Magnetosphere Shape

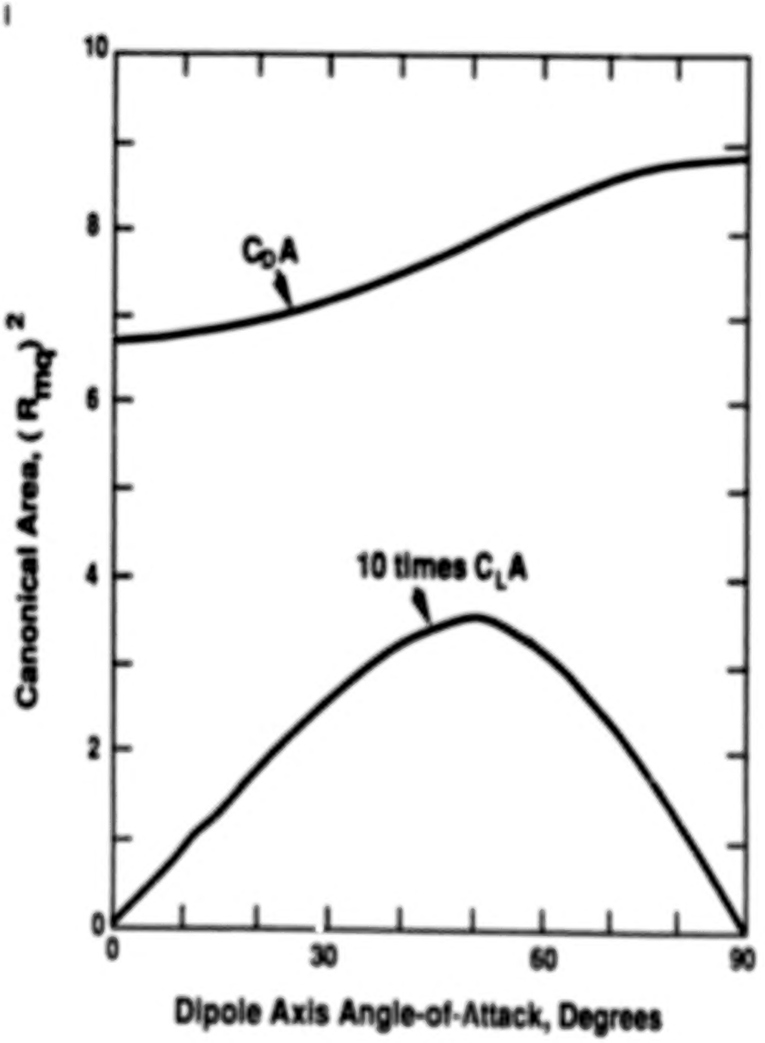


Figure 4. Magsail Aerodynamics

The aerodynamic data was used to generate parametric "bare magsail" performance at one AU solar radius with a "average" solar wind ,( $q = 1.0\times10^{-9}$ N/m$^2$ ). This data, included as figure 5, shows that the magsail scales favorably with size. In effect, the drag is increasing with current loop area while the mass increases with loop perimeter. The "bare" magsail for each dipole moment was sized for minimum mass including the superconductor, structural reinforcement ( if required), and the Thermal Protection System (TPS). When this data was crossplotted it was discovered that every minimum mass loop had 57,700 amps of current circulating and all had a hoop force of 720 Newtons. These optimums are determined by the $j/\rho$ and lineal TPS masses assumed. Improvements in technology, or limits on hoop force to meet acceleration requirements, will shift the optimums, but a minimum mass magsail will have the same current and same hoop force regardless of size (dipole moment). This means only one type of magsail cable must be developed and different lengths are used for different applications. This should provide significant cost savings, design flexibility, and mission safety.

Note: $j/_{ro}$ = 2,000,000 amp-m/kg & Lineal Mass of TPS = 0.0288 kg/m

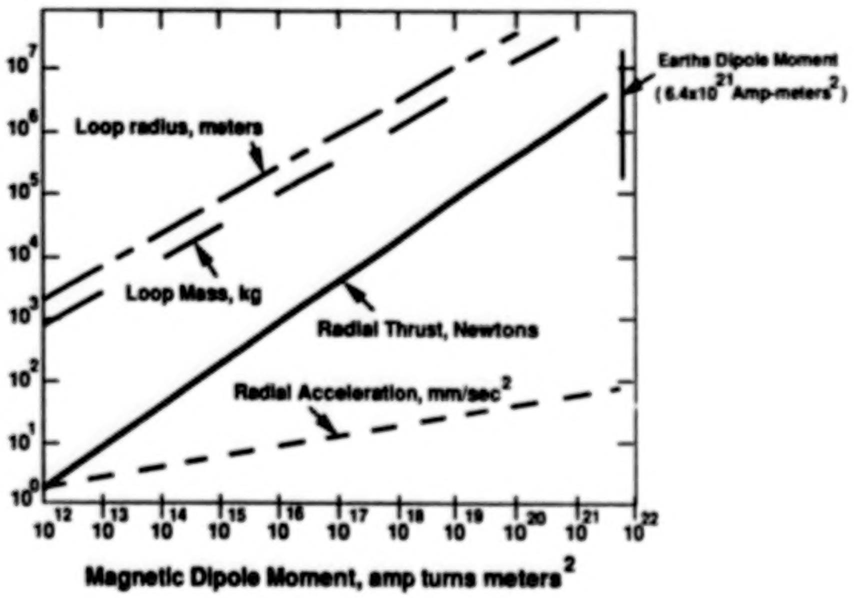


Figure 5. Bare Magsail Performance

## Interstellar Magsail Missions

This paper was aimed at the interstellar applications of magsails, which are further in the future, but a better "fit" with magsail capabilities.The use of magsails to slow a laser lightsail propelled interstellar exploration vehicle was introduced in reference 1. In that paper, the limits to laser lightsails were discussed and some example missions were presented. As a point of reference, the trip time for a one-way 1000 ton manned exploration mission to a star ten lightyears away was 107 years assuming a 1000 Terawatt laser with a beam divergence angle of $1.0\times10^{-10}$ radians. The vehicle dry mass was 3035 tons, of which 1000 tons is payload, 1156 tons is a metalized kapton lightsail, 687 tons is magsail, and 193 tons is a fusion pulse rocket and propellant to reduce the time spent in the doldrums. ( The doldrums are the regions outside the boundary which separates the target star's solar wind from the interstellar medium.) The magsail is travelling between 3000 and 500 km/sec in this region and because the mass flow is so small, it takes several years elapsed time and almost a light year of distance to decelerate between these velocities. Adding a small rocket reduces the total deceleration period by four to five years.

The relatively long trip times are due to low acceleration and deceleration rates caused by a variety of physical limitations. Initial lightsail acceleration is limited by temperature constraints on the metal lightsail material, and the duration of acceleration is limited by the focusing capability of the laser optics ( at distances approaching one lightyear the diameter of the image of the laser power-limited aperture greatly exceeds the diameter of the lightsail). Initial deceleration of the magsail was excellent, but a disportionately long time was spent decelerating between 3000 and 500 km/sec.

Solutions to these problems seem to be at hand. The acceleration issues have been solved by Geoffrey Landis who introduced the subject of dielectric lightsail materials in reference 6 (with help from Robert Forward). A laser lightsail made from pure dielectric materials, operating at the proper wavelength, will absorb very little laser light and can operate at intensities many orders of magnitude above metal reflectors. This means laser power can be increased, acceleration raised, and acceleration distance shortened. Shortening the acceleration distance means laser beam imaging and pointing accuracy can cease to be a problem. The principal remaining acceleration problem becomes where to build a multi-thousand Terawatt laser. A candidate laser power station is shown in figure6. It uses a small asteroid to stabilize the focusing mirror during operation.

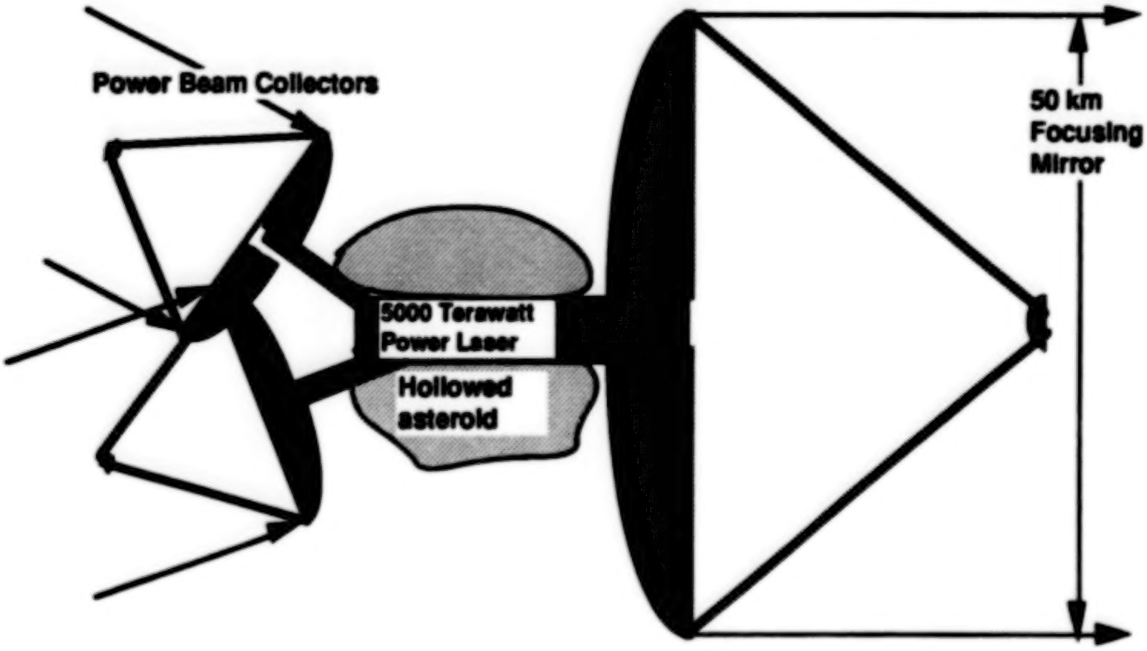


Figure 6. Possible Solar System Laser Power Station

The deceleration issues were the low drag to mass ratios of the initial magsail designs and the slow trip through the"doldrums". The current magsail designs have much better drag to mass ratios because they have bigger radii. This is the area-to-perimeter issue discussed earlier. By minimizing the design loop current to just barely sustain the hoop force required to counter the deceleration forces fed through the shrouds, a very large but very light magsail, is possible. The magsail proposed now is 3100 km in diameter versus 1000 km in the 1988 study and masses about 1000 tons. However, it provides deceleration times roughly half of those in the 1988 study. This magsail has a Magnetic Dipole Moment of $1.0\times10^{19}$ amp-$m^2$, with a current of 1,350,000 amps, and a hoop force of 390,000 newtons.

## Predicted Performance (1990 Edition)

One-way trip time for the same 1000 ton manned exploration mission descibed in reference 1 is now 37 years, of which 0.8 years is spent accelerating, 17.4 years is spent coasting at half the speed of light, and 18.8 years is spent decelerating. The initial vehicle masses 2344 tons, of which 1000 tons is payload, 950 tons is magsail, and 394 tons is lightsail. The vehicle is propelled by a 5000 Terawatt laser and reaches half the speed of light in 0.21 lightyears. The laser focusing mirror has a 50 km aperture and the lighsail is 50 km in diameter.

An alternative design carries a 201 ton fusion rocket and 159 tons of propellant. It has an initial mass of 2780 tons, requires a slightly heavier magsail (1028 tons) because it has to decelerate the fusion rocket also, but uses the same lightsail. Because of the higher initial mass it takes 0.32 years to accelerate,. It coasts for 17.8 years , and then decelerates for 14.5 years. The net result is a total savings of 4.4 years. Provided advanced fusion rocket technology is available, the time savings is probably worth the additional cost.

## CONCLUSIONS

Advancements in technology have increased the probability and the usefulness of the magsail. Using current technology, it can compete with advanced rocket systems to deliver people and cargo to Mars. Using straightforward extrapolations of todays technolgies, it can deliver manned vehicles to nearby stars within the time constraints of a single human lifetime.

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Vision-21, Space Travel for the Next Millenium
NASA Lewis Research Center, Cleveland, OH, April 3-4, 1990

# COMPARISON OF SUPER-HIGH-ENERGY-PROPULSION-SYSTEMS BASED ON METALLIC HYDROGEN PROPELLANT FOR ES TO LEO SPACE TRANSPORTATION

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## ABSTRACT

This paper investigates the application of metallic hydrogen as rocket propellant, which contains a specific energy of about 52 kcal/g in theory yielding a maximum specific impulse of 1700 s. With the convincing advantage of having a density fourteen times that of conventional liquid hydrogen/liquid oxygen propellants, metallic hydrogen could satisfy the demands of advanced launch vehicle propulsion for the next millenium.

Provided, that there is an atomic metallic state of hydrogen, and that this state will be metastable at ambient pressure, which is still not proven, the present publication shows the results of the investigation of some important problem areas, which concern the production of metallic hydrogen, the combustion, chamber cooling and storage.

The results show, that the use of metallic hydrogen as rocket propellant could lead to revolutionary changes in space vehicle philosophy towards small size, small weight and high performance SSTO systems. The use of high metallic hydrogen mass fractions results in a dramatic reduction of required propellant volume, while gas temperatures in the combustion chamber exceed 5000 K. Furthermore it follows from this study, that hydrogen (liquid or slush) is the most favourable candidate as working fluid. However, jet generated noise has to come into intense consideration, due to the very high exhaust velocities, which are possible with metallic hydrogen propellant.

### Symbols and abbreviations:

Ce : Exhaust velocity
DV : Velocity increment
Hmet : Metallic hydrogen
LHmet : Liquid metallic hydrogen
SHmet : Solid metallic hydrogen
MA : Methyl alcohol
M0 : Overall launch mass
M1 : Payload mass
M8 : Propellant mass
Mn : Vehicle dry mass
LH2 : Liquid hydrogen
SLH2 : Slush hydrogen
SH2 : Solid hydrogen
RP1 : Rocket propellant number one

## CONTENT:

1. Introduction
2. Metallic hydrogen history
3. Main research efforts
3.1 Choice of propellant combinations
3.2 Thrust chamber performance parameters
3.3 Thrust chamber cooling
3.4 Storage concepts
3.5 Propulsion concepts
3.6 Propulsion performance
3.7 Environmental aspects
3.8 Reflections on costs
4. Summary

## 1. INTRODUCTION

During the last thirty years space technology became a more and more economic factor in many areas. In particular, in the fields of ES-LEO transportation systems but also in many fields of satellites applications the aspect of competitiveness gained in significance. Against the background of the expected increasing space activities in number during the next decades, assisted by the establishment of growing space stations as well as by expanded missions to other celestial bodies up to its colonization (moon and mars), the following aspects concerning the realization of those space projects should be regarded with priority:

a)Reduction of the specific transportation costs ($/kg-payload) by the factor 10 from today's level (about 25000 to 40000 $/kg)
b)Increase of space transportation capacities
c)Increase of space transportation vehicle performance
d)Reduction of negative implications to the environment (noise power, exhaust gas reactions with ambient air, required propellant mass per kg payload, etc.)
e)Increase of reliability

In view of these demands, the propulsion system represents the decisive influencing component of launchers. The performance of a rocket engine itself will be mainly influenced from the quality of the used propellant combination, which is characterized primarily by the specific impulse.

| PROPELLANT COMBINATION | Isp (s) | DV/Ce | M1/M0 | $/kg-M1 | M0/M1 |
|---|---|---|---|---|---|
| LOW ENERGETIC | <280 | HIGH | VERY LOW | HIGH | HIGH |
| MEDIUM ENERG | 280-330 | MEDIUM | LOW | MEDIUM | MEDIUM |
| HIGH ENERGETIC | 330-500 | LOW | MEDIUM | LOW | LOW |
| SUPER HIGH ENERGETIC | >500 | VERY LOW | HIGH | VERY LOW | VERY LOW |

Isp: Specific impulse
DV/Ce: Propellant performance parameter
M1/M0: Payload mass ratio
$/kg-M1: Specific transportation cost
M0/M1: Specific propellant consumption

**Tab.1-1: General correlation between propellant performance and main launch vehicle parameters**

In Tab.1-1 the effects of the realizable specific impulse on the characteristic performance parameters of launch systems is shown. A diminishing value of the engine performance parameter DV/Ce (corresponding to challenge c) as well as increasing lightweight construction capability lead to increasing payload ratios (M1/M0) and therewith to increasing space transportation capacities (corresponding to challenge b). Increased payload mass ratios could lead to reduced specific transportation costs (corresponding to challenge a).

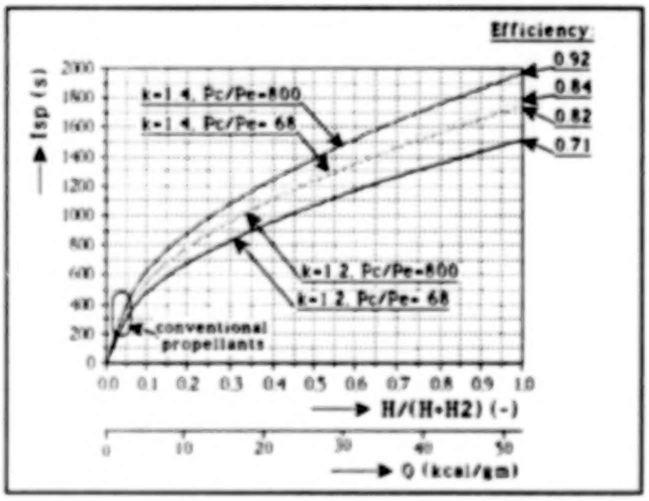

**Fig. 1-1: Influence of weight fraction of free hydrogen radicals in the propellant, H/(H+H2), respectively the specific energy, Q, on the specific impulse, Isp, for various efficiency factors; k: ratio of specific heats, Pc: chamber pressure, Pe: nozzle exit pressure**

As may be seen below, the use of super high energy propellants could reduce the propellant consumption considerably, in the course of which metallic hydrogen represents the most promising candidate. The interest of this study into metallic hydrogen as rocket propellant is based on the very high energy content, which is equivalent to high theoretical performance, which in turn is indicated by the specific impulse.

The specific impulse formula used, is as function of the overall net energy release per unit mass of propellant and is derived from the equation of the ideal exhaust velocity of a gas after thermodynamic expansion. Introducing an overall efficiency factor for the energy conversion (as a function of the ratio of specific heats, k, and pressure ratio, Pc/Pe ) lead to:

$$Isp=(2\cdot h\cdot Q)^{0.5}/g \text{ (s)}$$

where

h: Efficiency factor for energy conversion $=1-(Pc/Pe)^{(k-1)/k}$
Q: Specific energy [J/kg] (= 4.184E6·Q[kcal/gm])
g: Gravitational constant (9.81 $m/s^2$)

Using molecular hydrogen as working fluid, the energy release of the propellant can be calculated by multiplying the specific energy of metallic hydrogen by the weight fraction of the energized spezies in the propellant. For this case, complete free radical reactions and complete quenching of the metastable species to the ground state will be assumed.

In Fig. 1-1 the values for the specific impulse are plotted over the weight fraction of free hydrogen radicals in the propellant and the specific energy, considering various efficiency factors. In the following chapters, the results of the investigation of some identificated problem areas concerning the application of metallic hydrogen as rocket propellant will be given, on condition, that atomic metallic hydrogen will exist and be metastable.

## 2. METALLIC HYDROGEN HISTORY

The interest in metallic hydrogen as rocket propellant results from the very high bonding energy between the atoms of the hydrogen molecule, combined with a very high density (probably 1150 kg/m3). The amount of energy which is necessary for the production of Hmet, that means the homolytic dissociation of molecular hydrogen into the metallic atomic state ($H_2$ <--> 2H), could be induced by pressure energy.

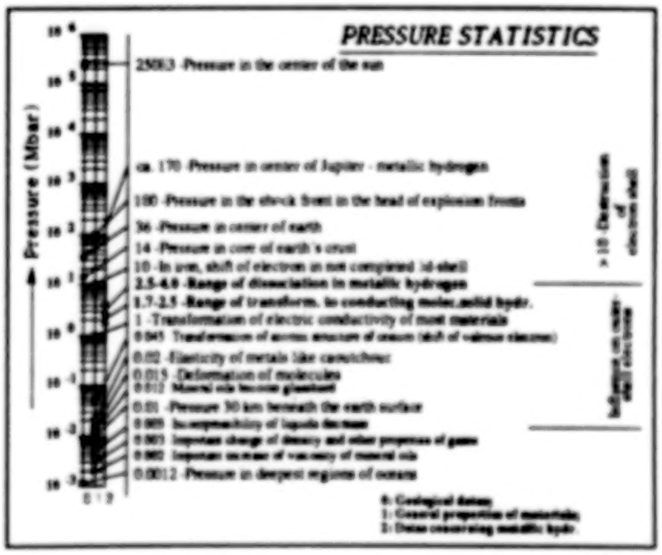

**Fig.2-1: Pressure impacts**

The corresponding development of the research on metallic hydrogen was not only aimed at the possible use as rocket propellant but also due to using the supposed superconducting characteristics. Fig.2-2 shows in a selfexplanatory way the essential acknowlegdments of the metallic hydrogen research efforts.

There are two different procedures showing the growing stage of knowledge, first on an experimental basis, and second on theoretical investigations. As may be seen, the main research

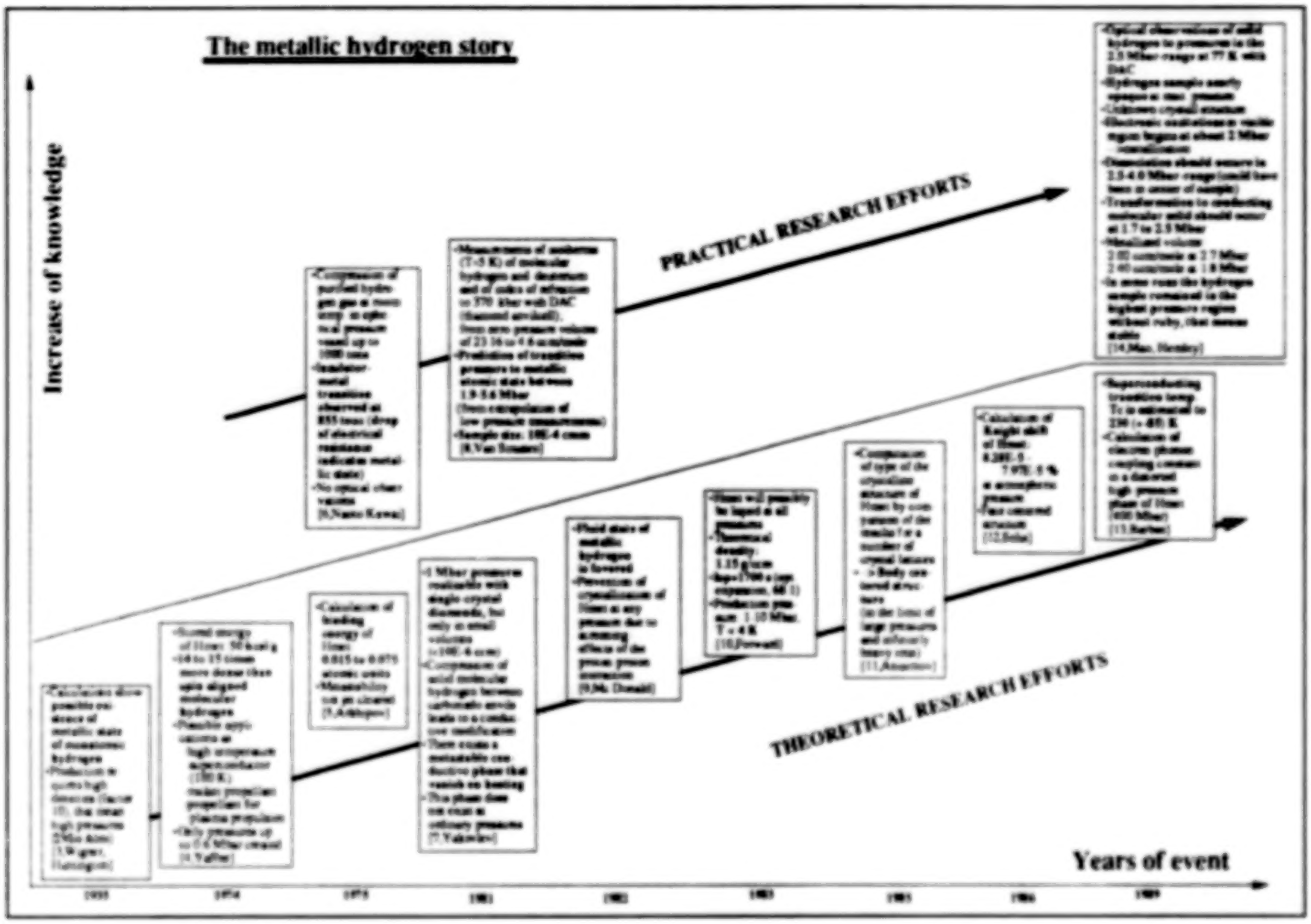
The metallic hydrogen story
Increase of knowledge
PRACTICAL RESEARCH EFFORTS
THEORETICAL RESEARCH EFFORTS
Years of event

Fig.2-1: Development of research efforts in the field of metallic hydrogen over the years

constraints concerned proving the existence of Hmet. The experimental investigations are very closely connected to high pressure research. To give an impression of the very high pressures of 2.5 to 4 Mbar, which will be required for Hmet production, Fig.2-1 illustrates different impacts due to the appearance of pressure.

The main properties of Hmet are summarized on Tab.2-1. For this study only those properties of metallic hydrogen are of interest, which are of importance for use as rocket propellant.

| | | | |
|---|---|---|---|
| State of aggregation | liquid [9,10] | solid [13,14] | |
| Stored energy (100%) | 52.1 kcal/g [4] | | |
| Density | 1.15 g/cm3 [10] | | |
| Storage temperature | 5 K [8] | 77 K [14] | room temp. [8] |
| Storage pressure | 2.5-4 Mbar [14] | ambient pressure | |
| Specific impulse | 1670 s [illegible] | | |
| Lattice structure | hcp [11] | fcc [12] | |
| Knight shift | 0.280.5 - 7.97(-5) % [12] | | |
| Superconducting transition temperature | 230 - 45 K [15] | | |
| Optical property | opaque [10] | not opaque [14] | |
| Dissociation pressure | 2.5-4 Mbar [14] | | |
| Transformation pressure to conducting molecular solid | 1.7-2.5 Mbar [14] | | |
| Metastability | yes [10] | no [7] | |

Tab.2-1: Properties of metallic hydrogen; properties important for this study are marked

## 3. MAIN RESEARCH EFFORTS

The present investigations will give a general idea of metallic hydrogen propellant application. Due to the wide range of investigation areas, only those will be presented, which have been identified as the most interesting ones. Fig. 3-1 illustrates the main research efforts.

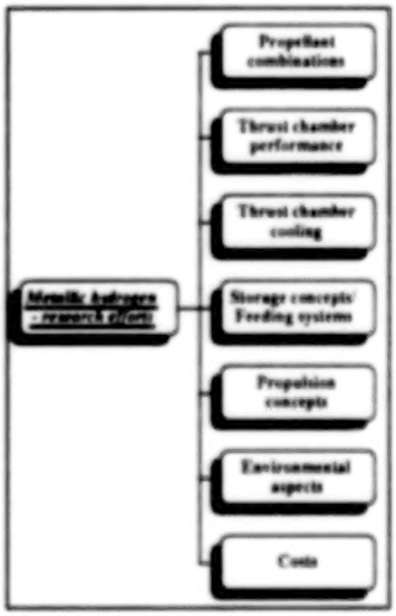
Metallic hydrogen - research efforts
Propellant combinations
Thrust chamber performance
Thrust chamber cooling
Storage concepts/ Feeding systems
Propulsion concepts
Environmental aspects
Costs

Fig. 3-1: Main research efforts

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## 3.1 CHOICE OF PROPELLANT COMBINATIONS

Due to the very high energetics of metallic hydrogen, yielding very high thrust chamber gas temperatures (as may be seen later), a secondary propellant component is required, which absorbs the heat of reaction and serves as expansion medium. The choice of the right working fluid should be done against the background of the general known propellant features.
Besides the combination of Hmet/LH2 as presented in the introduction, the following investigations include also more dense elements like slush hydrogen SLH2 and solid hydrogen SH2 which are cryogenic fuels, but also storable ones like the conventional used rocket propellant RP1 and another candidate, namely methyl alcohol, which offers higher performances compared to RP1. Tab. 3-1 lists the basic data of the investigated reactants fuels.

| Chemical | Formula | Enthalpy (cal/mole) | Phase | Temp.(K) | Density (g/cm³) | Designation |
|---|---|---|---|---|---|---|
| Hydrogen | H2 | -2154 | liquid | 20,27 | 0,0709 | LH2 |
| RP-1 | C1H1,9423 | -4500 | liquid | 298,15 | 0,773 | RP1 |
| Methyl alcohol | C1H4 | -57040 | liquid | 298,15 | 0,7866 | MA |

Tab.3-1: Reactants fuels data

The choice of the propellant combinations determines the tankage concept. For lack of data about metallic hydrogen properties (will it be a liquid or a solid?, storage temperature?, etc. as explained in chapter 3) various possible propellant combinations alternatives have been regarded. They are listed in Tab.3-2 (where -P stands for powder, -G stands for grain). The marked propellant combinations (PC-x) will be analysed below.

| | LHmet | LH2 | RP1 | MA | SLH2 | SH2 | SHmet-P | SHmet-G |
|---|---|---|---|---|---|---|---|---|
| LHmet | | PC1 | PC2 | PC3 | PC4 | | | |
| LH2 | | | | | | | PC5 | PC6 |
| SLH2 | | | | | | | PC7 | PC8 |
| RP1 | | | | | | | PC9 | PC10 |
| MA | | | | | | | PC11 | PC12 |
| SH2 | | | | | | | PC13 | PC14 |

PC: Propellant combination
PC1...4: Liquid systems
PC5...12: Hybrid systems
PC13,14: Solid systems

Tab.3-2: Investigated propellant combinations

## 3.2 THRUST CHAMBER PERFORMANCE PARAMETERS

The thrust chamber is the basic element of a chemical rocket engine. Typical chamber parameters (defined in Tab.3-3) are shown in the following figures 3-2 to 3-6, based on thermochemical computations which equate the heat of reaction of the propellant combinations and the rise in enthalpy of the combustion gases at frozen composition [16].

| Parameter | Symbol | Unit |
|---|---|---|
| Specific heat ratio | $\kappa$ | - |
| Characteristic velocity | $C^*$ | m/s |
| Gas temperature | $T_c$ | °K |
| Bulk density | $\delta_b$ | kg/m³ |
| Vac. specific impulse | $I_{vac}$ | m/s |
| Densitiy impulse | $I_d$ | Ns/dm³ |

Tab.3-3: Thrust chamber performance parameters

All parameters are plotted against the metallic hydrogen mass fraction (respectively weight fraction). All data are given for 68:1 expansion ratio. The energy storing capacity of the propellant gas molecules is indicated by the specific heat ratio. Increasing values of $\kappa$ indicate decreasing energy storage capabilities, due to a lower number of degrees of freedoms, and in turn gives lower engine performance.

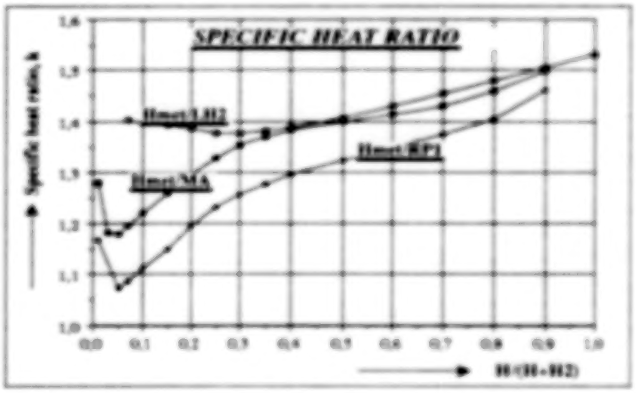

Fig.3-2: Theoretical specific heat ratio data for given propellant combinations (frozen flow, 68:1, Pc=50 bar)

Increasing atomic hydrogen weight fractions in the propellant yield higher thrust chamber gas temperatures and thus, will lead to higher atomic hydrogen mol fractions of the combustion gases. Simple atoms, however, have only three translational degrees of freedom, and hence, yield lower specific heats, which results in increasing values for $\kappa$.

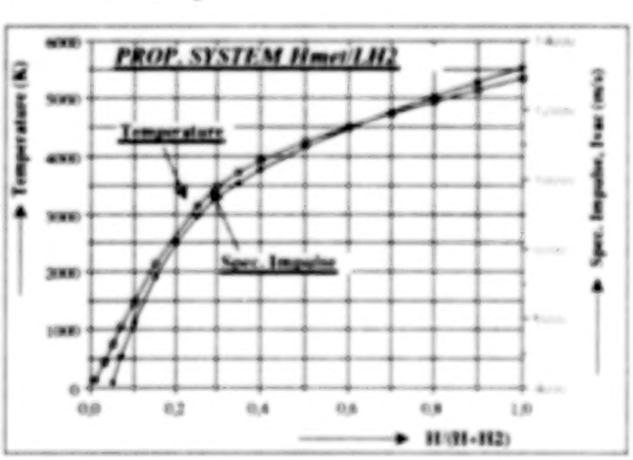

Fig.3-3: Theoretical data of gas temperature and vac. specific impulse for Hmet/LH2 combination (frozen flow, 68:1, Pc=50)

As may be seen from figures 3-3,4, the vacuum specific impulses of the systems Hmet/RP1 increases nearly linear for a wide range of Hmet weight fractions (nearly the same for system Hmet/MA), compared to the system Hmet/LH2. Moreover a very high temperature level is reached very soon with increasing Hmet weight fraction, compared to the system Hmet/LH2, which yields a more constant gradient for the temperature curve. Increased chamber pressures shift the curves to higher values for all combinations.

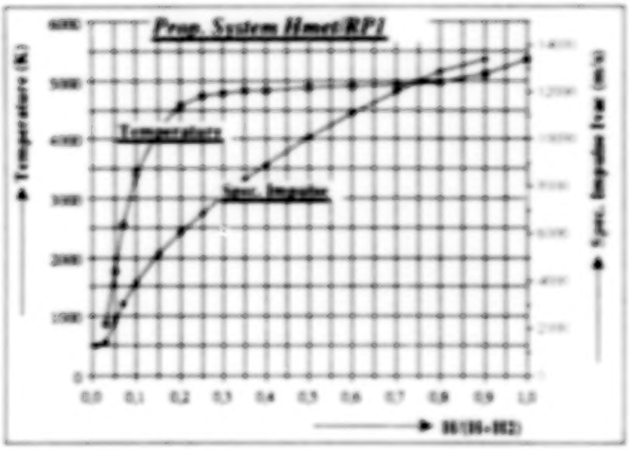

**Fig.3-4: Theoretical data of gas temperature and vac. specific impulse for Hmet/RP1 combination (frozen flow, 68:1, Pc=50)**

Results:

- Hmet propulsion systems with specific impulses of today's values (4500 m/s), have combustion gas temperatures much lower (beneath 2000 K).
- High Hmet weight fractions yield high specific impulses, but also high combustion gas temperatures (up to 5500 K)
- The gas temperatures of the systems Hmet/RP1,MA rise rapidly with Hmet weight fraction, those of System Hmet/LH2 rise slowly.
- An increase of %Hmet from 30% to 80% will hold gas temperatures nearly constant below 5000 K for systems Hmet/RP1,MA. The specific impulse increases constantly from about 7600/7000 m/s to 12500/12200 m/s within the same limits.
- The specific impules of the system Hmet/LH2 rise more rapidly with Hmet weight fraction than those of the systems Hmet/RP1,MA.
- The curves for temperature and Ivac of the system Hmet/H2 behave nearly identical.
- Hydrogen as working fluid is advantageous, due to less temperature criticality.

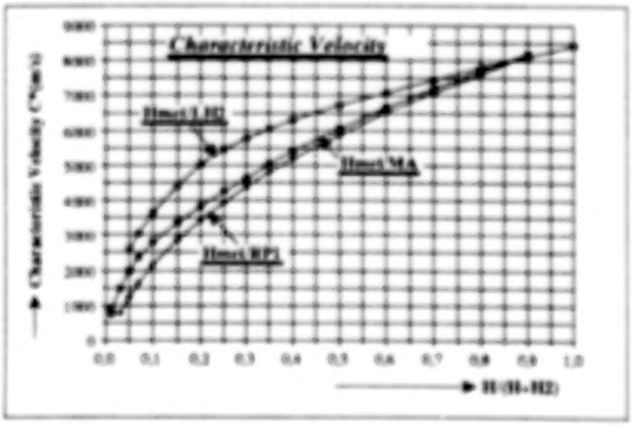

**Fig.3-5: Theoretical data of characteristic velocity and vac. specific impulse for given propellant combinations (frozen flow, 68:1, Pc=50 bar)**

The computations of the characteristic velocities yield the following results:

- The higher values of the characteristic velocity in the sequence H2, MA, RP1 indicate a combustion process of higher energy and efficiency corresponding to a lower value of propellant consumption.
- Hydrogen as working fluid represents the best alternative due to highest values for both, characteristic velocity and vac. specific impulse, at low Hmet weight fractions.

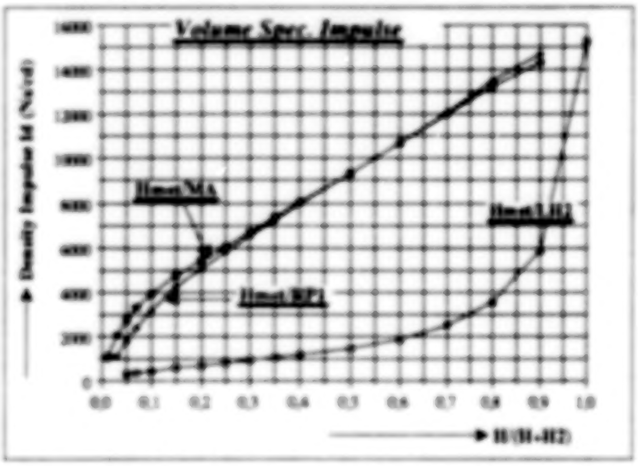

**Fig.3-6: Theoretical data of density impulse for given propellant combinations (frozen flow, 68:1, Pc=50 bar)**

The results of the computation of the volume specific impulse are:

- The more dense working fluids RP1 and MA shows an impressive advantage concerning tank volume reductions, compared to hydrogen, which yields no more than 430 kg/m3 of bulk density even with 90% Hmet.
- Therfore the density impulses are much higher, using RP1 or MA as working fluids.

To give a comparison of the combustion behaviors between the metallic hydrogen propellant combinations and other conventional liquid propellant combinations used today, the vac. specific impulse and gas temperature are plotted in Fig.3-7.
If the specific impulse with metallic hydrogen combustion is fixed on the level of the conventional high energetic combinations (LOX/LH2, LOX/F2), the gas temperatures will be lower. In the case of the constant gas temperature, the specific impulses are much higher.

Results:

- There are lower thermal risks of thrust chamber due to lower chamber temperatures, if metallic hydrogen propulsion is used in the realm of conventional specific impulses (low percentages of metallic hydrogen).
- An enormous increase in specific impulse arises, if chamber temperatures are not kept within conventional limits.
- The combination using hydrogen as working fluid shows the most extreme behavior in this sense.

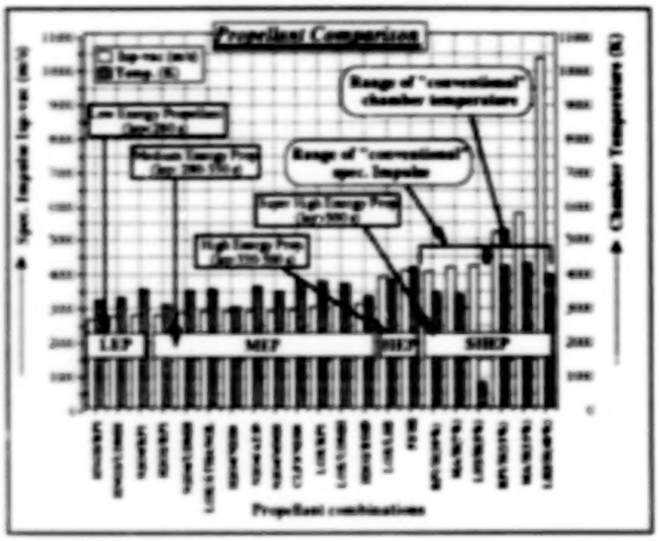

**Fig.3-8: Comparison of various propellant combinations (frozen flow, 68:1, Pc=50 bar)**

## 3.3. THRUST CHAMBER COOLING

In the case of increase of the metallic hydrogen fraction above 15% for the system Hmet/MA and Hmet/RP1 respectively above 40% for Hmet/LH2 the chamber temperatures will increase over the values for the conventional propellant combinations which lies in the range between 2500 K and 3700 K. Because of the high heat transfer rates from the hot gases to the chamber wall, thrust chamber cooling becomes a major design consideration. The objective was to investigate the influences of gas temperatures, arising from Hmet-combustion, on chamber cooling demands. The results can only be regarded as simple approximations.

Fig.3-9 shows schematically the cooling problem, which is basically one of heat and mass transport associated with conduction through a wall. It can be treated as a series type heat-transfer problem.

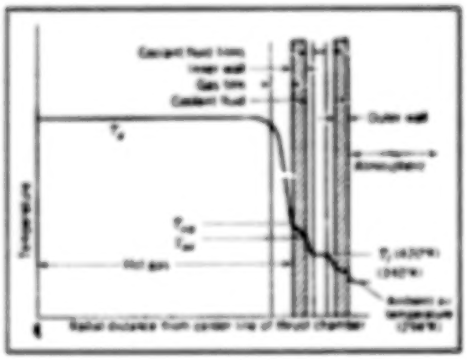

**Fig.3-9: Temperature gradients through chamber wall [18])**

The general steady-state heat-transfer equation can be expressed as follows:

$$q=H(Tg-Tl)=Q/A$$

where

q : Heat flux or heat transferred per unit area per unit time [W/m2]

H : Overall film coefficient or overall heat transfer coefficient [$W/m^2$ °K]

Tg : Absolute chamber gas temperature [K]

Tl : Absolute coolant liquid temperature [K]

Q : Heat transferred per unit time [W] across a surface A [$m^2$]

The determination of the overall film coefficient H is a rather complex problem. It can be expressed as follows:

$$H=1/(1/hg+t/k+1/hc)$$

where

t : Chamber wall thickness [m]

k : Thermal conductivity of chamber wall [W/m °K]

hg : Gas film coefficient [$W/m^2$ °K]

hc : Coolant liquid film coefficient [$W/m^2$ °K]

H is composed of the individual coefficients for the boundary layers and the chamber wall. The smaller H, the smaller is q. It is one of the major design goals to keep gas side heat transfer coefficient hg low, but the coolant liquid heat transfer coefficient and conductivity t/k high, in relation to hg. The cooling problem will be analysed in a very simple manner, based on the given data for q and H of the SSME thrust chamber. It will simply be answered, how much the cooling parameters q and H will change relative to the SSME data, as function of the relative change of chamber temperature (which is in turn dependent on Hmet weight fraction).

The following table 3-4 states the most important material parameters exemplary for the SSME, which uses the today's most developed integral CuAgZr design.

| Parameter | Unit | Integral CuAgZr design of SSME | Tendency |
|---|---|---|---|
| Thermal conductivity | W/cm K | 3.5 | should be high |
| Coefficient of thermal expansion | K·10-6 | 16.5 | should be low |
| Wall thickness | cm | 0.07 | should be low |
| Overall film coefficient H | $W/cm^2$ °K | 4.797 | should be high |
| Poissons ratio of inner wall material | - | 0.34 | should be high |
| Max. heat flux q | $W/cm^2$ | 2800<br>(for $\eta_{cooling}$=75%) | should be high |

**Tab.3-4: SSME chamber material parameters and overall tendency**

Fig.3-10,11 gives an impression of the cooling difficulties induced by chamber temperatures above today's levels. The percent changes of the heat flux, dq, the temperature drops from absolute chamber gas temperature to the absolute liquid coolant temperature, dT, and the change of the overall film coefficient, dH, are plotted over the change of metallic hydrogen weight fraction for the combinations Hmet/LH2 (Fig.3-10) and Hmet/MA (Fig.3-11).

The zero-line represents the SSME technology with chamber temperature of Tc=3637 K and coolant liquid temperature of Tl=420 K. The changes for dq have been computed for constant dH and just the other way round, based on the general steady-heat tranfer equation. It should be noted here, that this approxi-

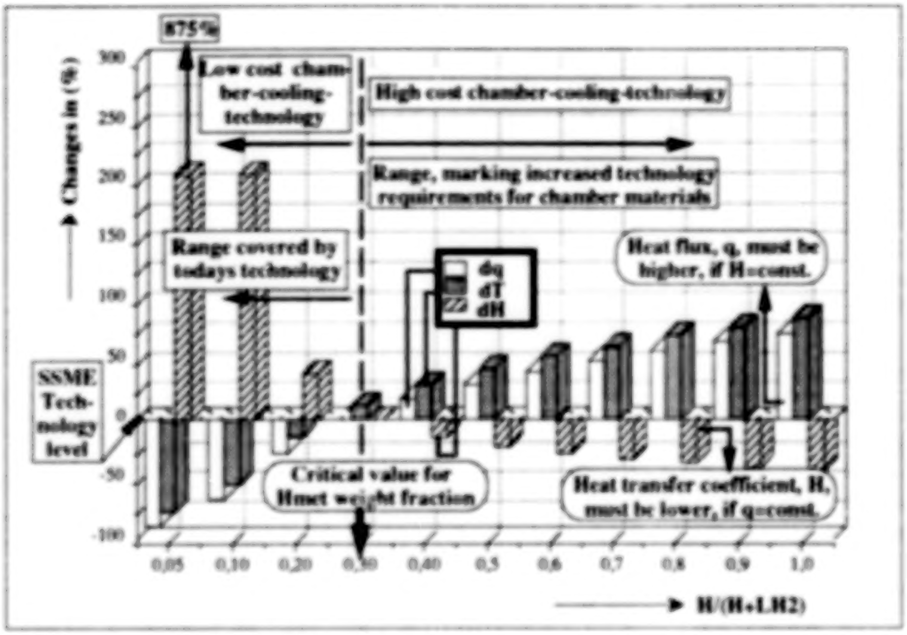

**Fig.3-10: Sensitivity of chamber cooling for propellant system Hmet/LH2, relative to SSME chamber cooling**

mat computation does not regard the influence of changed gas composition by using other propellants than in SSME. The results given below are optimistic limits. The points in the figures 3-10,3-11, where the bars cross zero, will probably move to the left, due to the fact, that the investigated propellants yield lower molecular weights than LOX/LH2, and hence yield higher gas side heat transfer rates.

The changes of premises indicate the beginning of the range, marking increased technology requirements for the chamber materials (at the point, where thought lines of communication between the bars are crossing the zero-line). This critical range begins for metallic hydrogen fractions obove 30 per cent with Hmet/LH2 propellant combination and above 5 per cent with Hmet/MA due to increasing chamber temperature.

This means for example that 50% Hmet in a Hmet/LH2 propellant yields a temperature drop of 52.9 per cent above the reference value for the SSME. This can be realized either by 40 per cent increase of the heat flux compared to the SSME with constant overall film coefficient, or by 28.5 per cent decrease of the overall film coefficient with constant heat flux capability. It it obvious, that tremendous efforts in the fields of material research are neccessery yielding material properties capable of meeting those demands.

On the other hand, Fig.3-10 gives the positive result, that today's cooling technology will be sufficient up to 30 per cent of Hmet fraction, representing the potential for low cost chamber technology.

Results:

- The propellant systems Hmet/LH2 and Hmet/MA shows different behaviors concerning cooling requirements.
- Todays cooling technology is applicable up to 30% Hmet for system Hmet/LH2.
- Today's cooling technology is applicable only up to 5% Hmet for the system Hmet/MA.
- These values are optimistic.
- The system Hmet/LH2 offers a great potential for cost savings of chamber and cooling technology if Hmet fractions beneath 40% are chosen due to their low chamber temperatures.
- The system Hmet/MA offers constant cooling conditions in the range between 20% and 60% Hmet weight fraction.
- Enhanced research towards materials with increased thermal conductivity and low thermal expansion coefficients is required.
- Further investigations concerning gas side heat transfer minimization and coolant side heat transfer maximization arenecessary.

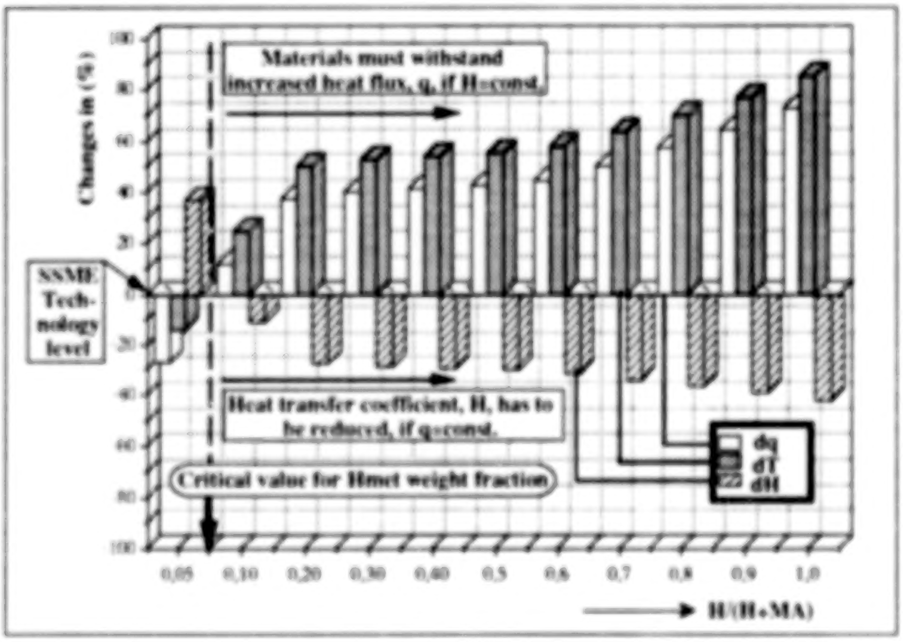

**Fig.3-11: Sensitivity of chamber cooling for propellant system Hmet/MA, relative to SSME chamber cooling conditions**

## 3.4 STORAGE CONCEPTS

This chapter contains parametric results of some reflections on basic design configurations of propellant tanks. These investigations base on the assumption of metallic hydrogen being stable at all pressures and in all different states of aggregation.

### 3.4.1 Influence on propellant volumes

An important propellant parameter is their density. High densities are desirable to minimize the size and weight of propellant tanks and feed systems.

The Hmet weight fraction dependent storage volumes for the propellant combinations Hmet/LH2,SLH2,SH2,RP1,MA have been computed for different payload masses (5 Mg, 20 Mg and 200 Mg) for a 9300 m/s-SSTO-vehicle mission. Absolute values have been connected with corresponding values for a conventional LOX/LH2 system (O/F=5; Expansion ratio 68:1; eq. flow) yielding 4111.67 m/s specific vacuum impulse. In Fig.3-12 the propellant volume fractions are plotted over the Hmet weight fraction for the different propellant combinations.

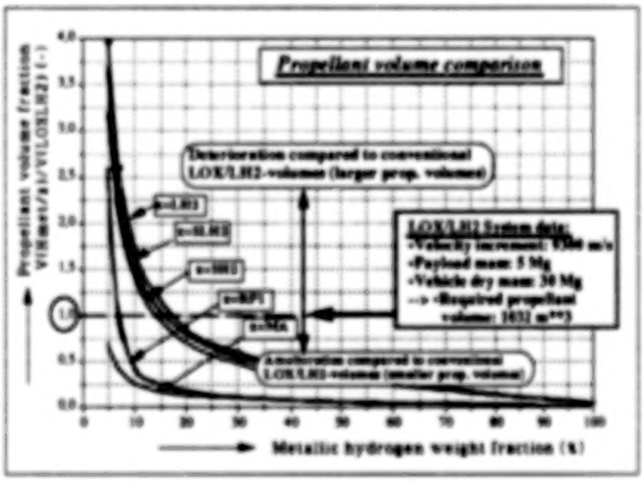

**Fig.3-12: Relative changes of propellant volume with metallic hydrogen weight fraction; factors at vertical axis indicate the change to the corresponding values for LOX/LH2 system**

All line points above a volume fraction of one are not as good as the conventional LOX/LH2 system, they have larger volumes. All line points belowa volume fraction of one represent potential mass reductions. Hmet weight fractions above 17% yield propellant volume reductions, compared to conventional systems.

In Fig.3-13 the volumes data are expressed in terms of diameters of spherical tanks to increse the vividness.

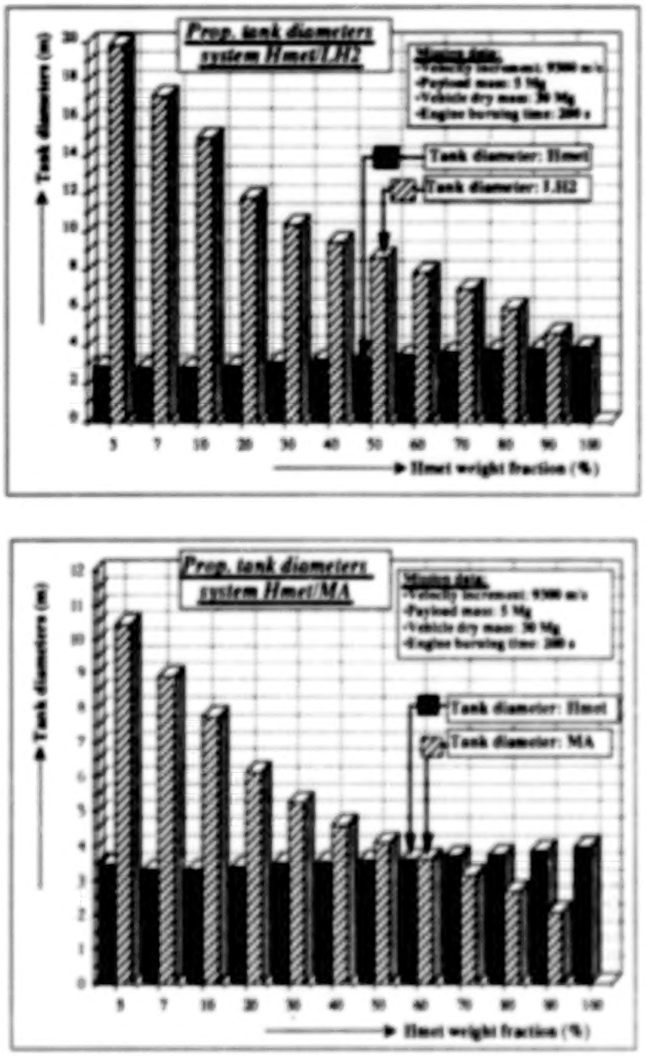

**Fig. 3-13: Required volumes expressed as diameters of spherical tanks of propellant components plotted over metallic hydrogen weight fraction**

Results:

- If hydrogen in liquid, slush or solid state is used as secondary propellant component at least 15%, 16 respectively 17% mealic hydrogen weight fraction is required to be smaller in propellant tank volume compared to the conventional LOX/LH2 system.
- Most effective reductions in propellant volumes are achieveable with methyl alcohol and RP1 as secondary component. Less than 10% Hmet weight fraction will be sufficient.
- 40% Hmet weight fraction yield propellant volume reductions less than a factor of 0.5 for all systems.
- Increasing Hmet weight fraction reduces secondary component propellant volumes very strongly meanwhile the Hmet propellant volume increases moderately.

### 3.4.2 INFLUENCE OF PROPELLANT STORAGE TEMPERATURE

Metastable metallic hydrogen propellant could call for storage temperature down to 5 K. If the cryogenic propellant components LH2, SLH2 or SH2 are used, storage temperature still down to 12 K ( for SH2) is required. The most serious tank design problems for cryogenic propellants reduces to the design of adequate thermal tank insulation. A simple approximation of the insulation thickness and specific density will answer the question, if storage temperatures in the mentioned ranges will be problematic.
The insulation requirements may be specified for the three phases listed in Tab. 3-5.

| Requiring phase | Objective | Material |
|---|---|---|
| Ground hold period | To reduce evaporative losses and [illegible] | Insulating blankets (removed prior to liftoff) |
| Boost phase | To reduce heat transfer due to aerodynamic heating | e.g. Laminated-type insulation |
| Coast flight in space | To prevent propellant from radiation from sun and the planets | Radiation shield (magnesium oxide, silver applied as coating onto light [illegible] aluminum) |

**Tab.3-5: Insulation requirements for propellant tanks**

From the mass of propellant tank point of view the ground hold period and boost phase are of importance. To investigate the latter point, the general steady-state heat-transfer equation has been used:

$$q=Q/A=(k/t)\cdot DT$$

where

| | |
|---|---|
| q | : Heat flux or heat, transferred per unit area per unit time [W/m2] |
| Q/A | : Heat transferred per unit time [W] across a surface A [m2] |
| t | : Chamber wall thickness [m] |
| k | : Thermal conductivity of chamber wall [W/m oK] |
| DT | : Temp. differential across the insulation (T2-T1) [K] |

Tab. 3-6 shows the regarded temperatures during the ground hold phase and the boost phase.

| T (K) | Ground hold phase Tg=294 | | | Boost phase Tb=[illegible] | | |
|---|---|---|---|---|---|---|
| | | DT | | | DT | |
| T1min | T1gmax | T2g-T1min | T2g-T1gmax | T1bmax | T2b-T1min | T2b-T1bmax |
| 5 | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| 14 | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| 294 | 294 | 0 | 0 | [illegible] | [illegible] | [illegible] |

T1 : Temperature of insulation surface near the tank wall
T2 : Temperature of the outer insulation surface
g : Ground hold phase
b : Boost phase
T1gmax = T1min+2.5
T1bmax = T1gmax+50 K

**Tab. 3-6: Absolute temperatures and temperature differentials**

As insulation a honeycomb light weight system has been choosen due to wide application for cryo-tanks. The cross-section through the honeycomb-supported tank structure may be seen in Fig. 3-14. The insulation consists of a honeycomb core filled with foam (isocyanate-type), installed between an inner and outer facing laminate type sheet. The space of the gap between the tank wall and inner insulation surface is purged with helium to reduce the vacuum degradation by infiltration of air and to serve as leak-detection device. This insulation system delivers a thermal conducitivity of about 2.883E-2 W-m/m²-°K (which is equivalent to 0.2 Btu-in/in²-sec-°F).

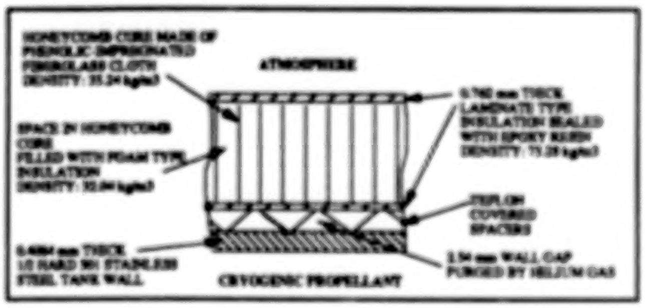
HONEYCOMB CORE MADE OF PHENOLIC IMPREGNATED FIBERGLASS CLOTH DENSITY: 35.24 kg/m3
ATMOSPHERE
SPACE IN HONEYCOMB CORE FILLED WITH FOAM TYPE INSULATION DENSITY: 32.04 kg/m3
0.762 mm THICK LAMINATE TYPE INSULATION SEALED WITH EPOXY RESIN DENSITY: 73.28 kg/m3
TEFLON COVERED SPACERS
0.4064 mm THICK 1/2 HARD 301 STAINLESS STEEL TANK WALL
CRYOGENIC PROPELLANT
2.54 mm WALL GAP PURGED BY HELIUM GAS

**Fig. 3-14: Construction elements of a cryogenic tank insulation design (external type); [from 15]**

The relative changes of thickness and density with storage temperature connected to the conventional liquid hydrogen system (T-storage: 20 K) is shown in Fig. 3-15. The approximated values are given for the ground hold and boost phase requirements.

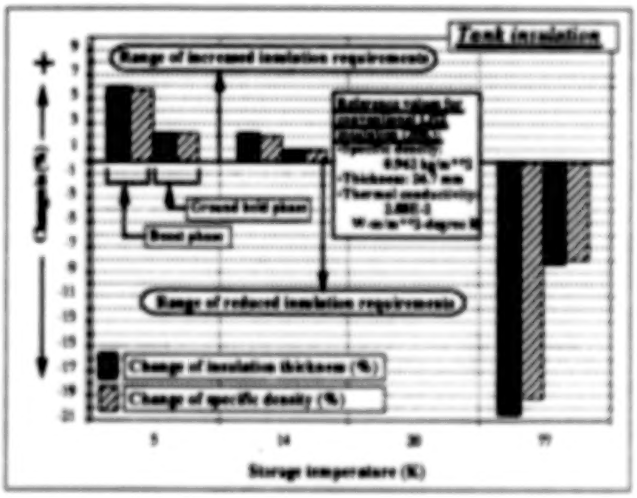

**Fig. 3-15: Approximation of per cent changes of the thikkness and specific density of propellant tank insulation material for various propellant storage temperatures, connected to conventional liquid hydrogen storage conditions (20 K)**

Results:

- Storage of propellants even at temperatures of about 5 K is technically feasible and will not increase the insulation masses dramatically
- Propellant storage temperatures in the range of slush hydrogen temperature (14 K) respectively solid hydrogen temperature (5 K) require an increase of insulation material thickness of about 2.25%/5.99 % for minimum values of ground- hold phase respectively 0.87%/2.2% for boost phase conditions, compared

to liquid hydrogen storage conditions (T=20 K)

- The increase of required specific density of insulation material will not exceed 5.73% in the case of ground hold phase, respectively 2.21% in the case of boost phase, if storage requires 5 K.
- If storage is needed at 77 K a dramatic decrease of insulation material thickness (-20.6%- ground hold phase) and density (-19.13%- ground hold phase) will happen.

## 3.5 PROPULSION CONCEPTS

The propulsion concept design will primarily be the question of tank arrangements, dependent on the state of aggregation of the propellants. Although the probability, that metallic hydrogen will exist in a solid state, a liquid system has also been investigated. A general review about the positive and negative characteristics of liquid, hybrid and solid hydrogen propulsion systems will be given below.

### 3.5.1 Liquid metallic hydrogen propulsion systems (Hmet/LH2,SLH2,RP1,MA)

The liquid systems are the propellant combinations PC1,2,3,4 from Tab.3-2. Fig.3-15.shows the overall propellant tank heights, based on a conventional bipropellant tandem arragement. The maximum diameters have been defined to 6 m for the hydrogen tank and 3.8 m for the MA tank.
As may be seen from Fig.3-15, the overall tank heights can be reduced dramatically with metallic hydrogen propulsion systems, compared to conventional launchers.
The principles of liquid metallic hydrogen propulsion concepts are shown in Fig.3-17 (p. 12). The feeding of the investigated liquid propellants can be done by conventional methods, either by gas-pressure systems, by turbopump systems or by combinations. A new feeding alternative of Hmet could be realized by electricity due to the conductive properties of Hmet. The required technology could be adopted e.g. from mass driver concepts. Magnetic fields moving down the Hmet propellant ducts acting as mass driver. Disadvantageous is the need of extra power required on board. It is conceivable, that, during the next millenium, energy transmission from the ground to the vehicle e.g. by laser beams could reduce this problem only to one of energy conversion. Some aspects concerning the differnt feeding systems are given:

- If metallic hydrogen is available at temperatures in the range of 5 K further research is necessary as to rotating parts of a turbopump system
- Lower mass flows make the use of turbopump system easier
- If lower percentages of Hmet are used (together with over proportional secondary mass requirements) a pressure feed system could be suitable for Hmet-feeding, a turbopump system for the secondary component

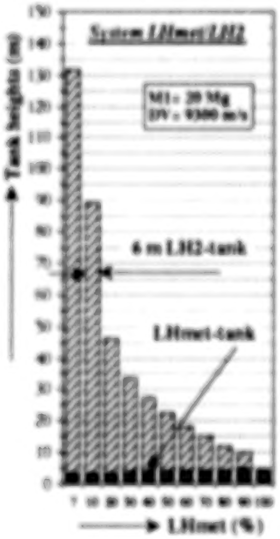

**Fig.3-15: Propellant tank dimensions for liquid metallic hydrogen system (m1: 20 Mg; DV: 9300 m/s)**

### 3.5.2 Hybrid metallic hydrogen propulsion systems (SHmet-P,G/LH2,SLH2,RP1,MA)

The systems PC5 to PC12 (see Tab.3-2, p. 4) use metallic hydrogen in solid form in conjunction with liquid working fuels. Metallic hydrogen in solid state is most likely. It can be classified as follows:

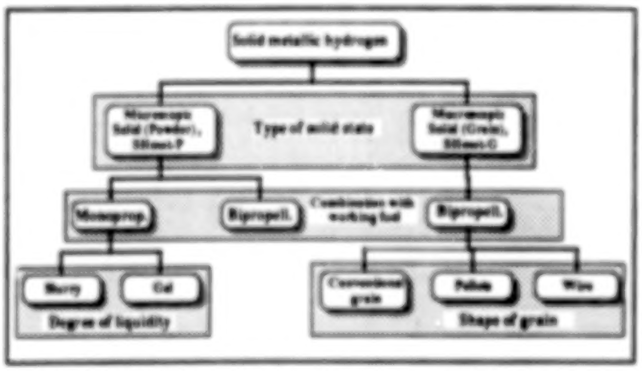

**Fig.3-16: Different forms of solid metallic hydrogen**

SHmet could be stored as powder (microscopic particle size, SHmet-P) or as grain (macroscopic particle size, SHmet-G). The powder concept can be subdivided according to the number of required tanks into monopropellant and bipropellant systems. Monopropellant systems offer the chance to reduce the complexity of the overall storage system. Solid hydrogen could be used as suspension in direct combination with the secondary propellant component as slurry or gel. The latter is more viscous than a slurry propellant.
The conventional grain concept represents the classical solid grain, which is embedded inside the thrust chamber. Small pellets are stored in an extra tank. They will be inserted into the thrust chamber like the ammunition of a submachine gun. The rolling-up of solid metallic hydrogen wire represent another

grain concept.

Common to all solid concepts except suspensions, is the lower package respectively storage density due to air-spacing. However, this should be without prejudice for the overall system, if the fraction of metallic hydrogen is great enough, as mentioned in previous chapters. The different concepts are explained below. Most of them have been previously described for conventional metal combustion.

### 3.5.2.1 Powder bipropellant concept

This concept regards the solid hydrogen propellant as powder, which is stored isolated from the respective working fluid. It will further be distinguished between separate injection (concept A) and common injection (concept B) into the thrust chamber, as may be seen from Fig.3-17. The SHmet-P propellant will be fed by high pressure gas. The working fluid will be fed by a turbopump driven by a turbine, based on an open combustion tap-off cycle.

In concept A the SHmet-P will be injected directly into the thrust chamber. Concept B introduce a mixing of the components before injection into the chamber. The mixing process is realized by means of the ejector principle. The main advantages and disadvantages of the two diergol concepts are summarized In Tab.3-7.

<table>
<tr><th></th><th>Concept A</th><th>Concept B</th></tr>
<tr><td rowspan="2"><b>Advantages</b></td><td colspan="2">•High storage density<br>•Easy controllable force of reaction</td></tr>
<tr><td>•No mixing of fuels before injection</td><td>•More simple ignition head</td></tr>
<tr><td rowspan="2"><b>Disadvantages</b></td><td colspan="2">•Feeding of SHmet-P only by means of pressure gas<br>•Lump appearance in the tank due to humidity<br>•Dust formation in the tank<br>•Danger of clogging in the ducts/lines, valves<br>•Complicated filter technique required<br>•Complicated tank geometry<br>•No literature available<br>•Unfavourable SHmet-P tank form</td></tr>
<tr><td>•Complicated injection system<br>•Mixing process in thrust chamber</td><td>•Design of ejector system<br>•Temperature gradients during mixing in case of temp. differences</td></tr>
<tr><td><b>Problems</b></td><td colspan="2">•Determination of optimal particle size<br>•Production of homogeneous particle size<br>•Fueling of SHmet-P<br>•Flow behavior (wall influences)<br>•Valve technology<br>•Monitoring instrumentation (flow, quantities and levels in tank)<br>•Thermal behavior due to particle friction<br>•Ignition behavior</td></tr>
</table>

**Tab.3-7: Different aspects concerning SHmet-P-bipropellant concepts (A: separate propellant injection; B: common injection)**

### 3.5.2.2 Powder monopropellant concepts (slurry/gel)

A monopropellant system is a potential alternative to reduce the system complexity and therewith a good chance to reduce the space transportation costs. The problems of slurry combustion are described in literature [17]. Although these investigations concentrated on coal-oil slurry combustion, many aspects can be adopted for a metallic hydrogen slurry concept.

Slurry means a suspension of solid particles inside the liquid working fluid.

A gel is a liquid containing a colloidal structural network that forms a continuous matrix and completely encloses the liquid phase. For comparison of both concepts, see Tab.3-8. Illustrations of the concepts can be seen in Fig.3-17.

<table>
<tr><th></th><th>Slurry concept</th><th>Gel concept</th></tr>
<tr><td rowspan="2"><b>Advantages</b></td><td colspan="2">•Reduction of overall system complexity<br>•Reduced number of components<br>•Small vehicle size<br>•Research projects just under way</td></tr>
<tr><td>•No mixing of fuels before injection</td><td>•No propellant sloshing<br>•Handling ease</td></tr>
<tr><td rowspan="2"><b>Disadvantages</b></td><td colspan="2">•Feeding of SHmet propellant by means of pressure gas (if gel propellant)<br>•Danger of clogging in the ducts/lines, valves<br>•Complicated filter technique required<br>•Special pumps necessary due to high viscosity<br>•Bypass system required (if clogging)<br>•Flow behavior dependent on temperature<br>•Coolant of chamber is problematic<br>•High viscosity (can be reduced by heating)</td></tr>
<tr><td>•Propellant sloshing<br>•Required constant mixing</td><td>•Storage temperature dependance<br>•Shear forces required to make it flowing<br>•Additives (as gellactants) required</td></tr>
<tr><td><b>Problems</b></td><td colspan="2">•Filtering of the slurry to prevent clogging<br>•Mechanical stabilization in the anticipated environment<br>•Influence of Hmet weight fraction on flow behavior<br>•Determination of optimal particle size<br>•Production of homogeneous particle size<br>•Influence of particle weight distribution<br>•Fueling<br>•Flow behavior (wall influences)<br>•Valve technology<br>•Monitoring instrumentation (flow, quantities and levels in tank)<br>•Thermal behavior due to particle friction<br>•Ignition behavior<br>•Combustion characteristics (droplet size, evaporation behavior, burning velocity,...)</td></tr>
</table>

**Tab.3-8: Comparison of SHmet-P-monopropellant concepts**

### 3.5.2.3 Hybrid grain concepts

Here macroscopic grains of Hmet are regarded. It is completely unknown how solid metallic hydrogen in concentration of 100% will interact with its environment. From solid atomic hydrogen imbedded inside a solid molecular hydrogen matrix, high regression velocities in the range of 2.1 m/s are known. For this study metallic hydrogen mass flows will be assumed to be in the range of today's magnitudes, possibly by means of additives.

In the case of the conventional grain concept solid metallic hydrogen propellant is contained within the combustion chamber, in which the liquid working fluid will be injected. A more detailed critical examination may be seen from Tab.3-9.

The pellet concept means, that spherical pellets of SHmet will be injected into the thrust chamber. Disadvantageous is the lower propellant package density in the tank and fluctuation of the combustion, dependent on the injection rate. Advantageous is the flow rate controllability and engine shut down capability in case of emergency. Production could be easy due to the small size particles, delivered by the diamond anvil cell.

The wire concept is a completely new one. Solid metallic hydrogen is spooled onto a coil which could be driven by electric energy or pressure gas. Full thrust controllability is given. Disadvantageous will be the lower propellant package density combined with additional mass for the coil and bearings as well as the difficult sealing of the feeding lines. Moreove, the solid propellant has to be pliable.

The different grain concepts are compaired in Tab.3-9 while illustrations are shown in Fig.3-17.

| | Conventional grain | Pellets | Wire |
|---|---|---|---|
| Advantages | -High propellant density<br>-Nearest probability of Hmet use | | |
| | -Less cooling problems of the chamber wall<br>-Low package density<br>-Proven concept<br>-Partially thrust control | -Fully thrust control<br>-Feeding problematics known from DAEDALUS<br>-Easy production in small cells<br>-Engine shut down | -Fully thrust control<br>-Engine shut down<br>-Mechanical feeding by pressure gas |
| Disadvantages | -High concentrated energy density | | |
| | -High burning velocity<br>-Additives necessary to reduce burning velocity<br>-Feeding of liquid component by pressure gas | -Package density dependent on pellets size<br>-Sealing<br>-Combustion fluctuations | -Package density dependent on wire geometry<br>-Complicated feeding system<br>-Difficult sealing of feeding lines<br>-Required pliability |
| Problems | -Influence of electric and magnetic fields<br>-Combustion behavior (velocity, stability) | | |
| | -Combustion cut off | -Optimum pellets size<br>-Statistic deviation of optimum size<br>-Pellets feeding rate | -Propellant distribution in chamber |

**Tab.3-9: Comparison of potential grain concepts**

### 3.5.3 Solid metallic hydrogen propulsion system (SHmet-P/SH2)

A solid propellant metallic hydrogen propulsion system is conceivable as a solid hydrogen matrix, in which solid metallic hydrogen particles are imbedded. It makes more sense to use a microscopic powder rather than macroscopic solid particlesbecause of the more homogeneous combustion behavior.

A concept like this offers the known conventional advantages, listed in Tab.3-10. They vanish immediately, if e.g. stability of solid particles can't be assured in case of quarrelsomeness between SH2 and SHmet (e.g. due to different component temperatures or mechanical respectively thermal sensitivity). A simple illustration of the concept can be seen in Fig.3-17.

| | |
|---|---|
| **Advantages** | •No feed systems<br>•No valves<br>•Simple in construction |
| **Disadvantages** | •Difficult propellant production<br>•Low storage temperatures<br>•No thrust modulation<br>•Long duration storage problematic<br>•Cigarette-burner (->small burning area, changing chamber volume) |
| **Problems** | •High temperature gradients through propellant during combustion<br>•Unknown combustion reaction velocity as function of % Hmet |

**Tab.3-11: Analysis of solid propellant metallic hydrogen propulsion concept**

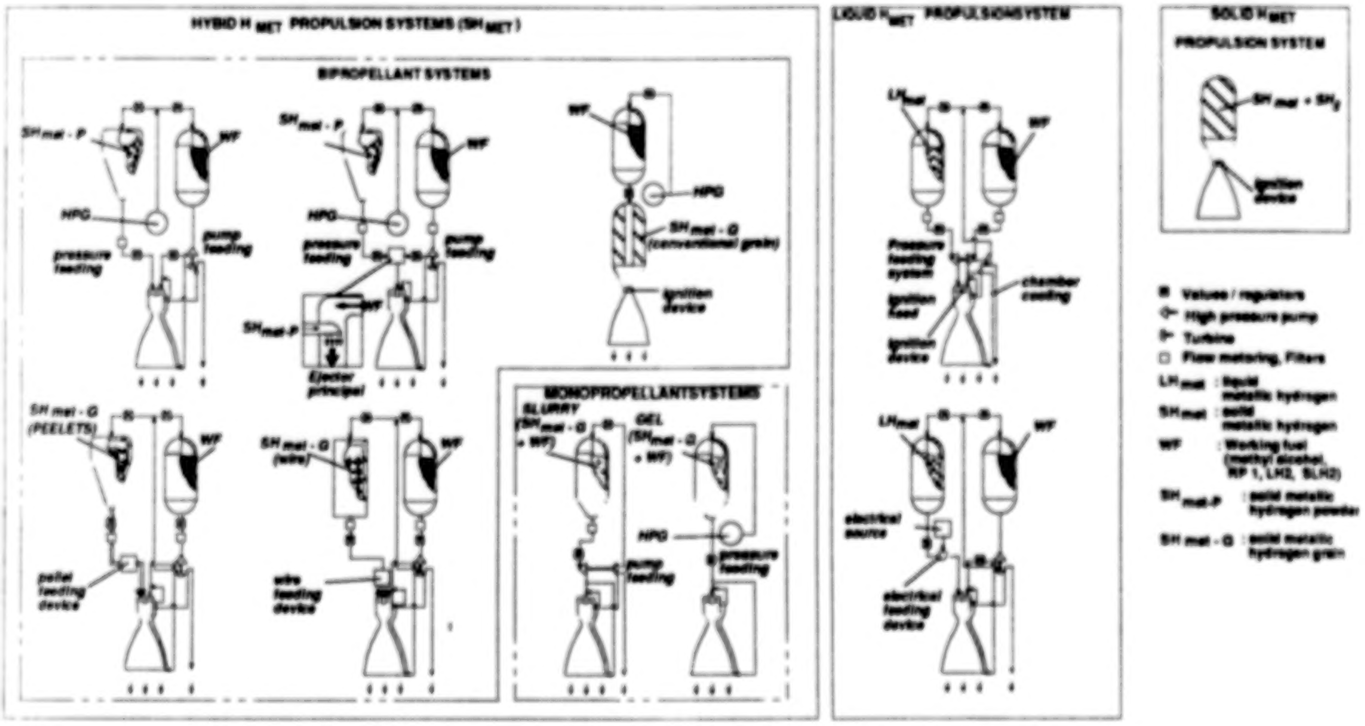
HYBID H MET PROPULSION SYSTEMS (SH MET)
BIPROPELLANT SYSTEMS
SHmet-P
WF
HPG
pressure feeding
pump feeding
SHmet-G (conventional grain)
ignition device
Ejector principel
SH met - G (PEELETS)
pellet feeding device
SHmet-G (wire)
wire feeding device
MONOPROPELLANTSYSTEMS
SLURRY (SHmet-G + WF)
GEL (SHmet-G + WF)
LIQUID HMET PROPULSIONSYSTEM
LHmet
Pressure feeding system
ignition feed
ignition device
chamber cooling
electrical source
electrical feeding device
SOLID HMET PROPULSION SYSTEM
SHmet + SH2
ignition device
Values / regulators
High pressure pump
Turbine
Flow metering, Filters
LHmet : liquid metallic hydrogen
SHmet : solid metallic hydrogen
WF : Working fuel (methyl alcohol, RP 1, LH2, SLH2)
SHmet-P : solid metallic hydrogen powder
SH met - G : solid metallic hydrogen grain

**Fig.3-17: Illustrations of possible metallic hydrogen propulsion concepts**

## 3.6 PROPULSION PERFORMANCE

The following figure 3-18 gives the relations between metallic hydrogen percentage in the fuel and global vehicle mass fractions for the most advantageous propellant combination, Hmet/LH2. Of importance arepayload mass ratio M1/M0, dry mass ratio Mn/M0 andpropellant mass ratio M8/M0 which gives total vehicle mass in the sum.

Thevalues given base on simple basic rocket equation calculation rather than on a detailed mass model. Launch masses required for reaching the velocity increment of 9300 m/s with constant vehicle dry mass of 30 Mg, are inserted into the columns for each calculation.

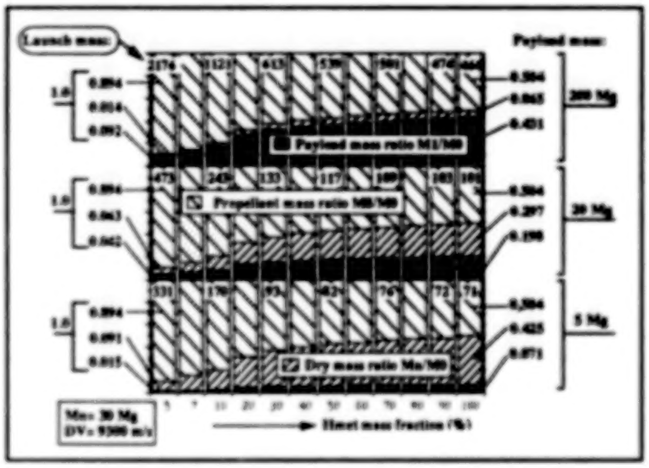

**Fig.3-18: Approximations of the main vehicle mass ratios dependent on metallic hydrogen percentage and varying payload masses (SSTO-calculation)**

Results:

- Resulting vehicle launch masses for given mission demands are low, compared to existing launchers, even at low Hmet mass fractions.
- Payload mass ratios were kept constant for different payload masses.
- Use of metallic hydrogen propellant yields a great potential for launch mass reductions.
- Increase of dry mass ratio with %Hmet results from constant payload mass.
- At low metallic hydrogen weight fractions of 5%, with a payload mass of 5 Mg represent mass ratios of todays launchers (note:here SSTO calculation with DV=9300 m/s).
- The propellant mass ratio decreases to a minumum of 50.4% with 100% metallic hydrogen .
- Payload mass ratios increase with payload mass for all combinations, while the dry mass ratios decline. Today, dry mass ratios in the range of 10% to 13% are feasible. All computed dry mass ratios above this range (see Fig.1) indicate free mass potentials which could be used either for higher payload masses or for heavier but more stable and therefore saver vehicle structures.
- Dry mass ratios for 200 Mg of payload are lower than 6.5% in each case, which indicates the need for better leight weight structures if payload mass is not to be reduced.

## 3.7 ENVIRONMENTAL ASPECTS

One dominant characteristics of super high energy metallic hydrogen propellant is the specific impulse. However there exists limits for the specific impulse primarily due to acoustics.The negative effects to the surroundings of the launch vehicle:

- Mechanical effects (e.g. on ground equipment by vibrations etc.)
- Chemical effects (e.g. on the atmosphere the vehicle is flying through)
- Thermal effects (e.g. on the atmosphere or on the ground)
- Acoustical effects (on the surrounding but also on the vehicle itself)
- Emergency destruction effects (around the point of vehicle destruction)

Not all of these effects seem to be critical for the operation of the launch-vehicle. The present paper discusses only the problematic effect, which arises due to the enormous exhaust velocities, dependent on metallic hydrogen weight fraction.

The primary impact comes from the sound, wich is defined as mechanical oscillation inside an acoustic medium. It is measured as sound pressure and sound velocity. The oscillating pressure p has the most dominant destructive influence on technical structures with resulting effects on the environment. Practically the sound pressure level Lp is measured in decibel:

The sound pressure produced by rocket engines can be divided into jet-stream noise and combustion noise. The jet-stream noise rises with the exhaust velocity and is therefore the dominating sound source.

$$Lp\ [\mathrm{dB}] = 10\ \lg_{10}\left(\frac{\tilde{p}^2}{\tilde{p}_0^{\,2}}\right)$$

where:

$\tilde{p}$: Oscillating pressure

To quantify the effect of acoustic noise, the power of sound has been calculated. Between 0.7 and 1.6 the Mach emitted power of sound by a jet-stream is raising with eight to the power of the exhaust velocity v. With an exhaust velocity greater than mach two (v > 2 Mach), the jet-stream power of sound rises with 3 to the power of the exhaust velocity (/27/, p. 281). Fig.3-19 shows the results of the parametric calculation of the acoustic noise level as function of the jet exhaust velocity with the distance r from the source as parameter.

Sound levels over 200 dB are not at all acceptable for technical structures. It should be mentioned that values over 194 dB would be equivalent to an alternating pressure greater than 105 N/m2!

As limiting values for sound pressure level it may be proposed [19]:

-145 dB as maximum stress limit for conventional rocket technical structures

-160 dB as maximum stress for highly stable designed structures and launch facilities (submersible launch towers, concrete shelters for measuring tools etc.).

Whereas from another authority [20] a maximum alternating sound pressure stress of 0.1 bar (= $10^4$ N/m²) may be regarded as tolerable. This limiting value is equivalent to a sound level of 174 dB. Future research activities at the Technical University of Berlin will include detailed investigations of the correlation betwee high exhaust velocities and generated noise power.

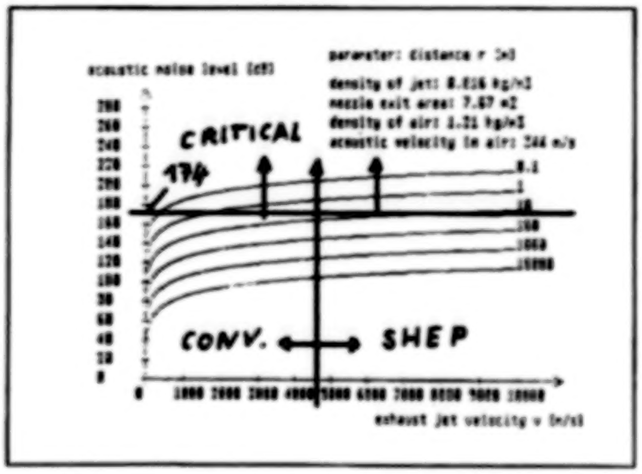

**Fig.3-19: Acoustic noise level as a function of exhaust jet velocity with distance r as parameter [19]**

Results:

- Gas exhaust velocities above those of conventional systems (4500 m/s), enhance the maximum stresses of structures (not only of vehicle) near destructive limits, due to increased noise levels.
- Metallic hydrogen weight fraction is an important parameter to be considered concerning acoustical effects.
- To take advantage of high energetic propulsion systems, methods concerning effective noise reduction (below 174 dB) have to beinvestigated.
- The only effective method in the moment to reduce the noise power, seems to be air augmentation.
- Vehicle mass savings (due to high propellant energetics) may probably be compensated partly by mechanical devices to realize air augmentation.

## 3.8 REFLECTIONS ON COSTS

The introduction of a new propulsion concept will be favoured, if there will be a potential for cost savings, namely for development and operation costs. In particular, the specific space transportation costs ($/kg-payload) respectivily the vehicle lauch costs are of importance. They can be reduced through high payload capabilities, high launch rates and low system complexity. Mostly, space system costs can be expressed as function of masses, as have been done in this study. It should be noted, that the following reflections are more general rather than based on detailed analysis.

As may be seen from Fig.3-18 (chapter 3.6), the use of a Hmet/LH2 propulsion sysstem yields overall launch masses, much lower than today's launchers. Hmet vehicles yield furthermore higher payload capabilities with less system complexity. The low launch masses respectively the high launch mass ratios represent a possible potential to reduce the system costs.

In Fig.3-20, the approximated ranges of Hmet-vhicle launch costs are marked, based on a parametric comparison with past, todays and near future launchers.

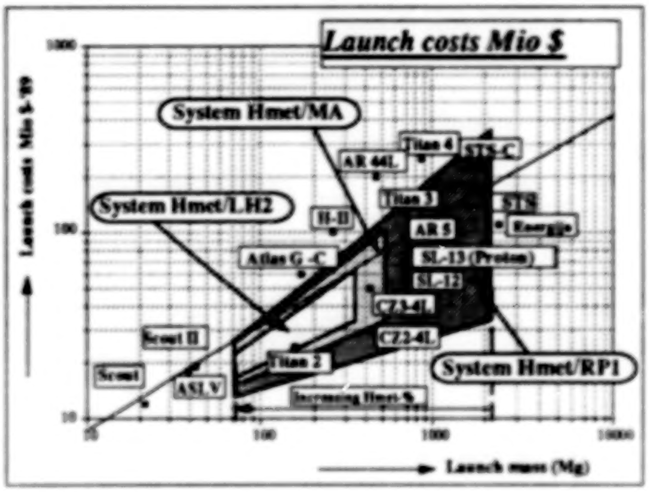

**Fig.3-20: Approximation of launch costs of metallic hydrogen transportation systems, compared to various launchers [1]**

Conclusions:

- Propellant combinations using more energetic working fluids (first hydrogen, second methyl alcohol, third RP1) yield lower launch masses and hence lower launch costs.
- Launch costs can be reduced with increasing metallic hydrogen weight fraction.
- Launch costs of Hmet-vehicles are much lower than conventional ones carrying the same payload mass.
- Areas for the Hmet-vehicles represent the upper limit of the expected launch costs, because that the mass values are based on SSTO calculations (SSTO systems are less complex compared to staded vehicles, and will therefore lead to reduced launch costs).

Fig.3-21 shows the correlation between payload capability and specific costs ($/kg-payload). The approximated range for Hmet systems is marked. Space transportation based on metallic hydrogen system will be less costly, compared to conventional systems, due to much lower launch masses for the given payload masses.

Further reflections on costs of Hmet-sysems are:

- The super high energetic propellant combination will require enhanced security demands which could lead to higher operational costs.
- Cost reduction potential due to less complex ground infrastructure (smaller propellant storage facilities, smaller hangars and launch towers, etc.)
- Design of reusable space vehicles using metallic hydrogen propellant systems could be advantageous due to smaller overall vehicle size.
- Use of hydrogen based Hmet combination could be much

cheaper than hydrocarbon combinations, due to probably spreading hydrogen house keeping world wide during the next millennium.

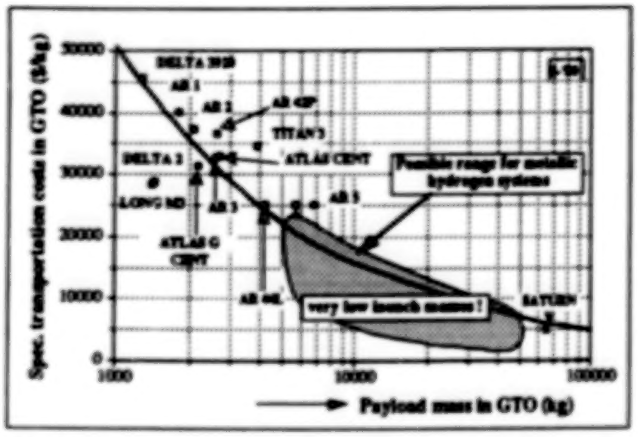

**Fig.3-21: Approximation of specific transportation costs ($-'89/kg) of metallic hydrogen transportation systems compared with conventional systems;** data of given launchers have been converted into GTO data [1]

## 4. SUMMARY

The study investigates metallic hydrogen for the application as rocket propellant. Due to its very high theoretical specific energy of 52 kcal/g, yielding a maximum specific impulse of 1700 m/s, and its high density of 1150 kg/m³, metallic hydrogen would be an interesting propellant candidate. However, there are many restrictions concerning the knowledge about metallic hydrogen:

- The required atomic state of Hmet has not yet been proven (required pressures for dissociation in the range between 2.5 and 4.0 Mbar have not been achieved today).
- Uncertainty about the metastability of metallic hydrogen (probably Hmet will not be metastable).
- Uncertainty about basic chemical and physical properties, like state of aggregation (probably Hmet will be a solid).

All investigations took place on condition that Hmet will exist in an atomic state and will be stable.
Different types of potential propulsion systems (liquid, hybrid and solid system) have been analysed. Three different propellant combinations have been compared: metallic hydrogen with hydrogen (liquid, slush and solid hydrogen) as working fluid, with RP1 and with methyl alcohol as working fluid. The main system parameter is the metallic hydrogen mass fraction, which influences the overall propulsion and vehicle performances primarily.
It may be seen from this study, that metallic hydrogen is of advantage, compared to propellants used today. The overall vehicle masses enhance the complexity of possible Hmet launchers will decrease, due to increased propulsion performance. Nevertheless, gas exhaust velocities must not be increased unlimited, due to noise impact. For that reason, the Hmet propellant combination using LH2 as working fluid is of advantage, yielding improvements of system performances even with today's exhaust velocities.
The main results and conclusions followed the investigation of combustion characteristics, thrust chamber cooling, storage concepts, propulsion system performance, environmental loads and costs are summarized below:

- All system parameters depend largely on metallic hydrogen mass fraction.
- Combustion gas temperature is mainly dependent on metallic hydrogen mass fraction (up to 6000 K with high percentages; down to 1500 K for low percentages [conventional LOX/LH2 system :3750 K]).
- There are lower thermal risks of thrust chamber due to lower chamber temperatures, if metallic hydrogen propulsion is used delivering specific impulses of today's systems (low percentages of metallic hydrogen).
- An enormous increase in specific impulse arises, if chamber temperatures are not kept within conventional limits (combination using hydrogen as working fluid shows the most extreme behavior).
- The propellant systems Hmet/LH2 and Hmet/MA shows different behaviors concerning cooling requirements.
- Today's cooling technology is applicable up to 30% Hmet for system Hmet/LH2 .
- Today's cooling technology is applicable only up to 5% Hmet for the system Hmet/MA .
- System Hmet/LH2 offers a great potential for cost savings in the fields of chamber and cooling technology.
- Enhenced material research towards increased thermal conductivity and low thermal expansion coefficients is required if high Hmet mass fractions are used.
- Metallic hydrogen weight fractions in the range of 40% will lead to increased payload mass ratios and reduced propellant mass ratios compared to conventional systems.
- Hydrogen as working fluid combined with Hmet represents the most interesting propellant alternative due to
  - lowest overall vehicle masses.
  - highest payload mass potential with respect to realizable dry mass respectively structure mass ratios even at low Hmet weight fractions.
- High percentages of metallic hydrogen may lead to a dramatic decrease of overall propellant volume due to high density of Hmet (1.15 g/cm3).
- The most effective reductions in propellant volumes are achieveable with methyl alcohol and RP1 as working fluids.
- Low temperature propellant storage (for typical prelaunch phases) at about 5 K could be achieved with conventional insulation techniques.
- Metastable liquid state of metallic hydrogen would be of advantage, compared to solid state, due to technical proven components (feeding system, valves, etc.).

- If Hmet was metastable in solid state, a hybrid system would be advantageous, using Hmet as powder or in the shape of macroscopic pellets
- Noise from rocket engines represent one of the most critical impacts to the environment
- The advantage of high specific impulses will be compensated by the disadvantage of enhancement of the maximum stresses of structures near their destructive limits due to acoustic power.
- Methods concerning effective noise reduction (below the limit of 174 dB) have to be researched (air augmentation)
- Vehicle mass savings (due to high propellant energetic and hence low propellant masses) may be compensated partly by mechanical devices required for air augmentation.
- Propellant costs will depend on high pressure facility capabilities
- There is a potential of launch costs and specific transportation costs savings due to the very low vehicle launch masses
- There could exist a cost reduction potential due to less complex ground infrastructure (smaller propellant storage facilities, smaller ground launch facilities, etc.)
- Use of reusable SSTO space vehicles using metallic hydrogen propulsion systems will no longer be a problem

Future research activities should concentrate on the following open technology problem areas:

- Development of high pressure facilities generating required pressures for Hmet production up to 4.0 Mbar
- Detailed mass model, dependent on the storage concept
- Detailed investigation of air augmentation techniques

**GENERAL MESSAGE:**

- The use of metallic hydrogen as rocket propellant could lead to revolutionary changes in space vehicle philosophy towards low weight, small size and high performance SSTO systems
- Hydrogen (liquid or slush) is the most favourable candidate as working fluid.
- Much more research in the field of high pressure physics is required to come to reliable statements about the chemical and physical properties of metallic hydrogen.
- The technical risks concerning the use of metallic hydrogen as rocket propellant may be controlable.

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N91-22155

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# 21ST CENTURY PROPULSION

**V. E. (Bill) Haloulakos**
**C. Boehmer**
**McDonnell Douglas Space Systems Company**
**Huntington Beach, California**

## NASA Symposium
## "Vision 21"
**Space Travel for the Next Millenium**

**NASA Lewis Research Center, Cleveland, Ohio**
**3-4 April 1990**

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## SPACE TRAVEL IN THE NEXT MILLENIUM?

❐ **How do we predict the future?**

- **We do so by examining the past**

  **— We predict the near future by examining the immediate past and extrapolate into the far future by studying the more distant past**

- **We also use a lot of imagination**

❐ **Constraint: In the use of our imagination, we are constrained to obey the "known" laws of physics**

❐ **Ray of hope:** The laws of physics are known to have been through many evolutionary and revolutionary changes. In fact, they are in a continuous state of flux

❐ **Example:** For many centuries physics was taught with the atom being exactly what the word means, INDIVISIBLE. Now , of course, atom is just a name and its splitting is history.

## GOALS FOR THE 21ST CENTURY

❐ Expanded space travel and establishment of permanent manned outposts

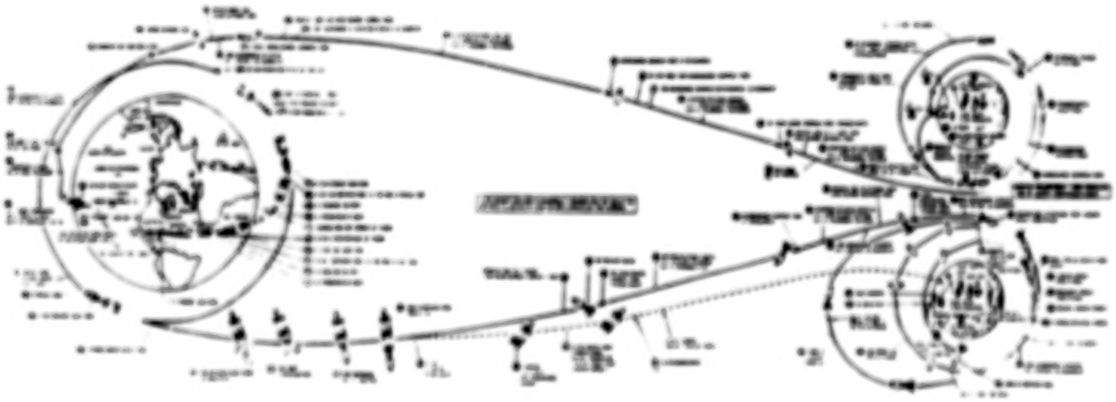
❐ Lunar and Mars outposts represent the most immediate future in space travel and both have history from the very recent past

# LUNAR OUTPOSTS

❒ The Apollo program was successfully conducted with chemical propulsion

- It was necessary to advance from the liquid oxygen/alcohol propellants of the V-2 to liquid oxygen/liquid hydrogen of the Saturn V upper stages in 15 years

# APOLLO 16 MISSION PROFILE



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# MARS OUTPOSTS

- ☐ Trips to Mars were planned and vehicles were designed for landings and returns
  - Unmanned 1984
  - Manned 1988
- ☐ Trade studies compared chemical cryogenics ($O_2/H_2$) with direct nuclear thermal propulsion using the NERVA engine
- ☐ The following charts summarize the trades and vehicle designs conducted between 1965 and 1973

---

# ON TO MARS (REVISITED)

C. BOEHMER

Presented to
2nd AIAA/JPL International Conference
on Solar System Exploration
Pasadena, California
22-24 August 1989

*McDonnell Douglas Space Systems Company*

MCDONNELL DOUGLAS

---

# HISTORY

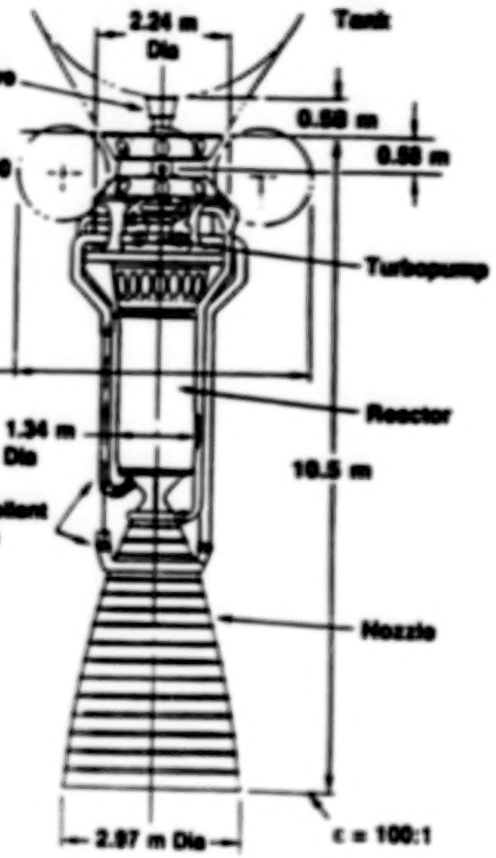
**Nuclear Engine**

- ☐ KIWI - A (Los Alamos) 1957-1960
- ☐ KIWI - B (Los Alamos) 1961-1964
  - KIWI-B4E Aug 1964
    - – 940 MW, 10-min, restart
- ☐ Phoebus (Los Alamos) 1965-1968
  - – 4100 MW, 30-min
- ☐ NRX (Aerojet/Westinghouse) 1964-1967
- ☐ XE-1 (Aerojet/Westinghouse) 1969

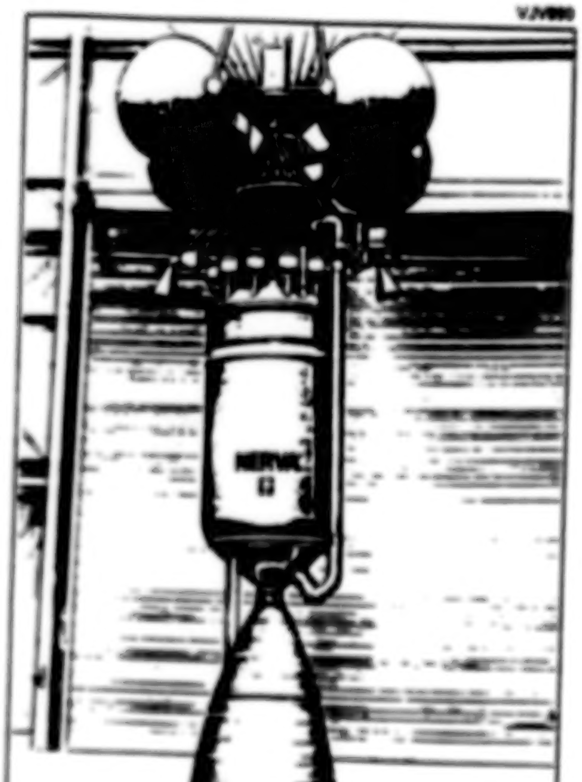
**Nuclear Stage**

- ☐ Douglas Aircraft 1965-1969
- ☐ McDonnell Douglas 1969-1973
  - Nuclear Flight Definition Study
    - – Saturn derivative
    - – Shuttle
  - Nuclear Stage System Definition Study
    - – Propulsion module
  - Reusable Nuclear Shuttle (RNS)
    - – Lunar
    - – LEO-GEO
    - – Earth – Mars

NERVA FLIGHT ENGINE CONFIGURATION


2.24 m
Dia
Tank
Tank Valve
Engine/Tank
Interface
Engine Sta 0.00
0.58 m
0.58 m
Turbopump
4.00 m
Dia
Reactor
1.34 m
Dia
10.5 m
Propellant
Lines
Nozzle
2.97 m Dia
ε = 100:1

FULL SCALE MOCK-UP OF NR-1 FLIGHT ENGINE, RATED AT 1500 MW, AND 75,000 LB THRUST



[illegible]

# NUCLEAR ROCKET REACTOR TESTING

## LASL (LANL)

| | Date | Power Level (MW) | Run Time (min) |
|---|---|---|---|
| KIWI - A | July 1959 | 70 | 5 |
| KIWI - A′ | July 1960 | 85 | 6 |
| KIWI - A3 | Oct 1960 | 100 | 5 |
| KIWI - B1A | Dec 1961 | 300 | 1 |
| KIWI - B1B | Sept 1962 | 900 | 1 |
| KIWI - B4A | Nov 1962 | 500 | 0 |
| KIWI - B4D | May 1964 | 1020 | 1 |
| KIWI - B4E | Aug 1964 | 940 | 10 |
| TNT | Jan 1965 | — | — |
| Phoebus - 1A | June 1965 | 1090 | 11 |
| Phoebus - 1B | Feb 1967 | 1500 | 30 |
| Phoebus - 2A | June 1968 | 4100 | 13 |
| Pee Wee - 1 | Dec 1968 | 500 | 40 |
| NF - 1 | June 1972 | 434 | 108 |

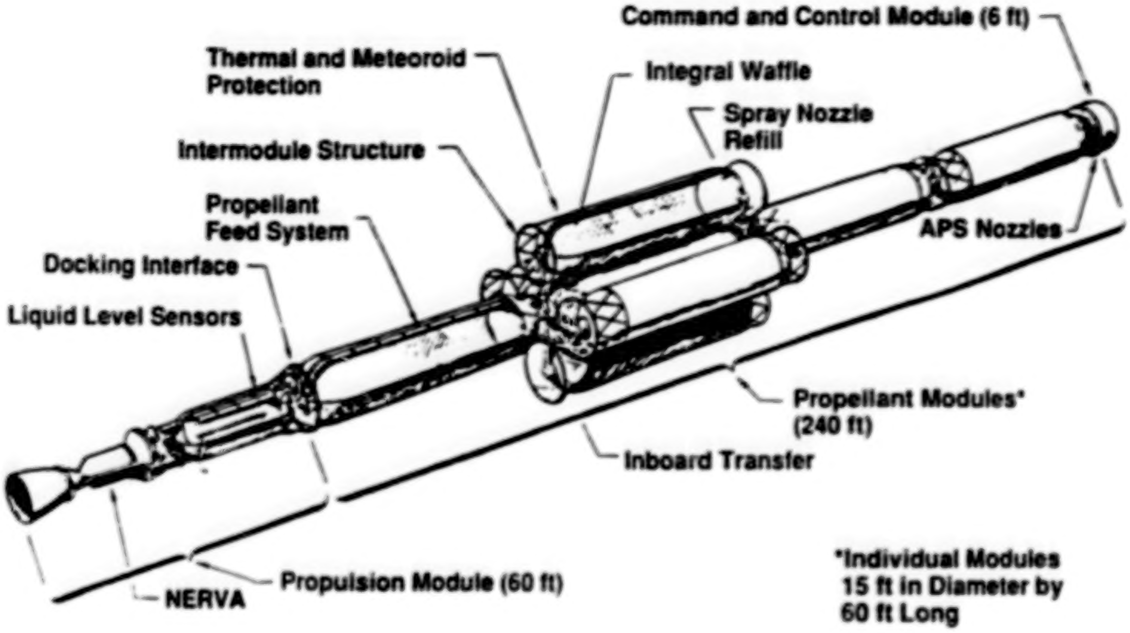
[illegible]

# NUCLEAR ROCKET REACTOR/ENGINE TESTING WESTINGHOUSE/AEROJET

| | Date | Power Level (MW) | Run Time (min) |
|---|---|---|---|
| NRX - A2 | Sept 1964 | 1100 | 5 |
| NRX - A3 | April 1965 | 1100 | 17 |
| NRX - EST | March 1966 | 1100 | 28 |
| NRX - A5 | June 1966 | 1100 | 30 |
| NRX - A6 | Dec 1967 | 1100 | 60 |
| XE | Mar 1969 | 1100 | 10 |

VJV872.1

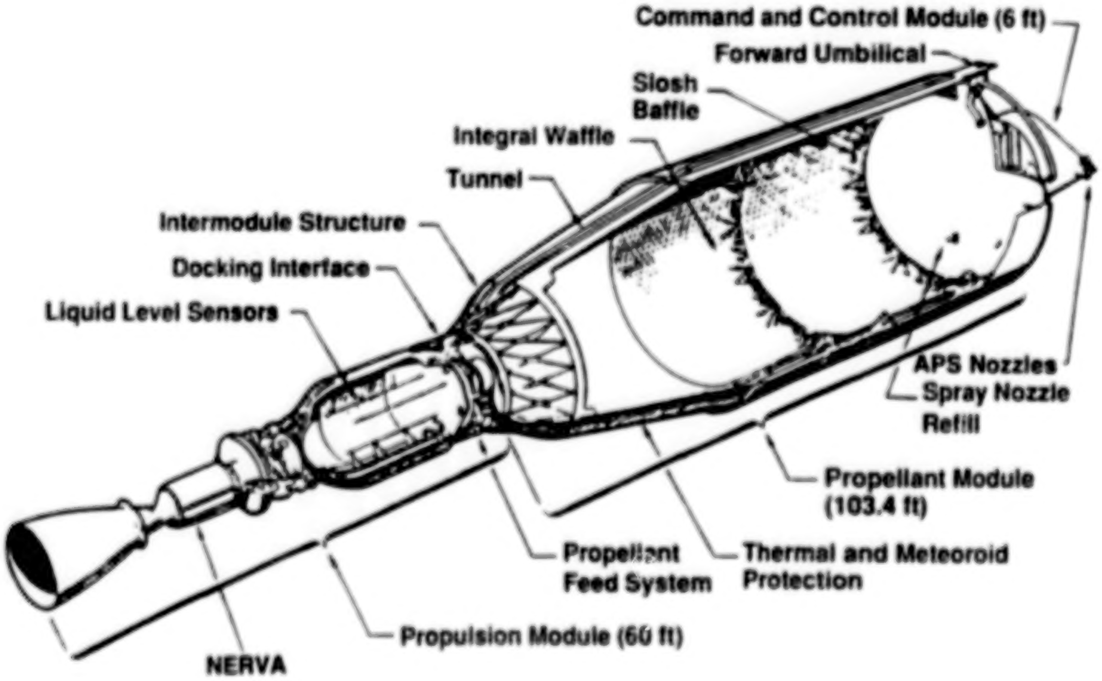
## CLASS 3 MULTI-MODULE RNS

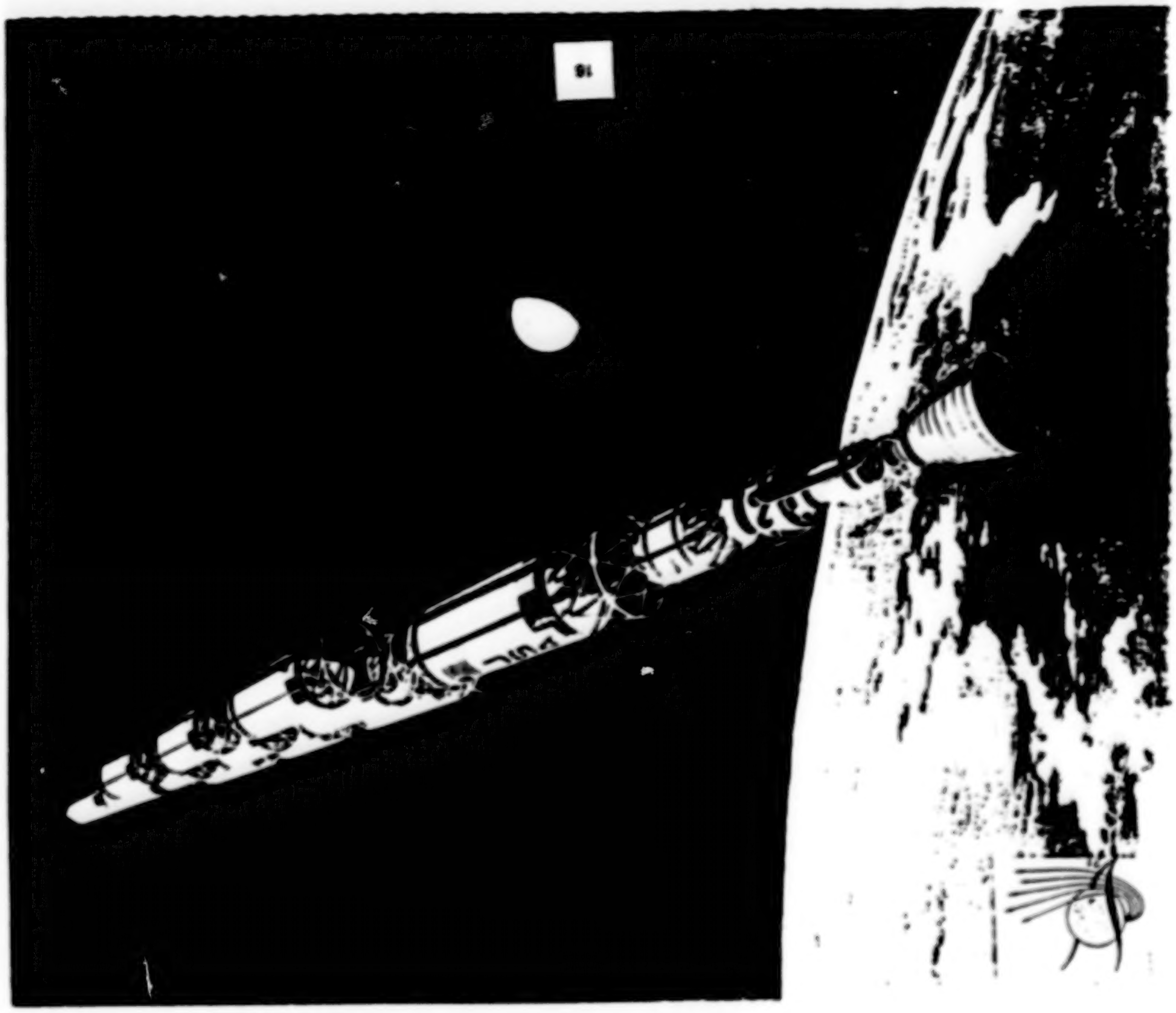

Command and Control Module (6 ft)
Thermal and Meteoroid Protection
Integral Waffle
Spray Nozzle Refill
Intermodule Structure
Propellant Feed System
APS Nozzles
Docking Interface
Liquid Level Sensors
Propellant Modules* (240 ft)
Inboard Transfer
Propulsion Module (60 ft)
NERVA

*Individual Modules 15 ft in Diameter by 60 ft Long

VJV873.1

## CLASS 1 SINGLE-MODULE HYBRID RNS


Command and Control Module (6 ft)
Forward Umbilical
Slosh Baffle
Integral Waffle
Tunnel
Intermodule Structure
Docking Interface
Liquid Level Sensors
APS Nozzles
Spray Nozzle Refill
Propellant Module (103.4 ft)
Propellant Feed System
Thermal and Meteoroid Protection
Propulsion Module (60 ft)
NERVA



# NUCLEAR STAGE DESIGN/PROGRAM CONSIDERATIONS

VJX042 M1CV

- ☐ Structure
- ☐ Propulsion
  - ● Auxiliary
  - ● Nuclear
- ☐ Electrical
  - ● Power
  - ● Guidance/Navigation
  - ● Control
  - ● Communications
- ☐ Thermal Control
- ☐ Radiation Protection
  - ● Configuration
  - ● Shielding
- ☐ Propellant Control
  - ● Pressurization
  - ● Chilldown

# NUCLEAR STAGE DESIGN/PROGRAM CONSIDERATIONS

(Continued)

VJX942 M1CV

- ❑ Weight/Mass Control
- ❑ Ground/Orbital Operations
  - ● Assembly
  - ● Maintenance
- ❑ Safety
- ❑ Reliability
- ❑ Manufacturing
- ❑ Test Plan
  - ● Component
  - ● Systems
  - ● Battleship
- ❑ Program Schedule
- ❑ Costs

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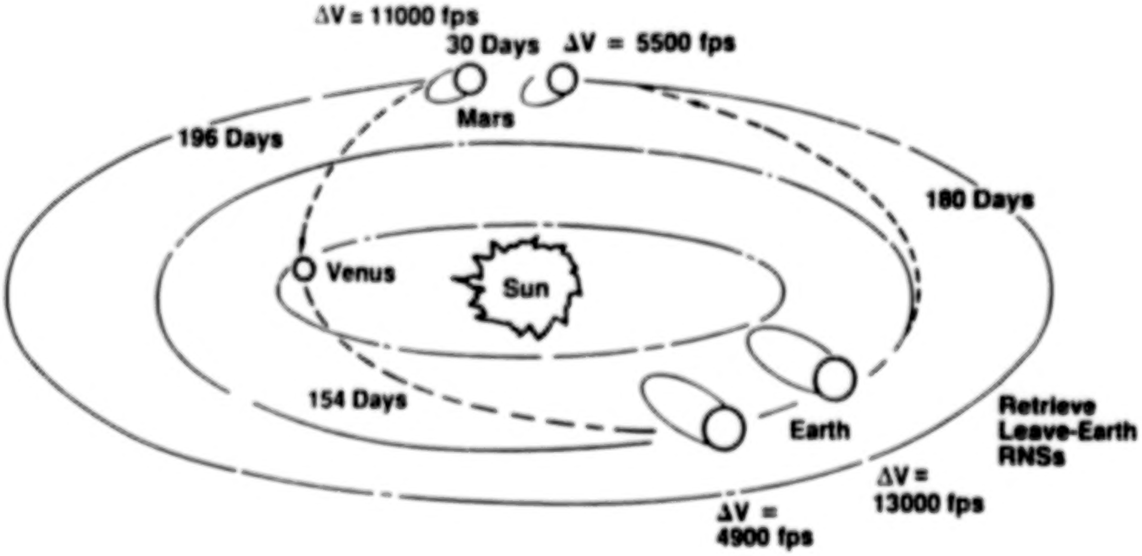
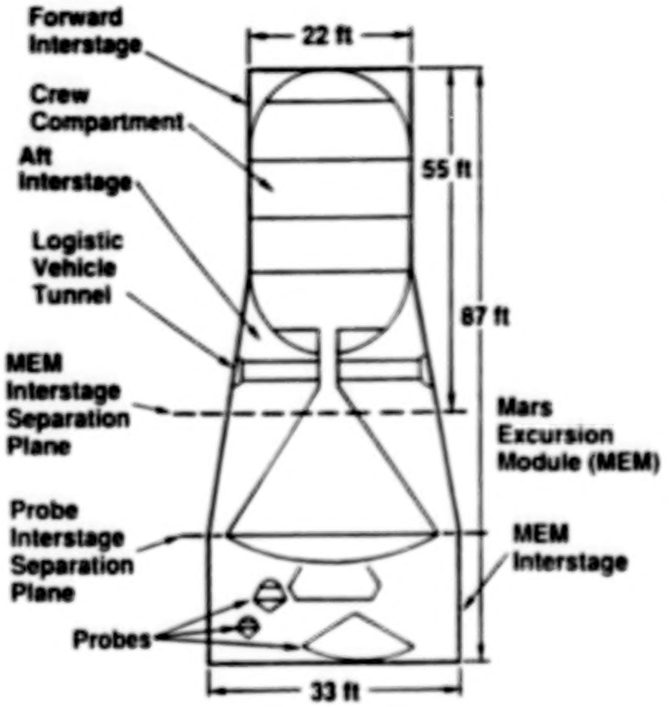
VJV877.1

# 1988 MARS LANDING PROFILE


ΔV = 11000 fps
30 Days
ΔV = 5500 fps
Mars
196 Days
180 Days
Venus
Sun
154 Days
Earth
Retrieve
Leave-Earth
RNSs
ΔV =
13000 fps
ΔV =
4900 fps

## MSFC MANNED MARS SPACECRAFT CONCEPT

VJV878.1


Forward Interstage
Crew Compartment
Aft Interstage
Logistic Vehicle Tunnel
MEM Interstage Separation Plane
Probe Interstage Separation Plane
Probes
22 ft
55 ft
87 ft
33 ft
Mars Excursion Module (MEM)
MEM Interstage

| | Weight (lb) |
|---|---|
| **Mission Module** | 82,900 |
| **Mars Excursion Module** | 95,290 |
| **Mars Probes** | 36,000 |
| **Venus Probes** | 4,000 |
| **Interstages** | 21,000 |
| **Total Spacecraft** | 239,190 |

## PROPELLANT REQUIREMENTS – BASELINE MODE

VJV879.1

- 262-nmi Departure Orbit
- 300,000-lb Capacity RNS

| RNS Number | | ΔV (1000 fps) | Wp (1000 lb) | A'Cool (1000 lb) | Wp (%) |
|---|---|---|---|---|---|
| 4 | Arrive - Earth | 4.9 | 36 | 2.5 | 13 |
| | *Midcourse | 0.7 | 10.2 | 0 | 3 |
| | Leave - Mars | 11.0 | 129.5 | 6.8 | 46 |
| | *Orbit Trim | 0.2 | 6.5 | 0 | 2 |
| | Arrive - Mars (No. 2) | 4.6 | 102.8 | 5.7 | 36 |
| 3 | Arrive - Mars (No. 1) | 0.9 | 25.3 | 0 | 8 |
| | *Midcourse | 0.4 | 21.0 | 0 | 7 |
| | Leave - Earth (No. 2) | 6.3 | 242.7 | 11.0 | 85 |
| 1.2 | Retrieval (Per RNS) | 6.7 | 26.4 | 4.0 | 10 |
| | Leave - Earth (No. 1, Per RNS) | 6.7 | 208.5 | 10.0 | 73 |

* $I_{SP}$ = 450 sec

Total Propellant Required = 1,097,800 lb

[illegible]

# LAUNCH CONSIDERATIONS FOR MANNED MARS
## CAPTURE AND LANDING MISSION (1988)

**Class 1**

- ☐ 4 stages, 1,097,800 lb $LH_2$
- ☐ Each stage weighs 69,245 lb dry
- ☐ Assume ALS 180,000 lb
- ☐ Mission requires 8 ALS launches
  - 4 launches of stage (70,000 lb) + 110,000 lb $LH_2$
  - 4 launches of 174,000 lb $LH_2$ (assume 6000-lb tank)

**Class 3**

- ☐ 4 stages (1 propulsion, 8 propellant, 1 command and control module)
  - Assume 1,335,000 lb $LH_2$
- ☐ Each stage weighs 85,000 lb dry

Requires:
- 4 launches for propulsion (30,475 lb) plus command and control module (4615 lb)
- 32 launches for propellant module (40,075 lb) (containing 34,000 lb $LH_2$)
- 6 launches 42,000 lb $LH_2$

Total 42 launches (Space Shuttle/Titan IV)

[illegible]

# NUCLEAR VERSUS CHEMICAL PERFORMANCE COMPARISON (SAME-STAGE TECHNOLOGY)

| Mission (Elliptic Capture Orbits) | OLV Booster | Number of Launches Saturn V | Mission | Payload | OLV/ Booster | Number of Launches Saturn V |
|---|---|---|---|---|---|---|
| Planetary Capture Missions | | | Planetary Flyby Missions-2 | | | |
| 1978 Venus capture | Advanced Chemical<br>Nuclear-restart | 5<br>3 | 1977 triple planet | 220,000 lb | S-IVC<br>Advanced chemical | 4<br>4* |
| 1980 Venus capture | Advanced Chemical<br>Nuclear restart | 5<br>3 | | | Nuclear<br>Nuclear-restart | 2<br>– |
| 1982 Mars capture | Advanced Chemical<br>Nuclear restart | –<br>4 | 1978 dual planet | 200,000 lb | S-IVC<br>Advanced chemical | 3<br>3* |
| | | | | | Nuclear<br>Nuclear-restart | 2<br>– |
| 1978 Mars capture | Advanced Chemical<br>Nuclear restart | 8<br>4 | 1976 dual planet, powered | 200,000 lb | S-IVC<br>Advanced chemical | 3<br>– |
| | | | | | Nuclear<br>Nuclear-restart | 2<br>– |

* Requires two launches for payload

# VISIONS OF THE FUTURE

[illegible]

- ☐ Deep space travel requires energy and delta velocity (ΔV) in particular
- ☐ Propulsion systems with $I_s > 1500$ sec are needed

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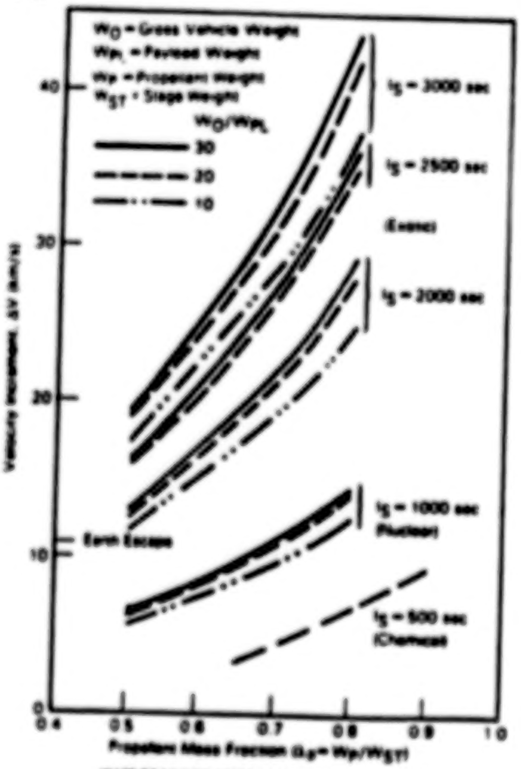
# PROPULSION SYSTEM ΔV CAPABILITIES

VJK178



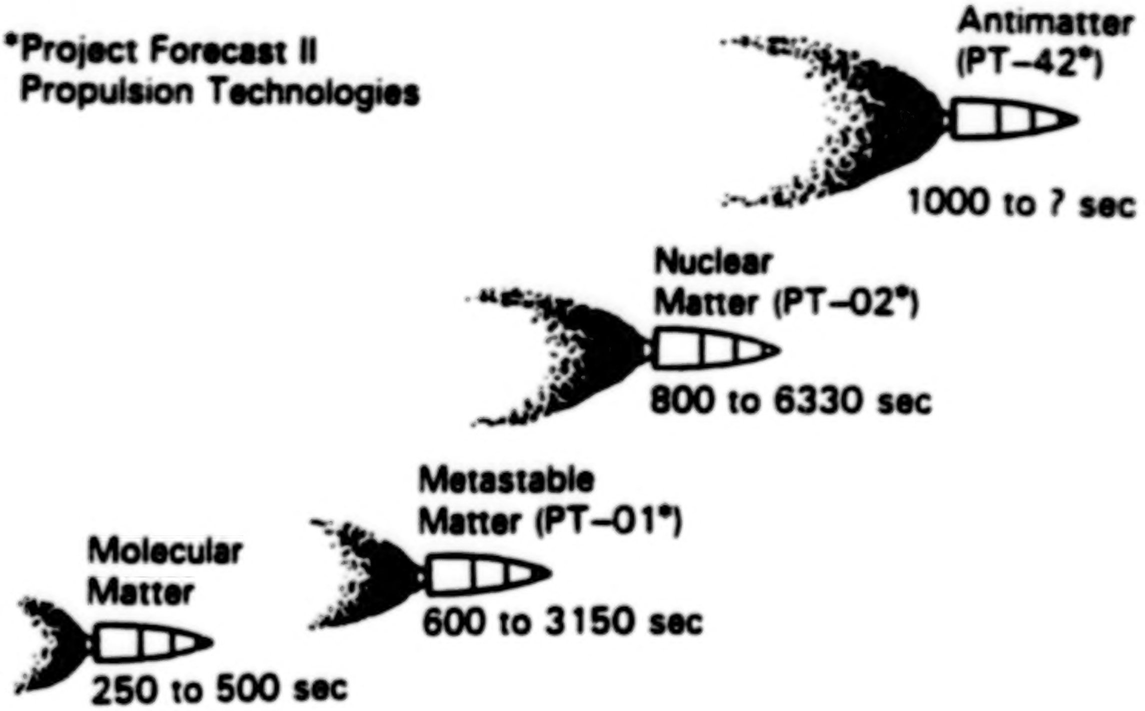
MASS FRACTION EFFECTS ON ΔV CAPABILITIES

- Chemical propulsion can barely achieve earth escape with a single-stage vehicle
- Higher $I_s$ propulsion systems substantially increase payload delivery capabilities
- New exotic propulsion systems need to be developed and made economical to produce, store and use

VJ8716

# SPECIFIC IMPULSES OF ROCKET FUELS

*Project Forecast II
Propulsion Technologies



Antimatter
(PT–42*)
1000 to ? sec

Nuclear
Matter (PT–02*)
800 to 6330 sec

Metastable
Matter (PT–01*)
600 to 3150 sec

Molecular
Matter
250 to 500 sec

26

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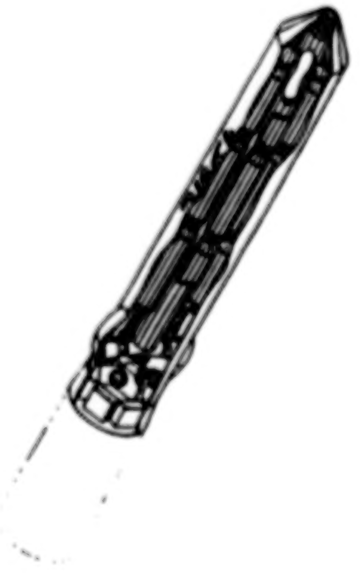
MDC H5034C
AIAA 89-2629
MAY 1989



## FUSION PROPULSION SYSTEMS

V. E. HALOULAKOS, MCDONNELL DOUGLAS

R. F. BOURQUE, GENERAL ATOMICS

Presented at
AIAA/SAE/ASME/ASEE
25th Joint Propulsion Conference
11-13 July 1989
Monterey, California

McDonnell Douglas Space Systems Company

MCDONNELL DOUGLAS

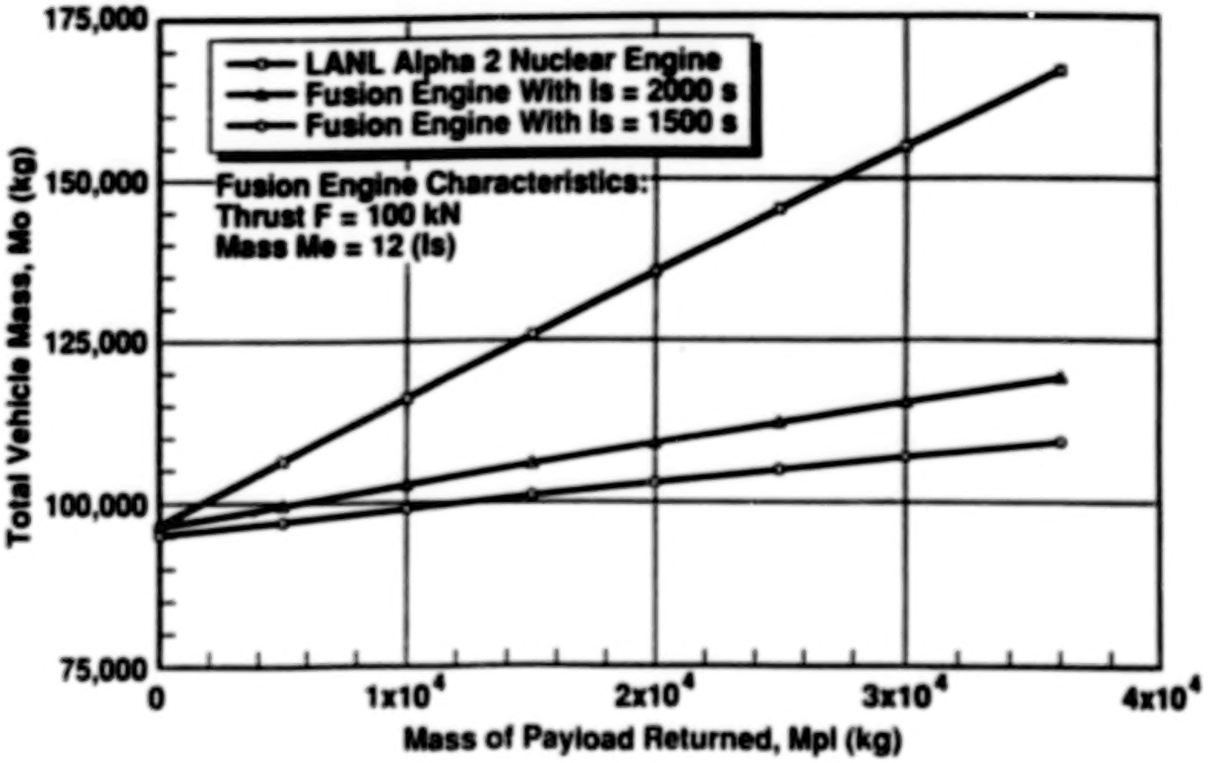
# SPACE TRANSFER VEHICLE COMPARATIVE DESIGN DATA

LEO - GEO - LEO Mission, Mpl = 36,000 kg ; Del V = 9 km/s;
Burn Time = 3675s (Constant)

| | | Chemical Cryogenic, 6 RL - 10's | Nuclear, 4 Alpha 2's | Fusion, Me=12 (ls) |
|---|---|---|---|---|
| **Rocket Engine** | | | | |
| Thrust | (kN) | 400 | 278 | 208 |
| Specific Impulse | (s) | 450 | 860 | 2500 |
| **Mass Breakdown** | | | | |
| Propellants | (kg) | 333,291 | 134,548 | 34,685 |
| Fuel | (LH$_2$) | 51,275 | | |
| Oxidizer | (LOX) | 282,015 | | |
| Propellant Tank | | | | |
| Total Volume | (m$^3$) | 970 | 1,937 | 499 |
| Mass | (kg) | 8,156 | 10,849 | 2,797 |
| Pressurization | | | | |
| Helium System | (kg) | 1,374 | 1,828 | 471 |
| Engine | (kg) | 792 | 10,270 | 30,000 |
| Miscellaneous | (kg) | 3,411 | 4,538 | 1,170 |
| **Total Vehicle Mass** | | 383,024 | 198,033 | 105,123 |

- ❐ The initial vehicle mass serves as the key criterion of optimization
- ❐ The higher specific impulse of the fusion system results in lower propellant and vehicle mass for the specified mission and payload

# FISSION AND FUSION ROTV COMPARISON
## TOTAL VEHICLE MASS vs. RETURNED PAYLOAD FROM GEO
Assumes Each Vehicle Has Delivered a 36,   -kg Payload to GEO

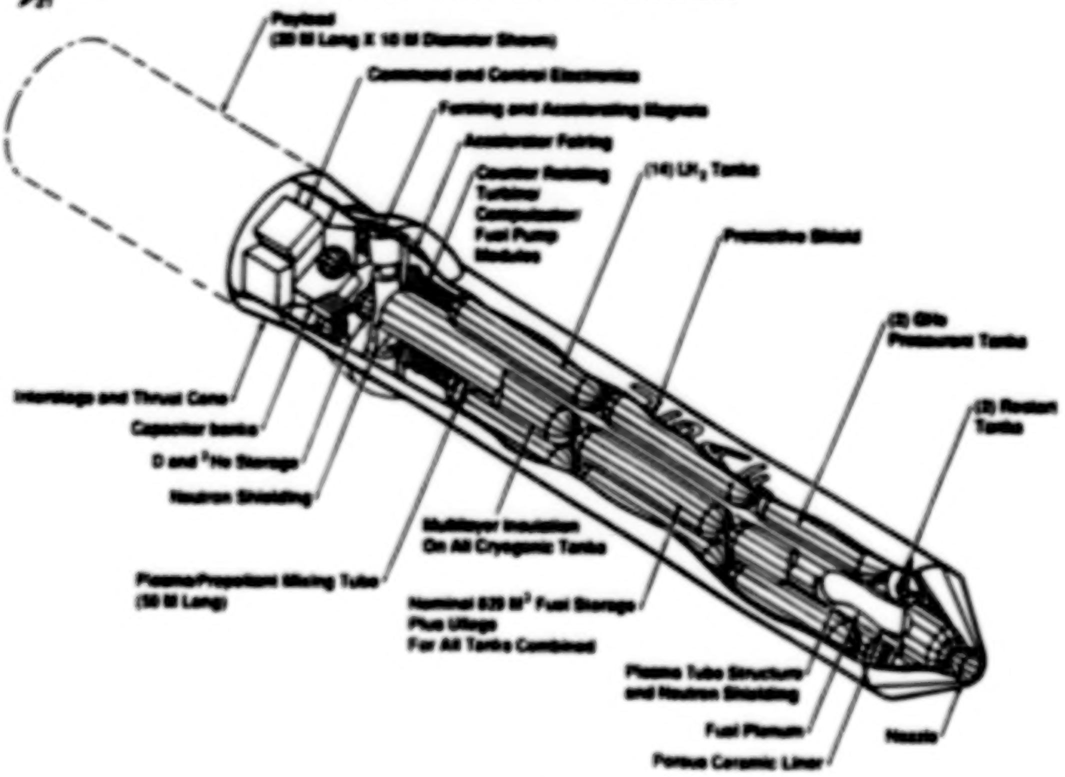


# MDC-GA FUSION SPACE TRANSFER VEHICLE SIZING DATA

**Mission:** Transfer 36,000 kg to Geosynchronous Orbit (GEO)
Return Vehicle Empty and with Original Payload
**Engine:** GA Fusion Engine
Is = 1500 s, Thrust = 100 kN, Mass = 12 (Is)

| Mass Breakdown | | LEO-GEO Empty Return | LEO-GEO Return Payload |
|---|---|---|---|
| Propellant | (kg) | 27,910 | 57,597 |
| Propellant tank | | | |
| Total volume | ($m^3$) | 402 | 829 |
| Mass | (kg) | 2,251 | 4,644 |
| Pressurization | (kg) | 379 | 783 |
| Engine | (kg) | 18,000 | 18,000 |
| Miscellaneous | (kg) | 941 | 1,942 |
| Total Vehicle Mass | (kg) | 98,070 | 118,967 |

# FUSION-PROPELLED REUSABLE ORBIT TRANSFER VEHICLE


Payload
(20 M Long X 10 M Diameter Shown)
Command and Control Electronics
Forming and Accelerating Magnets
Accelerator Fairing
Counter Rotating Turbines Compensator Fuel Pump Modules
(14) LH₂ Tanks
Protective Shield
(2) GHe Pressurant Tanks
(3) [illegible] Tanks
Interstage and Thrust Cone
Capacitor banks
D and ³He Storage
Neutron Shielding
Multilayer Insulation On All Cryogenic Tanks
Plasma/Propellant Mixing Tube (50 M Long)
Nominal 839 M³ Fuel Storage Plus Ullage For All Tanks Combined
Plasma Tube Structure and Neutron Shielding
Fuel Plenum
Porous Ceramic Liner
Nozzle

# MATTER/ANTIMATTER ANNIHILATION

☐ **Problems and promises**

- **Production:** USAF has priced a production facility at BNL for $10^{15}$ $\bar{p}$/yr at approximately $14 million (1989)
- **Containment:** Antiprotons have been stored for long periods at CERN and preliminary designs for transportable storage bottles have been proposed
- Antimatter use holds great promises for applications in
  - **Medicine:** Diagnosis and eradication of tumors
  - **Materials:** Location and cure of flaws, processing of composites, etc.
- This synergism should be explored and pursued

[illegible]

# FAR FUTURE PROPULSION SCHEMES

☐ **Matter/antimatter**

- Antimatter safely stored will deliver its energy, via beam technology, to a flowing propellant. Rates of energy delivery and propellant flowrate to control the level of the thrust to any desired level

☐ **Teleportation *"Beam me up, Scotty"***

- Matter destructuring (i.e., breakdown into particles), and restructuring to be first perfected on inanimate objects. Transportation to be done by beam technology

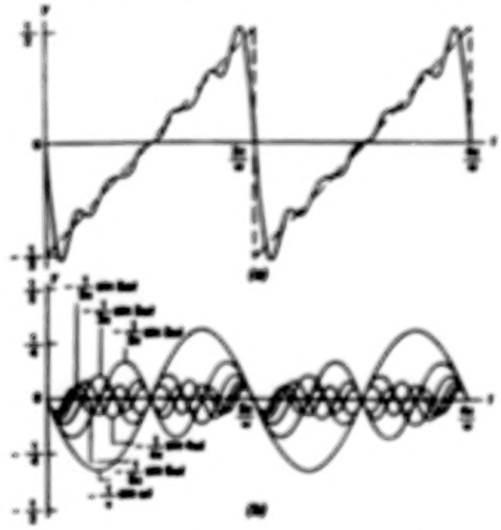
[illegible]

# FAR FUTURE PROPULSION SCHEMES (CONT)

☐ **Antigravity**

- Gravity waves will be fully characterized (Why not?)
- Each celestial body sends out waves whose amplitude and frequency are directly related to its mass and size
- The net gravity field at any point in space is the result of the gravity wave interference pattern
- Super sensors and supercomputers analyze this wave pattern and identify its basic components
- An "antigravity wave generator" will then generate waves of precisely the same amplitude and frequency but of opposite phase. Thus, by causing an exact destructive interference, it will precisely cancel out the gravity field
- A suitable propulsion system can then accelerate the vehicle to very high velocities with a rather low force and low energy expenditure

# WAVE INTERFERENCE



*(a)* The dashed line is a sawtooth "wave" commonly encountered in electronics. The Fourier series for this function is $y(t) = -\frac{1}{\pi}\sin \omega t - \frac{1}{2\pi}\sin 2\omega t - \frac{1}{3\pi}\sin 2\omega t - \cdots$.

The solid line is the sum of the first six terms of this series and can be seen to the approximate the sawtooth quite closely.

*(b)* Here we show the first six terms of the Fourier series which, when added together, yield the solid curve in *(a)*.

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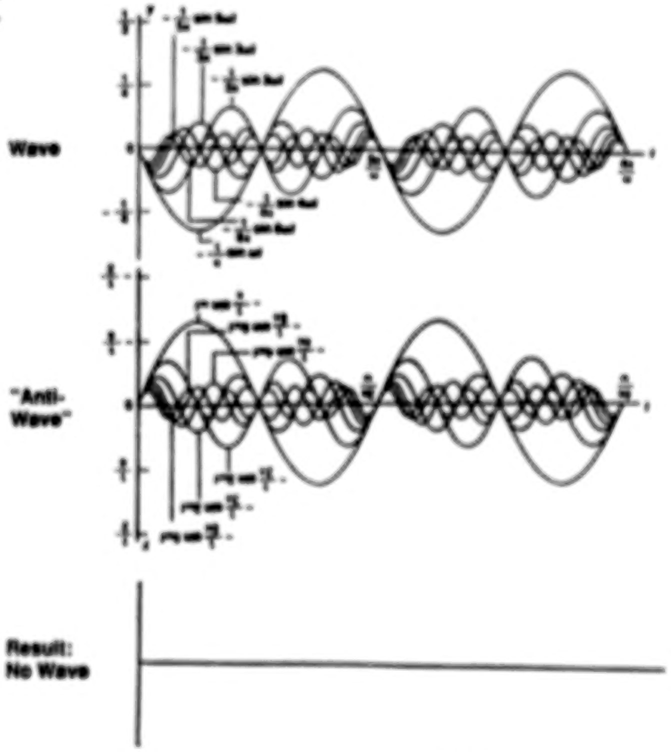
# WAVE DESTRUCTIVE INTERFERENCE



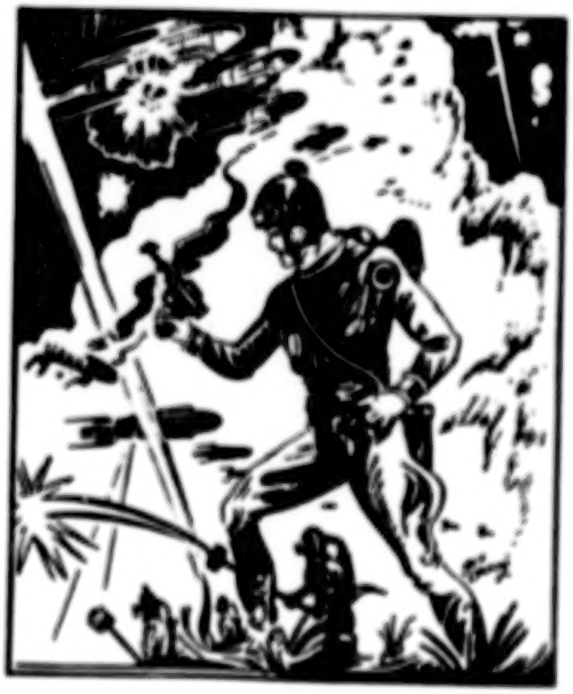
# SCIENCE FICTION?

***Yes!***

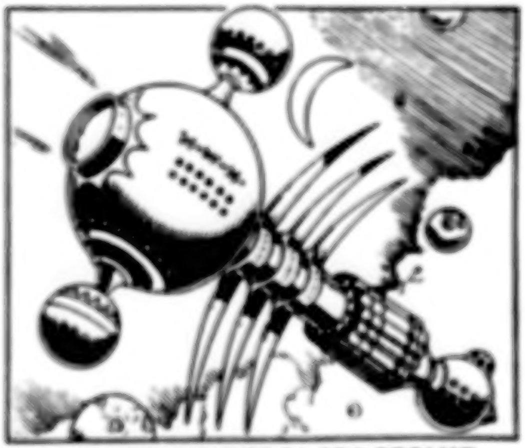
- ☐ **Science fiction of one time period is science fact of some later time**
- ☐ **Let us consider the following**

---

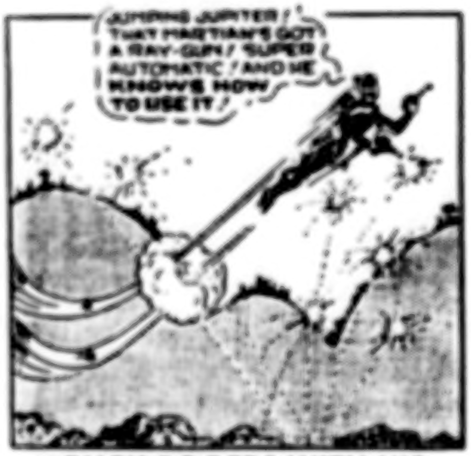
# BUCK ROGERS IN THE 25th CENTURY



# SCIENCE FICTION 1934



■ BUCK ROGERS' SPACESHIP


JUMPING JUPITER! THAT MARTIAN'S GOT A RAY-GUN! SUPER AUTOMATIC! AND HE KNOWS HOW TO USE IT!

■ BUCK ROGERS WITH HIS FLYING BELT

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OF POOR QUALITY

# SCIENCE FACT 1984

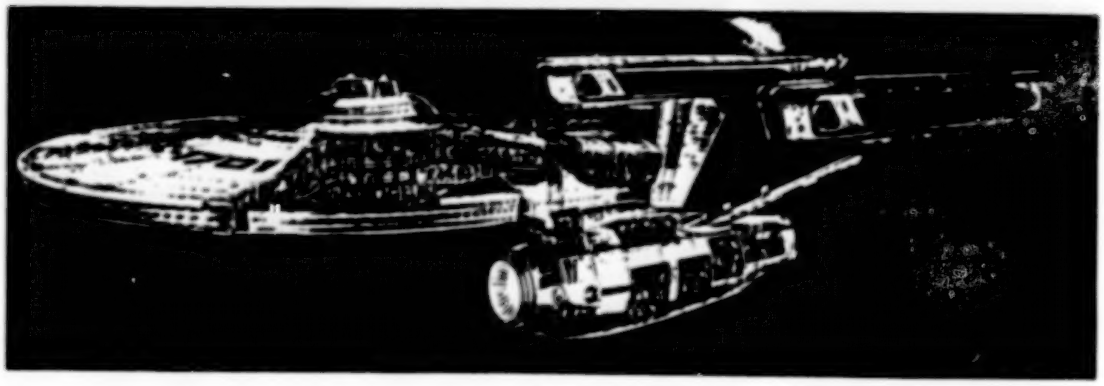
■ THE 25TH CENTURY ARRIVED IN 50 YRS

■ ASTRONAUT'S SPACE SHIP

■ NASA ASTRONAUT WITH HIS "FLYING BELT"

# "STAR TREK"
## SCIENCE FICTION 1966

- 23RD CENTURY STARSHIP ENTERPRISE POWERED BY AN ANTIMATTER REACTION CHAMBER CAN REACH SPEEDS UP TO WARP 8



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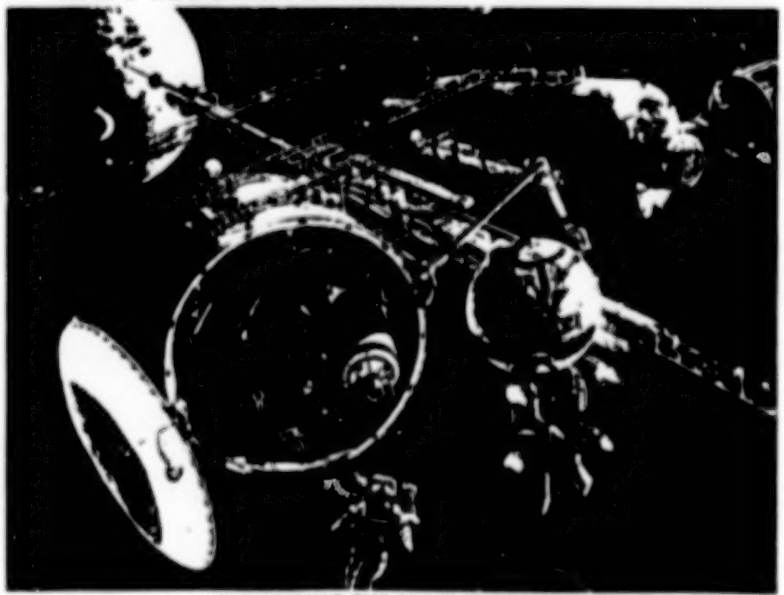
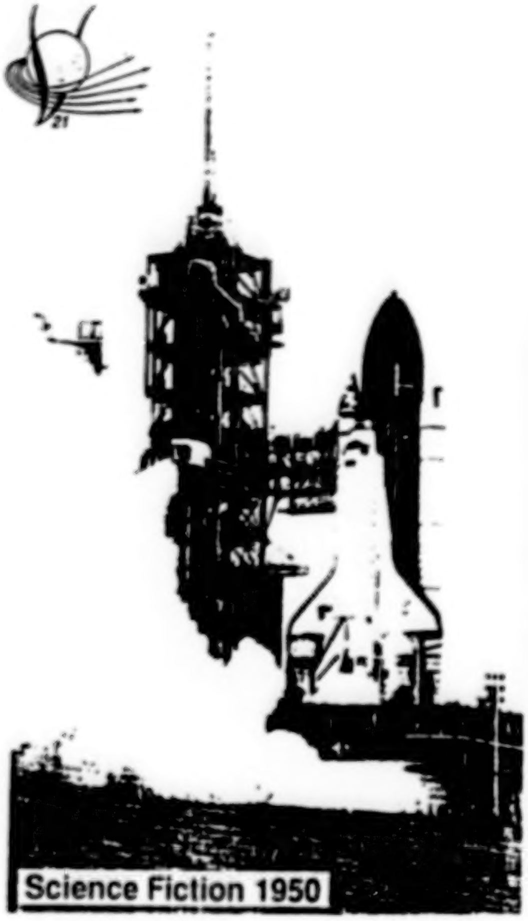
# SCIENCE FACT 2016?
## (50 YEARS LATER)

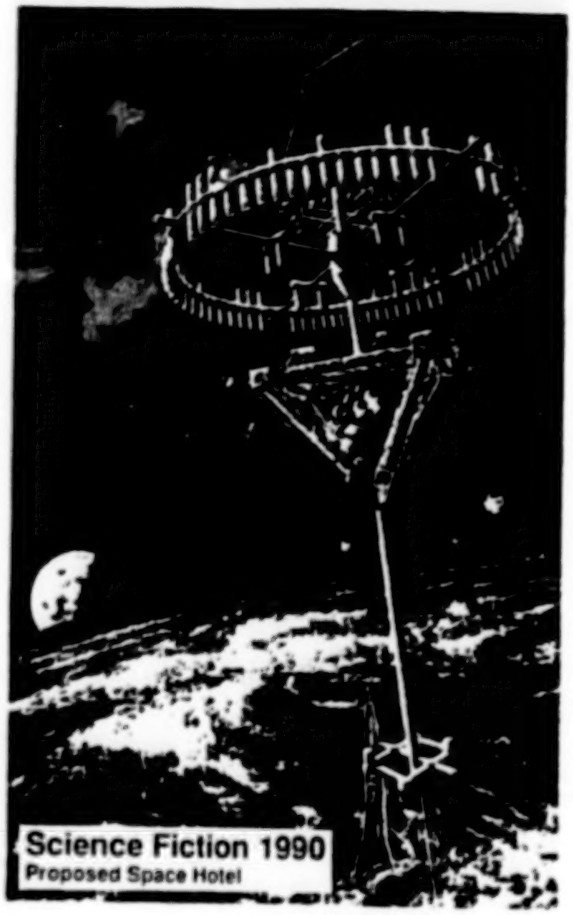
- WILL THE 23RD CENTURY HAVE ARRIVED?
- WILL STARSHIP ENTERPRISE AND STARBASE 12 HAVE BECOME REALITIES?



"PIONEERING THE SPACE FRONTIER"



Science Fiction 1950



Science Fiction 1990
Proposed Space Hotel

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[illegible]

## CAN WE DO IT?

- ❒ Recent history tells us: Yes!
- ❒ We went from Sputnik 1 to Apollo 11 in less than 12 years
- ❒ From the basic university lab atom splitting experiment (Berlin, Dec 1938) to Hiroshima and Nagasaki (Aug 1945) in 6.5 years
- ❒ We need a strong resolve, committment of resources, and dedication
- ❒ We should not be afraid of risk and failure

- ☐ **Some of the greatest advances and discoveries resulted from some apparent "failure" and setback**

  **Example: The negative results of the Michelson-Morley experiment resulted in the Theory of Relativity, $E = MC^2$ etc.**

  **The space exploration initiative, some times called "The Bush Push", will present us with unparalleled challenges and it will undoubtedly lead to even greater developments and discoveries**

***Go For It!***

## HISTORIC SCIENTIFIC MILESTONES

- ☐ 1905: $E = MC^2$
- ☐ 1938/39: Uranium Nucleus Split/Einstein writes to FDR
- ☐ 1945: Alamogordo – Hiroshima – Nagasaki (40 years later)
  - ● Notes: – Pressing need (WWII)
    – Manhattan Project, $$$, etc.
- ☐ 1955: – Antiproton is discovered
  – Antimatter becomes fact
- ☐ 1984: – Trapping and storage of antiprotons achieved
- ☐ 1995: ? ? ?
  (40 years later)

**SHOULD WE BE DOING SOMETHING?**

# FAMOUS PRONOUNCEMENTS

☐ "Heavier than air flying machines are impossible"
Lord Kelvin, President, Royal Society, 1895

☐ "Everything that can be invented has already been invented"
Charles H. Duell, Director of U.S. Patent Office, 1899

☐ "There is no likelihood man can ever tap the power of the atom"
Robert A. Millikan, Nobel Prize in Physics, 1923

☐ "Who the hell wants to hear actors talk"
Harry M. Warner, Warner Bros. Pictures, 1927

# LASER-BOOSTED LIGHTCRAFT TECHNOLOGY DEMONSTRATOR[*]

N91-22156

J. C. Richard,[†] C. Morales,[‡] W. L. Smith,[§] and L. N. Myrabo[¶]

Rensselaer Polytechnic Institute
Troy, NY 12181

## Abstract

The detailed description and performance analysis of a 1.4 meter diameter Lightcraft Technology Demonstrator (LTD) is presented. The novel launch system employs a 100 MW-class ground-based laser to transmit power directly to an advanced combined-cycle engine that propels the 120 kg LTD to orbit — with a mass ratio of two. The single-stage-to-orbit (SSTO) LTD machine then becomes an autonomous sensor satellite that can deliver precise, high quality information typical of today's large orbital platforms.

The dominant motivation behind this study is to provide an example of how laser propulsion and its low launch costs can induce a comparable order-of-magnitude reduction in sensor satellite packaging costs. The issue is simply one of production technology for future, survivable SSTO aerospace vehicles that intimately share both laser propulsion engine and satellite functional hardware.

## Introduction

In order for laser propulsion to enable a significant reduction in the cost of certain critical space systems, both launch and payload costs must be reduced by (at least) an order of magnitude. Canavan[1] was first to bring this fact to light, noting that a reduction in either category alone would have much less economic impact. This conclusion emerged from recent in-depth examinations of the economics for laser propulsion deployment of sensors, interceptors and decoys.[1,2] Furthermore, Canavan affirms that the minimum effective system must be able to launch 60–100 kg payloads. After evaluating cost projections, he concludes that a system designed for payloads smaller than this could increase costs significantly, reducing laser propulsion's margin with respect to both the threat and conventional chemical rocket alternatives.[1]

Apparently, the true costs of building and launching today's large satellite platforms are not widely known, as discussed in a recent *Aerospace America* article.[3] For example, sensor hardware can cost upwards of $200,000/kg and to put that sensor into geostationary orbit (GEO) typically requires $10,000/kg of payload. Flying the Shuttle Orbiter to a 250 km low Earth orbit may cost $3000/lb, but the actual price is really almost twice that amount because the shuttle itself is not amortized against the payload.[4] The projected launch cost goal for the Advanced Launch System (ALS) is roughly $300/lb to LEO. Hence, for laser propulsion to play a significant role in boosting future critical space systems, launch costs must fall to $100/kg or below.

Canavan has raised the fascinating issue of whether and how laser propulsion and its low launch costs could induce a substantial reduction in satellite package costs.[1] In the opinion of the authors, such reductions could be facilitated by an exceptionally close integration of the laser propulsive engine and satellite functional hardware. Pushed to the extreme, almost every vehicle component could be designed to serve multiple functions, in both transatmospheric and orbital flight modes.

Clearly, the ultimate configuration of any laser-boosted machine will be strongly driven, if not entirely dominated, by the mission it must perform: be it interceptor, decoy or sensor. A near infinite number of successful configurations could be alleged to exist, but it is most instructive to select a *specific mission*, and then to explore a single configuration from the initial design concept, through the preliminary engineering design process.

The advanced aerospace vehicle considered here is exemplary of a class of sensor machines that can be derived largely from an intimate integration of propulsion and sensor systems. The proposed design exploits the inherent advantages of advanced beamed-energy sources (i.e., high power lasers) and innovative combined-cycle (airbreathing/rocket) engines to accomplish this goal. The authors believe that this unique approach could ultimately enable a reduction in launch and sensor package costs by one or two orders of magnitude below present levels. However, as pointed out by Canavan,[1] the numbers of these sensor satellites may not be great enough to justify the expense of the entire laser launch facility for this application alone. Nevertheless, the launch facility is likely to be amortized over a great number of users and dissimilar mission applications anyway.

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[*] This research was sponsored by Lawrence Livermore National Laboratory, Subcontract No. 2073803, under the Laser Propulsion Program of the SDIO.

[†] Graduate student, Aeronautical Engineering.

[‡] Undergraduate student, Mechanical Engineering.

[§] Graduate student, Mechanical Engineering.

[¶] Assistant Professor, Mechanical Engineering.

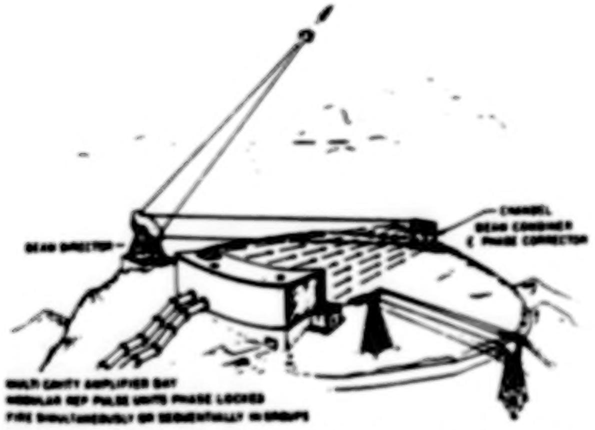
BEAM DIRECTOR
CHANDEL BEAM COMBINER & PHASE CORRECTOR
MULTI CAVITY AMPLIFIER BAY
MODULAR REP PULSE UNITS PHASE LOCKED
FIRE SIMULTANEOUSLY OR SEQUENTIALLY IN GROUPS

Figure 1: Ground Based $CO_2$ Laser System [After Kantrowitz]

## Ground-based Laser Launch Facility

Portrayed in Fig. 1 is a 100 MW-class ground based laser (GBL) that could be built with existing closed-cycle $CO_2$ laser technology using E-beam pumping. A powerful $CO_2$ electric laser could be fabricated in the next 4-5 years by assembling numerous (e.g., 100) smaller units (e.g., 1-3 MW each) into a large array. Redundant units would be built into the system so that inoperative units could be dropped out, with no loss in system utility. As shown in Fig. 1, a Chandel-type beam combiner could then be invoked to link all the output beams together.

All the units could be fired *simultaneously* to give the lowest pulse repetition frequency (PRF) of perhaps several hundred Hertz at the highest pulse energy ($E_p$); at the other extreme, each unit could be triggered *sequentially* to yield the lowest $E_p$'s and PRF's up to 10 kHz. Using E-beam pumping, laser pulse durations ($t_p$) can be limited to the range 0.1 to 1.0 $\mu$sec which smaller, near-term pulsed laser propulsion engines will require.

Configured in this manner, the GBL facility can be programmed to deliver a complicated pulse train sequence of PRF's, $E_p$'s and $t_p$'s with the utmost of ease. This pulse train can be calculated to exactly match what a laser propulsion engine will need along a given launch trajectory — i.e., normally a direct function of flight Mach number and altitude. The goal is to produce an efficient thruster without introducing too much flight hardware, which has added so much to the cost of chemical rockets, as Kantrowitz has observed.[4]

The essential point of this advanced launch scheme is to place as much of the system complexity as possible *on the ground* (no weight penalty here!), where it can be serviced easily. With this approach, laser-powered thrusters can be reduced to their simplest, and most reliable configuration.

The "straw man" GBL facility suggested here is set at the 250 MW level which is adequate to launch a 120 kg (dry mass), 1.4 meter diameter Lightcraft Technology Demonstrator (LTD) to low Earth orbit. The range of laser pulse energies required by its combined-cycle (airbreathing/rocket) engine is 40 KJ $< E_p <$ 70 KJ; PRF varies from 200 Hz to 10 KHz, and $t_p$ varies from 0.3 to 0.4 $\mu$sec — depending on the exact trajectory (i.e., Mach number vs. altitude) flown to orbit. With these parameters, the peak flux across the 1.0 m diameter LTD primary optic will fall in the range of 13.0 to 30.0 MW/cm$^2$.

This large GBL facility is a near term reality that exploits fifteen year old $CO_2$ laser technology. Kantrowitz[5] notes that the important costs for the $CO_2$ GBL installation, are for capital and operating expenses (which might add another 20% of the capital cost per year). Refer to Ref. [5] for an in-depth accounting of the economics for a GBL launch facility. However, for the long term, the Free Electron Laser (FEL) is the favored future system due to its promise of high electric-to-laser conversion efficiency and reliability.

## Aerospace Vehicle Concept

Kantrowitz[5] has noted that to make laser propulsion a serious contender for space transportation to LEO, we will need to develop propellants that can achieve high thruster efficiency at low incident laser flux levels. It is also apparent that atmospheric transmission problems must be considered, especially in the immediate vicinity of the vehicle where the beam must propagate unhampered through the thruster's rapidly expanding, and potentially absorbing, exhaust.

Adaptive transmitter optics can be invoked to successfully bring the power beam up through the atmosphere. The 10 m diameter beam-director mirror suggested by Kantrowitz[5] would allow a 10 $\mu$m beam to be focused on a one meter diameter vehicle base, out to a range of about 800 km. This performance is, of course, close to the diffraction limit.

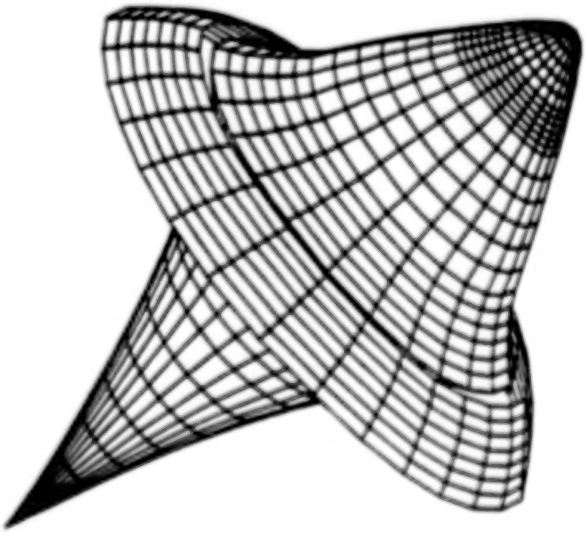

Figure 2: Lightcraft Technology Demonstrator (1.4 meter diameter)

It is the opinion of the authors that focusing mirrors mounted on the laser propelled vehicle are the most expedient way to permit both low flux levels in the atmospheric transmission link and elevated flux levels within the thruster (which are required for high propulsive efficiency). The reference "point design" for the Lightcraft Technology Demonstrator is configured around a 1.0 meter diameter parabolic mirror, as shown in Fig. 2. This primary receptive optical surface comprises the entire vehicle afterbody, which also serves as a plug nozzle for the laser-heated thruster.

In a related laser propulsion design effort at RPI last Fall (1987), The Apollo Lightcraft Project,[6,7] it was discovered that such a primary optic (PO) could also serve as a *telescopic lens* when not being used in the propulsive mode. Likewise, the LTD's 1.0 meter mirror could be "plugged-in" to the sensor function once it is in orbit. For example, it is known that optics in the 0.5 to 1.0 meter diameter range could well serve the purposes of mid-course tracking satellites positioned at altitudes of 1000 to 2000 km. Smaller PO's (e.g., 0.33 m) would be more suitable for reconnaissance satellites located in 200 to 500 km orbits.

This leaves the remaining problems of propagating the beam through the propulsive engine's exhaust, and of keeping the receptive PO mirror from melting down. As will soon become apparent, the two problems are found to be clearly interrelated. Most of today's high power $CO_2$ laser optics used under laboratory conditions are found to be water-cooled copper mirrors, sometimes silver coated, with a reflectivity of 99% (at best). These mirrors are heavy and the technology is not easily adaptable to the kind of lightweight design required for flight applications.

On the other hand, cryogenic-cooled, diamond-turned aluminum mirrors look especially promising. The 1.0 m diameter LTD mirror could easily be cut on Lawrence Livermore National Lab's LODTM, a state-of-the-art Large Optics Diamond Turning Machine. This lightweight, large optic technology is being developed for SDI applications, and reflectivities of uncoated surfaces have been pushed to 98% — which is close to the theoretical limit for aluminum. Various silver coating processes have been applied to improve the reflectivity of other high temperature faceplates such as titanium, molybdenum, beryllium/copper alloys (or perhaps stainless steel), but they cannot be expected to survive the abusive thermomechanical environment within a pulsed laser propulsion engine.

Hence, a cryogenic or water-cooled 3000–5000 series aluminum mirror appears to be the best candidate. A transparent, hard, protective coating such as aluminum oxide, silicon oxide, or vapor-deposited diamond must be applied over the bare aluminum to serve as an oxygen barrier. Water is an excellent coolant, its heat transfer characteristics are well known, and sacrificial coolant could be vented as high pressure steam to cool the mirror during engine operation.

However, $CO_2$ laser radiation is heavily absorbed by water vapor, so a dense steam wake behind the craft may prevent proper transmission and focusing of the beam into the LTD's engine. An alternative would be to utilize isotropic $CO_2$ in the GBL to shift the spectrum away from $H_2O$ vapor absorption lines; for the future free electron lasers, wavelength could be independently chosen to deliberately avoid the $H_2O$ vapor issue. Another problem with the use of $H_2O$ as a coolant is that condensed water droplets have been known to trigger localized damage on coated high power optics irradiated with pulsed beams.

In the rocket mode, the coolant will also serve as the reaction propellant, and regeneratively cool the rocket nozzle (mirror) along with other hot components. Clearly, one desires a propellant with the lowest molecular weight, such as $H_2$. Liquid hydrogen, unfortunately, has such a high specific volume that only 14 kg of $LH_2$ could be stored in the LTD's 70 cm diameter fuel tank. This is more than a factor of 10 too small.

For a number of reasons, then, $LN_2$ appears to be an ideal propellant for use in the LTD, and its specific gravity is 80.8% that of water. Heated to high temperature and pressure, nitrogen should produce specific impulses ($I_{sp}$) in the range of 725 to 1025 seconds — depending on whether the ejected gas is dissociated or not. This should produce thrust coupling coefficients (CC) of roughly 100 N/MW for a 40% efficient conversion process. Note that CC, $\eta_{th}$ and $I_{sp}$ are related by:

$$CC = \frac{2\eta_{th}}{g I_{sp}} \times 10^6, \quad (\mathrm{N/MW}) \tag{1}$$

where $g$ is acceleration due to gravity (9.8 $m/s^2$), $I_{sp}$ is in seconds, $\eta_{th}$ is the thruster efficiency and CC is in N/MW.

$LN_2$ is also favored because it is an inert and exceptionally clean propellant (especially important for high power optics), whose physical properties closely match those of air, which is 78% $N_2$ by volume. Hence, one would also expect that a combined-cycle pulsed laser propulsion engine would have little difficulty in shifting modes from *airbreathing* to rocket operation.

With regard to pulsed laser damage thresholds and active cooling limits of aluminum mirrors, the following comments can be made. The current time-average maximum absorbed flux limit for water-cooled copper mirrors with good optical figures is 200 $W/cm^2$. The single pulse safe operating limits (S.O.L.) for high power laser mirrors (see Ref. [8]) are:

$$\mathrm{S.O.L.} = \frac{F_{\perp} \sin\phi}{\sqrt{t_p}} \leq 10^4 \frac{\mathrm{J}}{\mathrm{cm^2 sec^{\frac{1}{2}}}} \tag{2}$$

where $F_{\perp}$ is the perpendicular incident fluence, $t_p$ is the laser pulse duration, and $\phi$ is the inclination of the surface to the incident laser beam. This operational limit assumes a factor of 10× safety margin below the actual catastrophic damage limit of $10^5$ $J/(cm^2 sec^{\frac{1}{2}})$.

For the LTD parabolic mirror shown in Fig. 2, note that the surface is inclined at the greatest angle along the beam centerline, where the laser flux is highest. For the sake of argument, assume the intensity distribution is gaussian, rather than "top hat" or constant. Most of the laser flux is received upon the outer annular portion of the mirror surface, which is inclined at an angle of 30° to the axis of symmetry. The conical tip has a half angle of 15°.

Assuming a maximum pulse energy of 70 kJ delivered in 0.3 $\mu$sec over the 7854 cm$^2$ (perpendicular area) LTD optic:

$$30^\circ \Longrightarrow \frac{F_\perp \sin\phi}{\sqrt{t_p}} = \frac{70{,}000 \times \sin 30^\circ}{7854 \times (0.3 \times 10^{-6})^{\frac{1}{2}}} = 8136 \; \frac{\mathrm{J}}{\mathrm{cm^2 sec^{\frac{1}{2}}}}$$

This is clearly below the $10^4$ J/(cm$^2$sec$^{\frac{1}{2}}$) maximum suggested operational limit. Note that the cone tip sees only 4212 J/(cm$^2$sec$^{\frac{1}{2}}$).

The breakdown fluence for aluminum surfaces is known to be roughly 10 J/cm$^2$. Beyond this limit, a plasma will be ignited on aluminum mirrors. Note that 70 kJ applied uniformly over a perpendicular surface of 7854 cm$^2$, results in a fluence of 8.9 J/cm$^2$. However, since most of the LTD's primary optic will be at 30°, this fluence is actually spread over twice the surface area, yielding 4.45 J/cm$^2$. Hence, plasma ignition on the aluminum does not appear to be a problem, except perhaps for the pointed tip of the parabolic mirror.

The absorbed heat flux (AHF) into the mirror is a function of the reflectivity $\rho$, local time-average beam intensity $P_{ave}$, and mirror inclination $\phi$.

$$\mathrm{A.H.F} = \left(\frac{P_{ave}}{A_\perp}\right) \sin\phi (1-\rho) \qquad (3)$$

Assuming a time-average laser power of 250 MW with a "top hat" intensity distribution across a 1 m diameter mirror with 98% reflectivity, one finds:

$$30^\circ \Longrightarrow \frac{(250 \times 10^6\ \mathrm{W})(\sin 30^\circ)(0.02)}{7854\ \mathrm{cm^2}} = 318\ \frac{\mathrm{W}}{\mathrm{cm^2}}$$

For the 15° cone tip, 165 W/cm$^2$ is absorbed.

Note that the absorbed flux on a 30° surface does exceed the 200 W/cm$^2$ limit for maintaining a good optical figure on the mirror. Although a distortion of several wavelengths is certain to occur, this *crude* figure will still be acceptable for the laser boost — since the focal length is only 60 cm, and the focal ring width is 1 cm. Once the vehicle is in space and the propulsive beam is shut off, the ideal mirror surface would relax to a high quality optical figure needed for an imaging telescope. The goal would be to avoid permanently bending/distorting the lens during the laser boost. Hysteresis effects under these flux conditions must be studied.

Three major issues still need to be investigated: (i) the scattering characteristics of diamond-turned aluminum mirrors at incidence angles of 15° to 45°, (ii) plasma breakdown threshold vs. mirror angle for grazing-incidence mirrors (avoid problem angles of 5° to 10°); and (iii) frost condensation on LN$_2$ cooled mirror surfaces.

Of the three issues, the last is potentially the most serious. During the laser boost "startup transient," the laser must be turned on first, then the LN$_2$ coolant valve would be opened. The objective is to prevent any part of the mirror surface from cooling below the dew point. If frost does form on the mirror it cannot perform its focusing function. Also, an ice layer just a few wavelengths thick will trigger full absorption of the incident laser pulse — potentially damaging the optic.

Finally, it should be noted that the harsh vibration and hypersonic aerothermodynamic environment which transatmospheric pulsed laser propulsion will subject onto the satellite sensor systems during the boost to orbit will necessarily make them highly survivable (by design).

## Engine/Vehicle Description

The LTD has a dry mass of 120 kg and is filled with 180 kg of LN$_2$ just prior to launch. A compressed-gas or steam cannon ejects the LTD with an initial velocity of 100-200 m/sec. One half second later, the GBL directs laser power to the vehicle's combined-cycle engine, and an airbreathing propulsion mode accelerates the LTD to Mach 7 and 120 kft. Thereafter, a laser-powered rocket mode (specific impulse ≥ 875 sec and CC = 100 N/MW) inserts the LTD into orbit, with 30 kg of liquid nitrogen (ullage) still remaining in the tank. This single-stage-to-orbit vehicle has a mass ratio of 2.0, and the entire final mass of 150 kg (i.e., 120 kg dry mass plus 30 kg of LN$_2$ ullage) becomes the payload.

Once in orbit, the LTD functions as an autonomous sensor satellite that is able to "look out" through its 1 meter diameter primary optic "eye." If this parabolic mirror is polished to optical quality, its resolution limit at 400 $\mu$m is 8 cm from an altitude of 180 km, or 17 cm from 360 km (see Table 1). Soon after arriving in orbit, the remaining LN$_2$ evaporates, thereby cooling down the vehicle, while pressurizing the propellant tank to 4000 psi. This compressed, cold, gaseous N$_2$ supply is then utilized by a simple 3-axis attitude control system to accomplish fine pointing required for sensor satellite functions throughout its life-

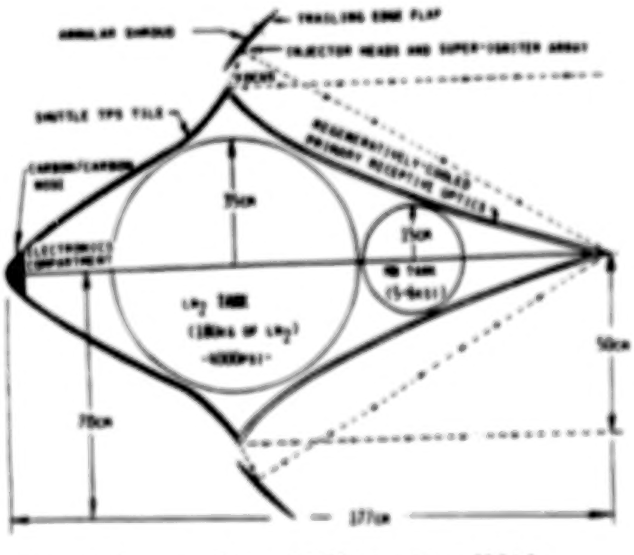

Figure 3: Cutaway View of LTD Aerospace Vehicle

| OPTICAL PARAMETER | WAVELENGTH, λ | | | |
|---|---|---|---|---|
| | 400nm | 700nm | 10.6µ | 1cm |
| $\delta\theta$ | $4.88\times10^{-7}$ | $8.54\times10^{-7}$ | $12.93\times10^{-6}$ | $12.2\times10^{-3}$ |
| $\delta(r)^\circ$ | $29.28\times10^{-8}$ | $512.4\times10^{-9}$ | $7.76\times10^{-6}$ | $7.32\times10^{-3}$ |
| δ(36,000 km) | 17.57m | 30.74 | 465.48 | $439.2\times10^{3}$ |
| δ(360 km) | 0.17m | 0.30 | 4.65 | $4.39\times10^{3}$ |
| δ(180 km) | 0.08m | 0.15 | 2.33 | $2.19\times10^{3}$ |

* FOCAL LENGTH = 60cm (ANGLES IN RADIANS, LENGTHS IN METERS)

Table 1: Resolution Limits for 1.0 m Diameter Mirrors

time. Of the 120 kg dry mass, only 18 kg is reserved for the sophisticated microcircuitry servos, star-tracker, etc. that control both laser-launch and satellite functions. Considering the recent developments in lightsats (Ref. [3]) and microsats, this mass allotment should pose no major problem.

As presently conceived, the LTD vehicle pictured in Fig. 3 could be constructed using state-of-the-art components derived from current liquid chemical rocket engine technology, advanced composite structures, and high power laser optics developed for SDI applications. As mentioned earlier, the regeneratively cooled, 1 m diameter primary optic is to be fabricated from aluminum using diamond-turned mirror technology. Since the reflectivity will be 98% at best, roughly 2% of the incident laser power must be carried away by sacrificial $LN_2$ coolant during the airbreathing boost mode. In the rocket mode, however, the primary optic will also serve as a plug nozzle; hence, the parallel with liquid chemical rocket technology becomes obvious.

A cutaway view in Fig. 3 shows the internal arrangement of hardware components for the LTD machine. The projected mass breakdown for the vehicle is given in Table 2. Note that the two most massive components are the 70 cm diameter $LN_2$ tank and the actively cooled 1 m diameter primary optic. The $LN_2$ tank is a filament wound pressure vessel similar to the ones made by Brunswick Defense in Lincoln, Nebraska. Brunswick makes a 26 in. diameter aluminum-lined spherical Kevlar tank which has a maximum operating pressure of 4,000 psig at 200°F, and a burst pressure of 6,000 psig. The pressure vessel is designed for storage of helium and nitrogen. It has an empty weight of 73.5 lb (33.33 kg).

## Primary Receptive Optic

Figure 4 displays the range of primary optic (PO) contours considered for the LTD machine. These contours are generated by rotating a parabola about an off-center axis. Note that the annular shroud airfoil has a circular arc cross-section with a flat bottom, and a chord ($C$) length of 22.5 cm. Five PO contours are displayed with focal rings positioned at 16.67%$C$, 25.0%$C$, 33.33%$C$, 41.67%$C$, and 50.0%$C$. The longest PO afterbody has its focus at the 16.67% chord point. As will become evident shortly, one wants to place the focus as far forward on the shroud as possible, yet at the same time minimize the PO weight penalty. For the LTD machine, this compromise was reached by placing the focus at 33.33%$C$. Note that the PO contours in Fig. 4 bear a striking resemblance to isentropic spike rocket engine nozzles.

Graham and Bergman were the first to build plug nozzle rocket engines back in the late 50's and early 60's while working for General Electric.[9] Figure 5 shows their 16,000 lb (71.2 kN) thrust, $H_2O_2$ uncooled plug nozzle thrust chamber mounted in a test cell. Incidentally, this engine developed a thrust level roughly equivalent to the LTD engine's maximum thrust in the airbreathing mode. As indicated in Fig. 6, six $H_2O_2$ decomposers were arranged around the base of the plug to provide exhaust gases for hot tests. Two, of the many external expansion nozzles which were tested, are displayed in Figs. 6A and 6B; an isentropic plug, and a 20° foreshortened plug. This rocket engine demonstrated good performance at off-design pressure ratios, and proved the feasibility of thrust vector control by selective combustor throttling. Subsequent analysis of the test data revealed that full isentropic plug nozzles could be subjected to substantial truncation/foreshortening, without degrading the thrust performance to any large degree. Figure 7 indicates the method used to foreshorten the plug nozzles, by replacing the tip with a cone at the appropriate tangent location.

For the first flight-weight demonstration of a regeneratively cooled plug nozzle rocket engine, Graham and Bergman decided to completely truncate the isentropic spike nozzle, replacing it altogether with the 42° half-angle cone shown in Fig. 8. Calculations were carried out to determine the minimum size of plug nozzle that could cool

| COMPONENT | MASS (Kg) | |
|---|---|---|
| $LN_2$ PROPELLANT TANK | 33.4 | |
| REGEN.-COOLED PRIMARY OPTIC | 33.0 | |
| SHROUD & INJECTOR PLATES | 15.0 | |
| ELECTRONICS & $C^3I$ | 18.0 | |
| TPS TILE (AC-C BASE) | 5.0 | |
| ATTITUDE CONTROL SYSTEM | 2.7 | |
| HIGH PRESSURE $N_2$ TANK (INCL. REGULATOR) | 3.2 | |
| INTERCONNECTING STRUCTURE | 9.6 | |
| DRY MASS | 120.0 Kg | PAYLOAD |
| ULLAGE ($LN_2$) | 30.0 Kg | |
| $LN_2$ PROPELLANT | 150.0 Kg | |
| TOTAL LAUNCH MASS | 300.0 Kg | |

Table 2: LTD Mass Breakdown

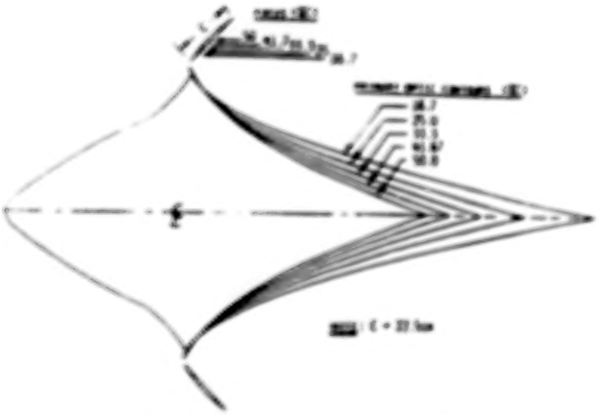

Figure 4: Primary Optic Contours vs. Focal Locations

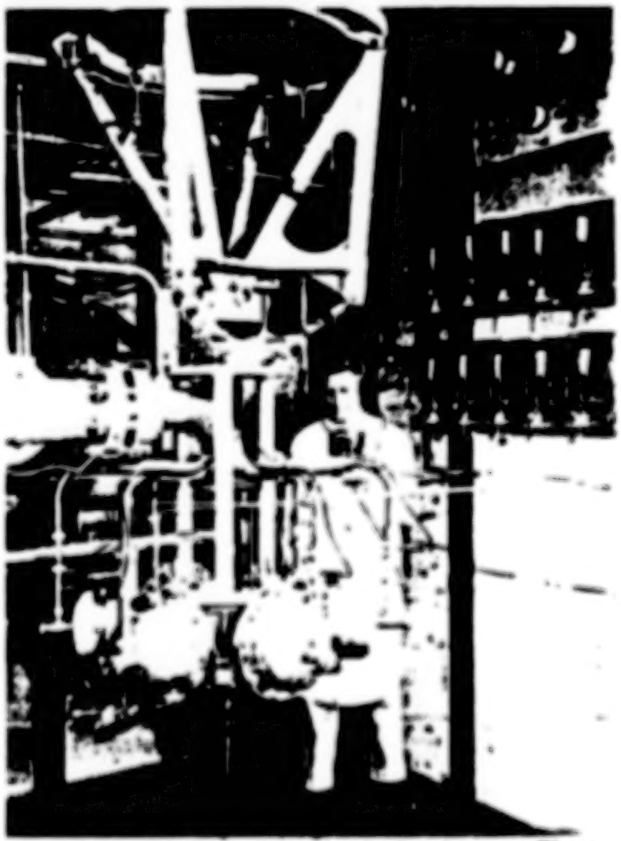

Figure 5: 16 Klb $H_2O_2$ Plug Nozzle Development Thrust Chamber in Test Cell (After Graham and Bergman, Ref. [9])

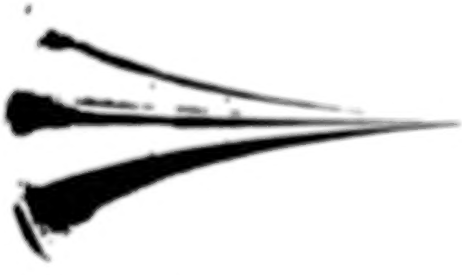

Figure 7: Isentropic Plug Nozzle, External Expansion (from Ref. [9])

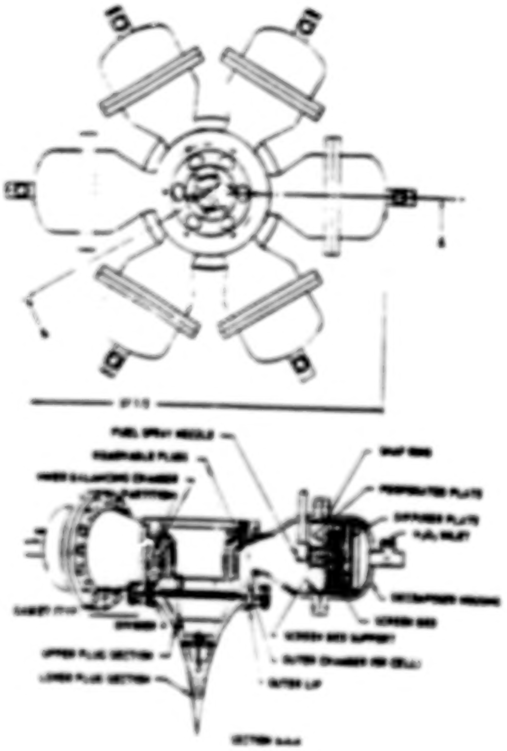

Figure 6: Detail of 16 Klb $H_2O_2$ Plug Nozzle Configuration (from Ref. [9])

itself with RP-1/LOX propellants. The result was the 50,000 lb thrust stainless steel engine displayed in Fig. 9. As indicated in Fig. 10, the plug nozzle had an exit nozzle diameter of 1.07 m (which is identical to that of the primary optic of the LTD machine). The overall maximum diameter of the 50,000 lb thrust engine was 1.34 m.

Figure 11 shows one of the eight cooled segments of which the engine is composed; each segment could be independently throttled to accomplish thrust vectoring. Pictured in Fig. 12 is an enlarged view of the lower cone segment, revealing the RP-1 coolant flow passageways.

The lower cone segments were fabricated from two thin stainless steel faceplates separated by numerous vertical ribs that formed the coolant passageways, all fused together by an electron-beam welding machine. Each of the eight lower cone segments had a mass of 2.077 kg and an area of 1089 $cm^2$; this translates to a mass penalty of 1.9 grams for every 1 $cm^2$ of regeneratively cooled surface area.

It is interesting to imagine how this method of construction might be applied to the design of a cooled primary optic for the LTD spacecraft. Similar techniques are no doubt being investigated for large cooled optics in SDI power-beaming applications. The shortest primary optic contour in Fig. 4 (i.e., focus at 30%$C$) has a total surface area of 17370 $cm^2$. At 1.9 $g/cm^2$, a stainless steel mirror might come in at 33 kg. Using aluminum, perhaps the 33.3%$C$ optic could be built for the same total mass. A schematic diagram showing the internal details of this primary optic design is given in Fig. 13.

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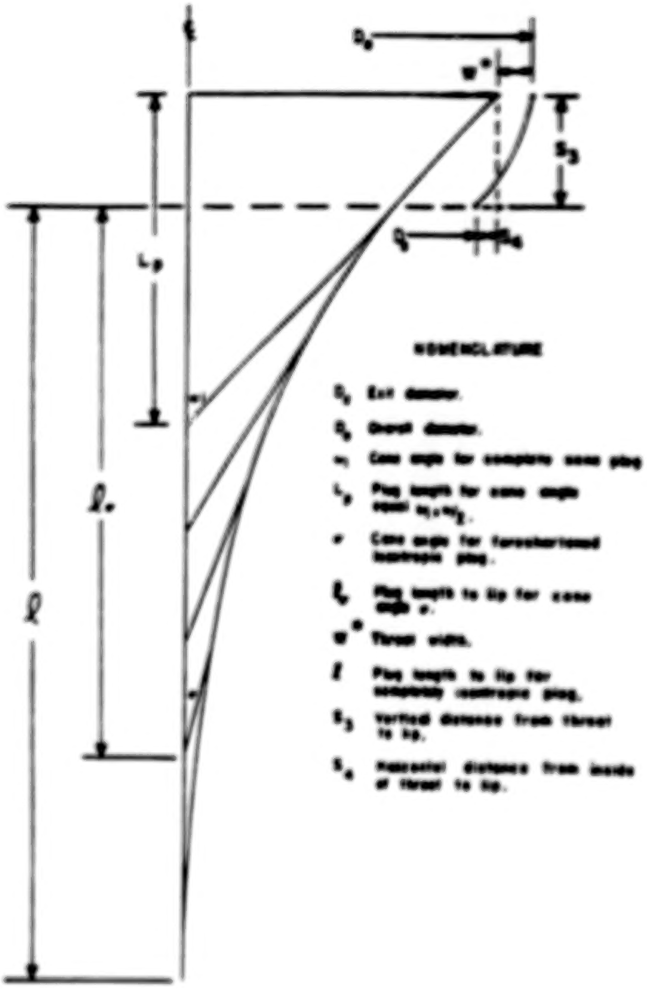
℄
$D_o$
$w^*$
$S_3$
$L_p$
$D_e$
$S_4$
$\ell_\alpha$
$\ell$
NOMENCLATURE
$D_e$ Exit diameter.
$D_o$ Overall diameter.
$\alpha_1$ Cone angle for complete cone plug
$L_p$ Plug length for cone angle equal $\alpha_1+\alpha_2$.
$\alpha$ Cone angle for foreshortened isentropic plug.
$\ell_\alpha$ Plug length to lip for cone angle $\alpha$.
$w^*$ Throat width.
$\ell$ Plug length to lip for completely isentropic plug.
$S_3$ Vertical distance from throat to lip.
$S_4$ Horizontal distance from inside of throat to lip.

Figure 8: Outline of 50 Percent Internal Expansion Plug Nozzle (from Ref. [9])

Figure 9: Cooled Engine Prior to Wire Wrapping and Thrust Mount Installation (from Ref. [9])

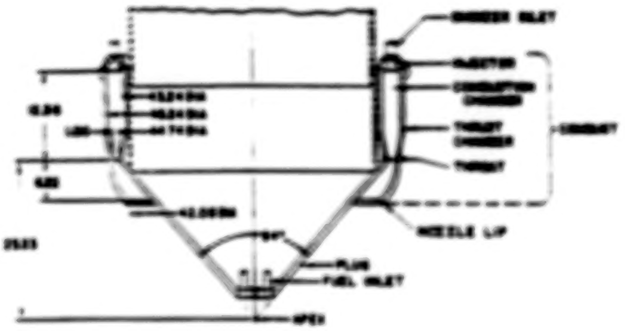
OXIDIZER INLET
INJECTOR
THROAT
COOLANT
NOZZLE LIP
PLUG
FUEL INLET
APEX

Figure 10: 50 Klb Plug Nozzle Configuration (Dimensions in inches) (from Ref. [9])

Figure 11: Cooled Segment, (from Ref. [9])

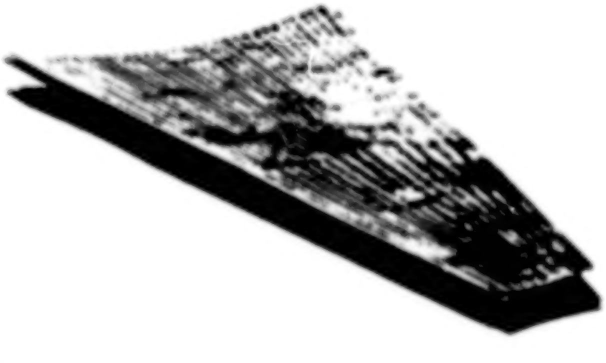

Figure 12: Cone Segment Assembly (from Ref. [9])

### Injector Head

Pictured in Fig. 14 is one of the eight ring injectors that forms a complete annulus at the top of the 50 Klb thrust rocket engine. One might envision a similar injector head, reduced in width to 1 cm, for the LTD spacecraft at the engine's annular focus region. As will be discussed below, laser supported detonation (LSD) waves must be ignited at this location with absolute reliability. Therefore, these injector heads could perhaps be covered with a tuned ignition array[10,11] or other material to minimize the beam energy and time required to ignite LSD waves. Other "hot section" surface areas which must be actively cooled within the LTD engine (see Fig. 3) include the shroud support struts, leading edge of the shroud airfoil, and entire lower flat surface of the airfoil — which is the primary impulse coupling region.

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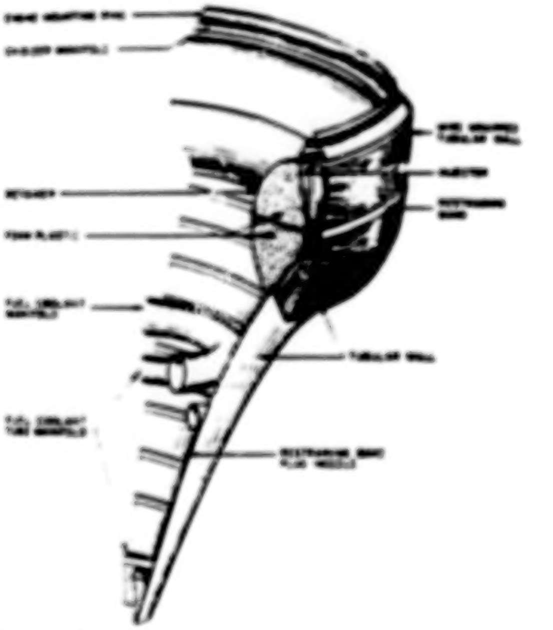

Figure 13: Quarter Sector of $33.33\%C$ Primary Optic (from Ref. [9])

## Mechanical Layout of the LTD Spacecraft

Now that the most massive LTD spacecraft items (such as the 70 cm diameter LN$_2$ propellant tank, regeneratively cooled primary optic, and rocket propellant injector head) have been described, the general spacecraft mechanical layout can be presented. With regard to the LTD cutaway view shown in Fig. 3, note that the entire external compression inlet forebody of the LTD is covered with black Thermal Protection System (TPS) tile, similar to that used on the shuttle orbiter lower heat-shield surface. Due to the high stagnation conditions at the LTD's nose, this region must also be capped with the same carbon/carbon material used on the Shuttle nose and wing leading edges. On the LTD, this lightweight thermal protection system will be fastened to a carbon/epoxy composite substructure that attaches directly to the LN$_2$ tank. The spherical, filament-wound Kevlar LN$_2$ tank serves as a structural backbone for the entire LTD vehicle, to which all subcomponents are rigidly attached.

Note in Fig. 3, that the electronics/instrumentation compartment resides in the nose section just forward of the LN$_2$ tank. Also located in this region are the cold N$_2$ gas jets required for spacecraft attitude control in the satellite mode. For spin-stabilization, additional jets would be placed out near the shroud, but still in the centerbody volume.

As well stated by Fuhs and Mosier,[3] "Sounding rockets, rather than scaled down satellites, are the best model for lightsat design, fabrication and operation." The LTD spacecraft can certainly be categorized as a lightsat, where cost is of prime concern. For this reason, the cost and increased complexity of the H$_2$O$_2$ attitude control systems are abandoned in favor of simpler cold gas jets. A small solar array could be integrated with the upper surface of the shroud and would recharge a few batteries when bathed in sunlight. Simple discrete devices would be used for power conditioning and regulation.

Figure 14: Injector Model III (from Ref. [9])

Finally, the advanced microelectronics used for signal processing, data transmission, guidance and control must be powerful enough to carry out multiple missions: (1) laser boost phase, and (2) satellite phase.

As indicated in Fig. 3, the annular shroud is firmly affixed to the LN$_2$ tank by way of a special shock-damping mechanism that attempts to isolate the primary optics and the rest of the spacecraft from the vibrating shroud element. Most of the thrust developed by the repetitively-pulsed combined-cycle engine is communicated directly to the shroud. Perhaps computer controlled active damping structures technology can be applied here.

A few comments should be made about the shroud's mechanical construction. As noted in Fig. 3, the shroud may have a trailing edge flap to act as a variable area nozzle, and also, perhaps to reduce the vehicle's frontal area (and hence, drag) when in an unpowered "coasting" mode within the atmosphere. The shroud structure will probably be the most abused of any in the spacecraft, and must be exceedingly strong. Perhaps this is the place for high temperature metal matrix composites and refractory materials.

When transitioning between airbreathing and rocket modes, servo-mechanical actuators will deploy shutters to close off the annular air gap (i.e., between the LTD centerbody and shroud) to shut off the air flow. At this time the propulsion mode becomes a pure rocket, and the LN$_2$ mass flow rate is increased dramatically by the pressure regulator. In this regard, it is appropriate to describe the propellant management system. Because of the small size of the LN$_2$ tank, the most expedient and reliable propellant delivery system would be to simply pressurize the tank. This approach is not accessible to less efficient propulsive systems which require a very large volume of propellants because tank weight quickly becomes prohibitive. Hence, a separate, small (25–30 cm diameter), high pressure (e.g., 3 to 6 Kpsi) tank (perhaps carbon fiber wound) and regulator is invoked for the LTD propellant delivery system.

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### Optical Systems Layout

For the LTD spacecraft, the primary optic can be designed to operate in four basic roles: (1) primary concentrating optic for laser propulsion, (2) sensor, (3) transmitter aperture, (4) receiver aperture. To make the transition between the first and second roles, an advanced sensor array (i.e., retina) would be deployed, by servo-mechanical actuators, into the 1 cm wide focal region of the PO mirror. Then, data from this distributed "insect eye" retina would be combined electronically or optically (like the six eye multiple mirror telescope) into a single high quality image.

To transition into the receive/transmit role, perhaps low power monolithic microwave integrated circuits (MMIC) can be used to drive phased-array elements (positioned in the focal region of the parabolic PO). As suggested by Browde in Ref. [3], this should be a good way to cut the weight of the radio frequency radiator and receive/transmit electronics, while providing an electronic means of beam steering.

### Description of Airbreathing Propulsion Mode

The LTD machine will utilize a combined-cycle engine design. The design incorporates both airbreathing and rocket modes of operation. The airbreathing mode is a form of External Radiation-Heated (ERH) thruster.[6,7] In this engine concept, a laser induced plasma is ignited on the lower surface of the shroud. This high pressure (e.g., 600 atm) plasma bubble expands as it is convected down the surface of the shroud. The expansion yields a higher pressure on the aft side of the shroud, producing a forward thrust. This "shroud lift" ERH thruster is pictured in Fig. 15. The performance of this engine is demonstrated in Figures 16, 17, and 18. The reader is encouraged to examine Ref. [7] for a detailed description of this propulsion concept.

### Analysis of LTD Vehicle Inlet Flow

In order to analyze the performance of the airbreathing engine, it was necessary to model the LTD vehicle's inlet flowfield. The basic assumptions that had to be made were mainly for the simplification of the analysis. While it was important that an accurate model of the flow over the vehicle forebody be developed, a simplified model had to be developed first. In the construction of the model, it was held in perspective that the optimization of the flow of fresh air to the thrust area was to be the primary objective. This flow of fresh air was necessary to solve the "refresh" problem that occurs with such a pulsed engine.

### Assumptions Made in the Analysis

It was of great importance that the air velocity at the location of thrust generation be as high as possible. This could best be achieved by using a vehicle forebody that was as slender as possible for supersonic flow and yet still satisfactory for subsonic flow. It was assumed that the flow was largely inviscid, although some viscous effects would eventually be included during the evolution of the model. The flow is dominated by compressible flow effects since the flight regime of interest could go up to Mach 8. The forebody was modeled as a double cone, made up of 30°, ane a 45°, semi-vertex angle cones.

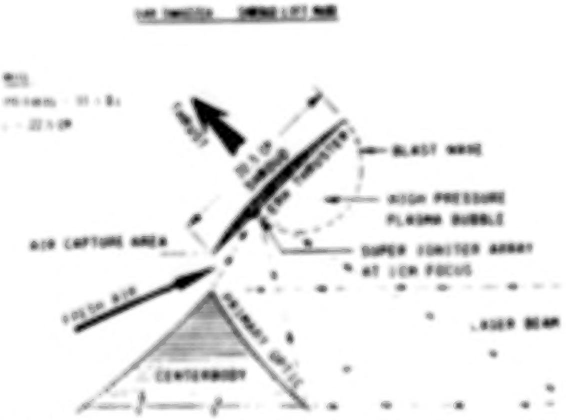

Figure 15: ERH Thruster: Shroud Lift Mode

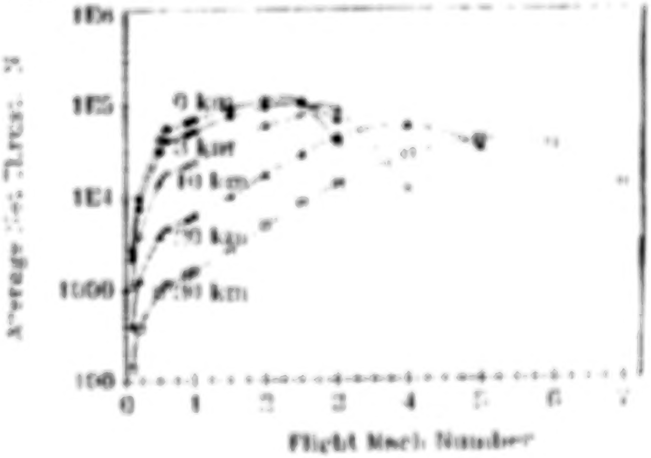

Figure 16: "Shroud Lift" ERH Thruster: Average Net Thrust

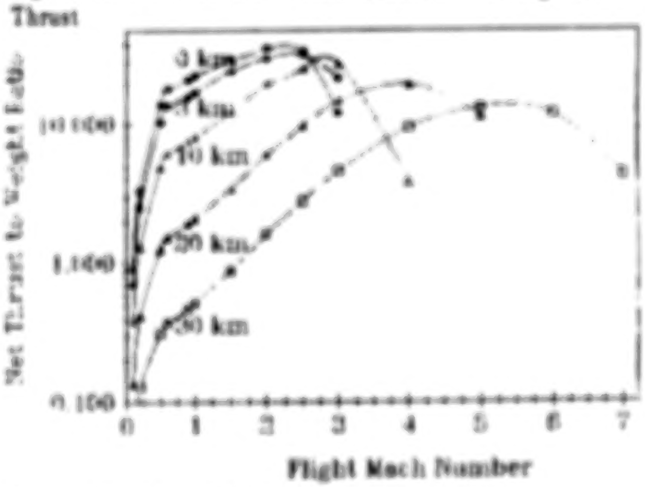

Figure 17: "Shroud Lift" ERH Thruster: Net Thrust to Weight Ratio

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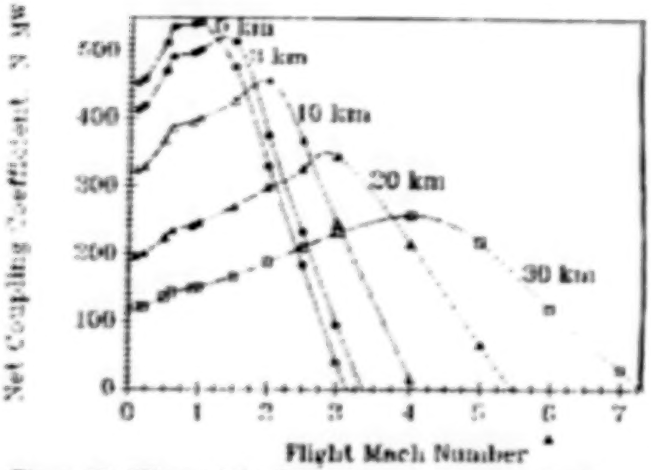

Figure 18: "Shroud Lift" ERH Thruster: Coupling Coefficient

To determine the velocity at the flow location, it was necessary to know the velocity of the air at the entrance of the shroud region. The latter required knowledge of the velocity over each of the cones making up the vehicle forebody.

For the subsonic part of the flight regime, it was assumed that the ratio of the maximum air velocity over the cone to the freestream velocity was similar to that over cones and/or spheres. For this vehicle, a ratio of inlet to freestream Mach number of 5 to 3 was assumed until the inlet velocity reached sonic value, and then it was assumed to remain approximately equal to one ($M_{in} = 1$) for the subsonic part of the flight.

## Analytical Procedure

With the inlet Mach number schedule assumed above, one less parameter was needed for the determination of the inlet velocity and mass flow rate. These values are needed for the calculation of the thrust-area refresh air velocity and the vehicle ram drag.

For supersonic flow conditions, the shocks formed over the vehicle forebody rendered the analysis a bit tedious. The Mach number after the oblique shock over the first cone forming the vehicle forebody was calculated with the oblique shock relations for a wedge. The same type of relations were used to calculate the stagnation pressure ratio $p_{t_1}/p_{t_a}$, and static pressure ratio, $p_1/p_a$ across the shock as well as the shock angle.

Once the Mach number after the first oblique shock ($M_1$) was known, the same process was employed in calculating the Mach number after the shock over the second cone ($M_2$), the pressure ratios $p_{t_2}/p_{t_1}$ and $p_2/p_1$, and the shock angle, if $M_1$ was supersonic.

The overall pressure recovery of the vehicle inlet is the stagnation pressure ratio from the inlet to freestream. This is obtained by multiplying the stagnation pressure ratios after the first and second shocks. Note that the inlet itself would have a normal shock which may reduce the refresh air Mach number to below the inlet Mach number schedule assumed. This and other assumptions are to be verified by more detailed models later.

With the above information, the inlet velocity can now be calculated as:

$$u_{in} = \sqrt{\frac{\gamma p_{in}}{\rho_{in}}} \tag{4}$$

where $p_{in}$, the inlet pressure, and $\rho_{in}$, the air density at the inlet, can be calculated from the ambient pressure, $p_a$, and the freestream Mach number, $M_a$, with

$$\frac{p_{in}}{p_a} = \left(\frac{1 - \frac{(\gamma-1)}{2}M_a^2}{1 - \frac{(\gamma-1)}{2}M_{in}^2}\right)^{\frac{\gamma}{(\gamma-1)}} \frac{p_{t_2}}{p_{t_1}}\frac{p_{t_1}}{p_{t_a}} \tag{5}$$

$$\frac{\rho_{in}}{\rho_a} = \left(\frac{p_{in}}{p_a}\right)^{\frac{1}{\gamma}} \tag{6}$$

The ram drag can be calculated as:

$$D_{RAM} = \dot{m}u_{in} \tag{7}$$

where

$$\dot{m} = \rho_{in}u_{in}A_{inlet} \tag{8}$$

The pressure recovery schedule, determined from the above calculations, is presented in Fig. 19.

## Trajectory Simulation

The trajectory simulation of the LTD was done on McDonald Douglas' Simulation and Optimization of Rocket Trajectories (SORT)[12] computer program. Three different launch angle cases of 30°, 45° and 60° (measured from horizontal) were run on the program.

In each case the LTD machine was given an initial velocity of 100 m/sec. At 0.5 sec the shroud lift ERH thrusters were engaged and the vehicle was flown until it reached a Mach number of 7.0, the upper limit of ERH thruster performance. When this velocity is attained, the ERH thruster model switches to the scramjet model and flies to an altitude of 160 kft. At this point the engines shut off and the LTD vehicle cruises until it reaches the highest altitude before diving back to earth. The craft is then pitched over to 0° with respect to the horizon and the laser heated rockets are engaged, accelerating the LTD to orbital velocity of 8 km/sec.

Figures 20 to 25 show the results of running the SORT program. The 30° initial angle case is the most undesirable of the three cases flown. In this case the vehicle starts to return to earth instead of entering orbit. The LTD also uses more power then is allowed (250 MW) for this vehicle. The 60° case obtains orbital velocity sooner and a greater altitude but uses too much fuel in the rocket mode in reaching orbital velocity.

In all three cases the scramjet only operates for a few seconds and does not add much acceleration to the LTD machine. This is due to the fact that the craft climbs through the atmosphere very quickly and attains altitudes where the scramjet produces very little thrust.

To compare a different mission profile, an LTD trajectory was run at 30° initial launch angle with the craft switching from shroud lift ERH thruster directly to laser-heated rocket. This mission bypasses the scramjet mode,

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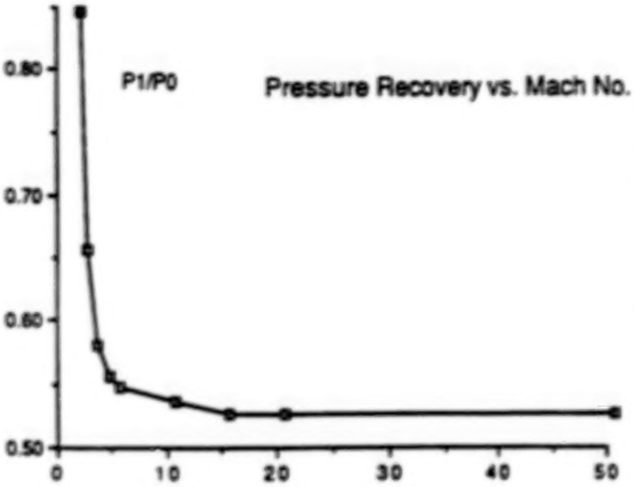

Figure 19: Pressure Recovery ($p_1/p_0$) vs. Flight Mach Number for the LTD Inlet

which the first three cases use. The specific impulse ($I_{sp}$) for the rocket was raised from 650 seconds to 875 seconds. Figures 26 to 31 show the performance of the LTD machine for this run. The craft climbs more quickly and travels further downrange than the first three cases while having a longer rocket phase. The greater specific impulse allows the craft to have a larger final mass as it goes into orbit.

The first three cases are useful for Low Earth Orbit missions while the ERH/Rocket mode can be used for higher altitude operations. In both these cases an onboard chemical rocket or a laser relay satellite is needed for the final thrust into orbital insertion.

## Conclusions

The proposed Lightcraft Technology Demonstrator should create a significant economic impact. Since the LTD machine integrates a laser propulsion engine with the satellite hardware, it results in a decrease in both launch costs and payload costs. The single-stage-to-orbit spacecraft becomes a sensor satellite after arrival in orbit. The LTD's dry mass of 120 kg plus 30 kg of $LN_2$ ullage represents the entire payload. This payload, containing the 1 meter diameter optic, is suitable for a sensor satellite mission at altitudes up to 2000 km. Since the laser propulsion engine and satellite functional hardware are shared, the LTD concept can lower both *launch* and satellite *payload* costs by an order of magnitude, or more.

The LTD machine utilizes a combined-cycle engine to reach orbit. This approach was chosen over an all-rocket system because of the need to reduce liftoff mass. The airbreathing boost to Mach 7 and 30 km altitude enables a vehicle mass ratio of 2.0 to LEO. An all-rocket approach would require a liftoff mass of 1000 kg to put the same 150 kg into orbit. The combined-cycle engine can do the same mission with a liftoff mass of 300 kg. Since airbreathing engines have impulse coupling coefficients (CC) of 2× to 4× greater than rockets, laser-boosted spacecraft with combined-cycle engines will outperform those with only rocket engines.

Liquid nitrogen was chosen as the propellant/coolant for the LTD machine. Although liquid hydrogen would give the highest specific impulse, its low density would allow only 14 kg to fit in the proposed 70 cm diameter tank. Water has been examined as a possible fuel for laser-energized engines, but was found to be inappropriate in this case. Water would result in an exhaust plume that is highly absorbent to the suggested 10.6 $\mu$m radiation. Liquid $N_2$ has a density only a little less than that of $H_2O$ (80.8%), allowing 180 kg to fit in the propellant tank. Also, $N_2$ has properties close to those of air, which is 78% $N_2$ by volume. This will simplify the transition between airbreathing and rocket modes. Additionally, $LN_2$ is inert, and will not contaminate the high power laser optics, and $LN_2$ cooled aluminum mirrors have been developed for SDI. A nitrogen fueled laser-heated rocket would have a specific impulse ($I_{sp}$) of 725 to 1025 sec.

Finally, it should be emphasized that the technology for the LTD machine exists today. Several critical "proof-of-concept" experiments for pulsed laser propulsion have already been demonstrated. The required ground based laser (GBL) system could be constructed with materials and technology that has been in existence for 15 years. The optics, fuel management system, and plug nozzle are all possible with present technology. Although certain aspects of the design could be improved with advanced technology, the proposed concept is a near-term, realistic demonstration of the feasibility and usefulness of laser-energized propulsion.

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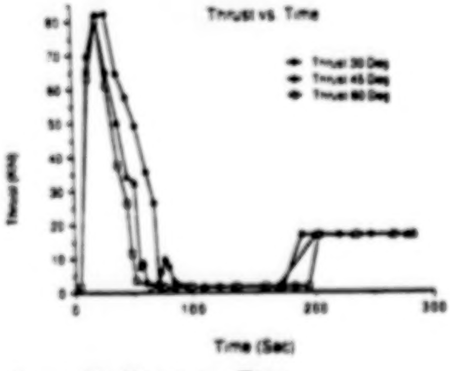

Figure 20: Thrust vs. Time

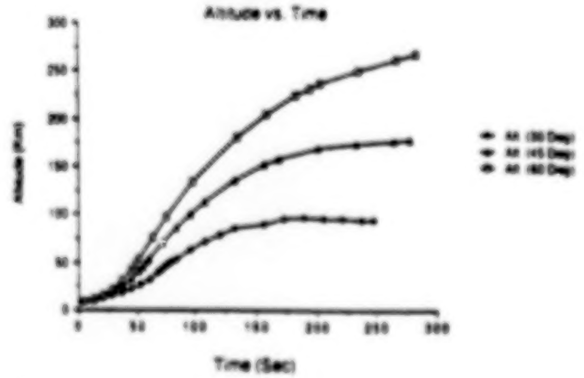

Figure 21: Altitude vs. Time

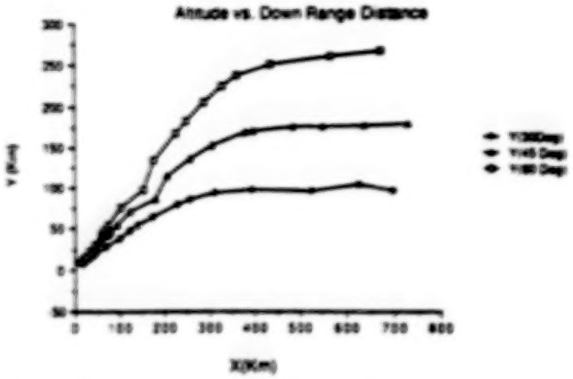

Figure 22: Trajectory (Altitude vs. Downrange Distance)

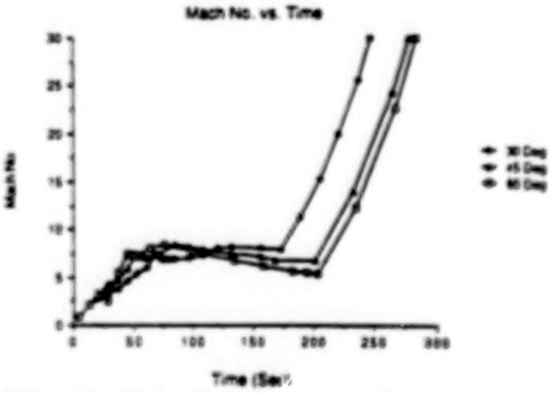

Figure 23: Flight Mach Number vs. Time

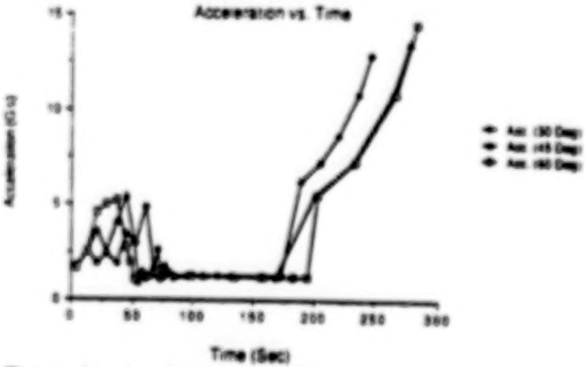

Figure 24: Acceleration vs. Time

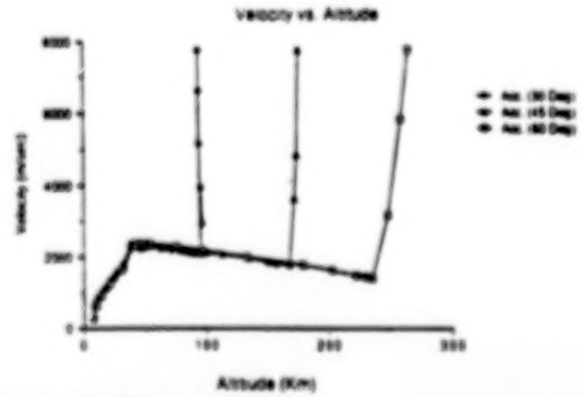

Figure 25: Velocity vs. Altitude

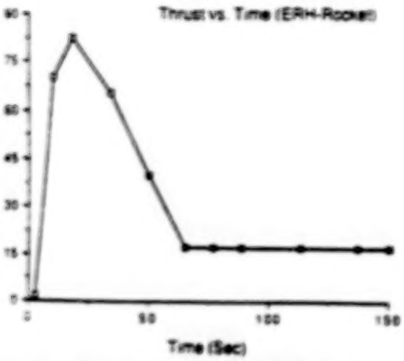

Figure 26: Thrust vs. Time (ERH-Rocket Only)

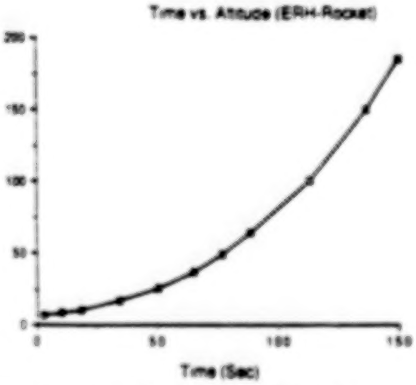

Figure 27: Altitude vs. Time (ERH-Rocket Only)

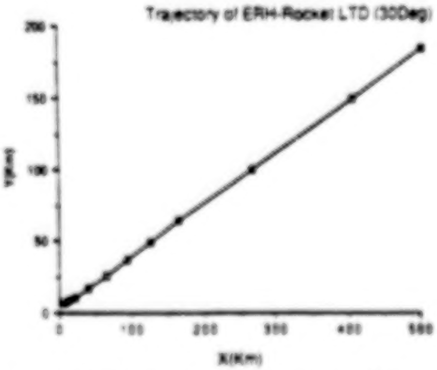

Figure 28: Trajectory (Altitude vs. Downrange Distance) (ERH-Rocket Only)

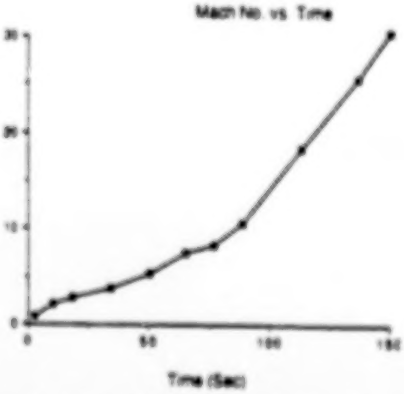

Figure 29: Flight Mach Number vs. Time (ERH-Rocket Only)

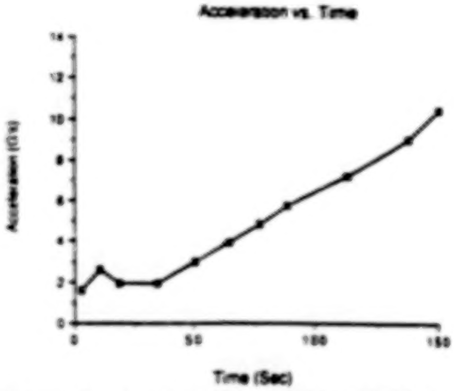

Figure 30: Acceleration vs. Time (ERH-Rocket Only)

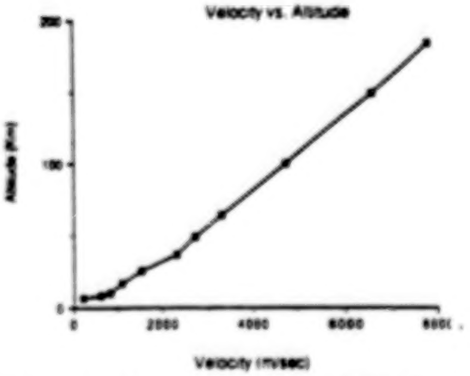

Figure 31: Velocity vs. Altitude (ERH-Rocket Only)

N91-22157

# GROUND-TO-ORBIT LASER PROPULSION — ADVANCED APPLICATIONS

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Laser propulsion uses a large fixed laser to supply energy to heat an inert propellant in a rocket thruster. Such a system has two potential advantages: extreme simplicity of the thruster, and potentially high performance -- particularly high exhaust velocity. By taking advantage of the simplicity of the thruster, it should be possible to launch small (10 - 1000 kg) payloads to orbit using roughly 1 MW of average laser power per kg of payload. The incremental cost of such launches would be of order $200/kg for the smallest systems, decreasing to essentially the cost of electricity to run the laser (a few times $10/kg) for large systems. Although the individual payload size would be small, a laser launch system would be inherently high-volume, with the capacity to launch tens of thousands of payloads per year. Also, with high exhaust velocity, a laser launch system could launch payloads to high velocities -- geosynchronous transfer, Earth escape, or beyond -- at a relatively small premium over launches to LEO.

In this paper, we briefly review the status of pulsed laser propulsion, including proposals for advanced vehicles. We then discuss qualitatively several unique applications appropriate to the early part of the next century, and perhaps valuable well into the next millenium: space habitat supply, deep space mission supply, nuclear waste disposal, and manned vehicle launching.

Space habitat supply depends primarily on the ability of the laser propulsion system to launch large total volumes at low cost, and with sufficient precision to avoid expensive rendezvous maneuvering. However, a key advantage is the laser system's ability to launch on short notice -- the ability to receive spare parts, emergency supplies, etc. on less than 24 hours notice could greatly simplify the logistics of space facilities. A crucial factor is the laser's cross-range capability, which allows a launch window of several hours per day to an inclined orbit.

Deep space mission supply requires the same properties as habitat supply, but also requires high specific impulse to reach Earth escape. Rendezvous with a deep-space mission could be aided by an on-board laser.

Nuclear waste disposal takes specific advantage of what is normally a disadvantage of laser propulsion -- small payload size. A laser launch system can demonstrate an almost arbitrarily low risk by launching a large number (100,000) of test payloads and allowing them to "crash" in various ways to verify emergency recovery systems. However, given that even a well-tested and reliable system can fail, the small payloads used would minimize the potential environmental damage from a failure. Very modest system performance would suffice for disposing of material on the Moon; a high-performance system could dispose of waste into deep space or into the sun.

Finally, launching manned vehicles requires relatively large payload capacity and places a premium on low acceleration. A gigawatt-scale laser propulsion system could provide the needed capacity, however, and could easily be designed and tested to provide the extrememly high level of safety needed for routine manned flight.

## Introduction

Laser propulsion uses a large stationary laser to send energy to a small rocket vehicle. Pulsed laser propulsion uses high-energy laser pulses to ablate a solid (or liquid) propellant. With a suitable laser pulse cycle [1], specific impulses up to 1000 seconds can be attained with inert, storable propellants. Pulsed propulsion also makes possible very simple thrusters (potentially just a block of solid propellant) which may not require cooled (or indeed any) nozzles. Such thrusters provide two additional advantages: they can produce thrust at an angle to the incident laser beam, and they can be remotely steered by controlling the beam profile.

The SDIO Laser Propulsion Program, started in 1987, has focussed its efforts on using nozzleless solid-propellant thrusters to launch very small payloads into low Earth orbit (LEO) [2]. A laser launcher takes advantage of the thruster's ability to accelerate at an angle to the laser to launch vehicles directly into LEO without a "kick motor". Ground-based guidance eliminates the need for on-board guidance and control hardware, allowing very cheap disposable vehicles -- potentially less complex than a modern refrigerator. The vehicles would necessarily be mass-produced, and should thus be very inexpensive.

The components of a first-generation laser launch system are shown in figure 1. The estimated cost of building such a system is roughly $500 million; it would be capable of launching some 30,000 20 kg payloads into LEO each year, for a total launch capacity of 600 metric tons (MT) per year. A design and some applications for such a system are given in Kare [3].

This is, however, only a first-generation system, such as might be built in the next 5 to 10 years. Larger, more reliable, and higher performance systems are certainly possible. The next section discusses some possible directions for improvement, and the following sections discuss some possible applications for such second- and later-generation sysems. The key properties of laser propulsion to keep in mind are:

Simplicity (of the laser-driven thruster and vehicle)
    Low cost, highly reliable, economically scalable to very small size

High Performance
    High $I_{sp}$ allows single-stage-to-anywhere
    Precision ground-based guidance

Safety
    Inert propellant means trajectory is always known; cannot go off course
    No explosion hazard -- during loading, at launch, or in flight
    Small vehicle -- worst crash is less destructive than a light plane crash

Low acceleration -- comparable to chemical rockets, not "cannons"

BUT --
Limited payload size compared to chemical rockets
    No fundamental limit, but capital costs of large systems are high

Less flexible than some self-contained systems
    Diffraction- and horizon-limited range
    Fixed launch site (vs., for example, Pegasus or SSX flexibility)
    Subject to weather delays

## Status of Pulsed Laser Propulsion Research

The double-pulse thrust cycle is illustrated in figure 2. A low-energy laser pulse evaporates a thin layer of solid propellant from a large block. This layer expands to of order atmospheric density, forming a gas layer millimeters to centimeters thick. A second, higher energy pulse forms a laser-supported detonation wave (LSD) wave at the solid surface -- a strong shock which heats the gas enough to create ionization that absorbs the laser beam. The laser beam energy in turn heats the gas behind the shock, maintaining the shock strength and keeping the wave going. When the shock has heated the entire gas layer, the laser turns off, leaving (ideally) a uniform gas layer at of order 10,000 K, which expands to produce thrust. Since the hot gas layer is very thin compared to the vehicle diameter, the expansion produces thrust efficiently without a nozzle.

Although the double pulse allows efficient heating of the gas to very high temperatures, the flat-plate nozzleless nature of the system remains even if only a single laser pulse is used. At low flux, a single pulse simply ablates the surface, creating a relatively cool, low velocity exhaust; this is an ablation-mode thruster.

Laser Propulsion Program research has consisted of computational modelling of the various phases of the thrust cycle, and of small-scale experiments using 1-100 Joule $CO_2$ lasers to generate single impulses on various propellant materials suspended in vacuum. These experiments generally measure the total impulse given to the target, and the mass lost by the target. These can be converted to a specific impulse (impulse/mass) and an efficiency (kinetic energy in the exhaust/laser pulse energy). The Program goal has been an efficiency of 40% at a specific impulse of 800 seconds (exhaust velocity of 8 km/s), but lower $I_{sp}$'s of 300 to 400 seconds (comparable to a liquid fuel rocket) are sufficient for launching payloads to LEO.

The four phases of the double pulse cycle are:

Evaporation
Plasma ignition
Propagation of Laser-supported detonation (LSD) wave
Expansion and recombination

These same phenomena occur with single laser pulses, but may overlap or change in importance -- in particular, an ablation-mode thruster may provide sufficient $I_{sp}$ for LEO launches with little or no plasma formation, but would correspondingly make the evaporation and expansion phases more critical.

Some major double-pulse modelling results:

- Long pulses (>100 ns, preferably >1 μs) are desirable
- Propellant must be a strong absorber in solid state
    - Long absorption depth puts too much heat into remaining propellant
- Low-ionization-potential "seed" strongly helps LSD-wave formation
- Full recombination is unlikely in high-Isp thrusters

Major experimental results:

- Enhanced efficiency and $I_{sp}$ with double pulses demonstrated
    - Strong dependence of impulse, mass loss on interpulse time
- 10x reduction of plasma ignition threshold with "invented" propellants
- Demonstrated 25 dyne-s/J (250 N/MW) coupling in air with "dimpled plates"
- Efficiencies (Exhaust kinetic energy/Laser pulse energy) demonstrated:
    - 8-10% at 600 - 800 s $I_{sp}$
    - 15% at 600 s $I_{sp}$ with long pulses
    - 20-30% at 200 s $I_{sp}$

Near future plans:

- 1 kJ, 1 μs pulse experiments
    - Goal is 20% efficiency at 600 s Isp and 40% at 300 s
- Ablation-mode tests
- Modelling and experiments at 1.06 μm for compatibility with SDIO FELs
- Rep-pulse experiments at substantial average power in 1991-92

## Directions For Growth -- Laser Propulsion in the 2000's

Laser propulsion has the nice property of growing essentially linearly from an initial system launching 20 kg payloads to gigawatt-scale systems launching multiton payloads. However, there are many ways to improve the basic system other than simply building a bigger one:

- Advanced vehicles
  - Primarily work of Myrabo -- Apollo Lightcraft [4] and Technology Demonstrator [5]
    - High mach number air-breathing performance
    - Efficient integrated structures
  - Emphasize performance rather than lowest vehicle cost
  - Great potential for 2nd and later generations
    - Vehicles must be re-usable; probably must be large(r) to be economic
    - Designs require lasers and/or relay mirrors in orbit
- Advances in lasers/optics
  - Free Electron Lasers
    - Short wavelength, tunable for maximum transmission
    - Potentially 25% efficient or better
  - Diode and diode-pumped lasers
    - Potentially as cheap as power semiconductors -- pennies per watt
    - Short wavelength, highly reliable ("no moving parts")
    - Potentially very efficient -- 50%? -- reduces power cost
  - Large, low cost beam directors via segmented active optics
    - >>10 meter diameters are possible
- Space-based relay mirrors increase flexibility, performance
  - Extend range over the laser's horizon
    - Much greater "reach" for orbital maneuvering
  - Increase launch windows to inclined orbits
  - Large mirrors (potentially easy in space) can give very long range
    - Range = $D_1 D_2/\lambda$
    - 100 meter mirror directly drives vehicles in GEO
    - 100 m mirror and 100-1000 m collector reaches Mars
- Space-based lasers eventually do the same
  - May be necessary at short wavelengths to avoid atmospheric limits
  - Can be direct solar or solar-electric powered

**Application 1: Habitat Supply**

Beginning with Space Station Freedom (or even with the Soviet Mir), more or less permanent habitats will exist in cislunar space. These will need many kinds of supplies, primarily transported (at least at first) from the Earth.

1. Routine (re)supply
   - Consumables: Food, water, air, fuel/reaction mass (which could be water)
   - Raw materials for space industrial products -- silicon, metals
   - Miscellaneous small items: parts, lubricants, laboratory supplies
   - Construction materials
2. Priority supplies
   - Replacement parts/tools
   - Specialized tools and hardware
   - Perishable samples or reagents -- even radioisotopes
   - Medical supplies

Routine resupply can be minimized through recycling, but highly efficient recycling will be complex and costly. Many items, notably raw materials for export products and fuel, cannot be recycled. Some items could be supplied from the Moon or other space sources: oxygen, possibly water, reaction mass, and even some raw materials and construction materials. But many items will come only from Earth until an extensive space mining and manufacturing economy develops. Laser propulsion offers:

1. Low-cost routine supply -- incremental launch costs of $10 - $100 per pound
   - Moderate handling costs
     - Minimal ground "payload integration" costs & delays
     - Space payload handling must be automated via small self-contained ''retrievers''
       - Can't have an astronaut out collecting every 100 kg parcel
   - Respectable total capacity
     - Inclined orbits: ~10 launches per day
     - Equatorial orbits: ~100 launches per day
2. Efficient launch to GEO, L4/L5, etc.
   - Ideal for laser launch -- trajectories stay above horizon; high $I_{sp}$ is well-matched
   - Modest laser on habitat (10% of GBL size) useful for apogee burn
3. Launch on demand; at most 24 hour delay, usually less
   - But requires at least 2 launch sites to allow for weather, equipment failures, maintenance
   - Also require very reliable hardware at the habitat if vehicles need help to rendezvous
     - Keep one ''ready'' rocket for extreme-emergency situations

NO conventional system offers priority supply (unless traffic is so heavy there is ~1 launch per day in any case). The cost is exorbitant even for the most optimistically-priced vehicles, such as the SSX, with a per-launch cost of $1 million. Yet priority suppy can drastically simplify logistics: if spares and emergency supplies can come from the ground, you don't have to carry everything you might ever need in a hurry.

### Application 2: Deep-Space Mission Supply

This topic is discussed in some detail in an earlier paper [6]. Laser propulsion is of limited direct use in driving deep space missions, because diffraction spreads the laser beam to an unusably large diameter over interplanetary distances -- although eventually, as the scale size of the laser transmitter and the receiving vehicle grow, the useful range can be interplanetary or even interstellar [7]. The most immediate use of laser propulsion is simply as a low-cost way to place mission components (fuel, structural mass, etc.) in Earth orbit.

However, laser propulsion can have more direct applications. Microspacecraft have been proposed [8] to preceed deep space missions and perform such preliminary tasks as selecting a landing site and sampling local conditions. A high-$I_{sp}$ laser launch system is ideal for launching such precursor probes.

A laser launcher could send out supply packages to rendezvous with a deep space mission, either en route or at its destination. However, the rendezvous velocity would be high for most trajectories, and even a very small error (or deliberate change) in the trajectory of the main mission would cause supplies to miss their target. Putting thrusters and guidance hardware on the supply packages would make them expensive -- essentially spacecraft in their own right -- and thus probably uneconomical.

The situation is different if the mission vehicle is large enough to carry a laser of respectable size -- at least megawatt-scale. It can then "reach out and grab" incoming supply packages over a large volume of space and a substantial range of relative velocities. For this application, the inert, storable nature of the laser propulsion propellant is critical -- a small supply package could not store cryogenic propellants.

The reach of the mission vehicle can be extended even further if the supply packages carry lightweight concentrators to collect the incident laser light. Since the laser can deliver power to such a concentrator for a longer time than to a thruster directly, the required size of the on-board laser is also reduced.

Although prompt supply is not possible even with a laser propulsion system over interplanetary distances, the ability to do a high delta-V launch (and to some extent, a high delta-V capture maneuver) means that a laser system could launch supply packages on much faster trajectories than those likely for chemical propellant systems. This could allow, e.g., getting specialized research tools to a Mars mission before it leaves the planet, when the need is only discovered after the mission arrives.

A major limitation is that any such deep-space mission support requires very high confidence in the on-board laser -- or limits supply packages to non-mission-critical items.

### Application 3: Nuclear waste disposal

Kantrowitz [9] has suggested using a laser propulsion system to dispose of high-level radioactive waste in space. The problem of finding an environmentally acceptable waste disposal site has consumed billions of dollars and met with enormous political complications of the 'NIMBY' (Not In MY Back Yard) variety. Disposal of waste in space has been studied fairly extensively [10], but conventional launchers (in addition to being very expensive) always present the spectre of a catastrophic accident releasing the radioactive payload into the environment. No amount of engineering design can eliminate that risk, and no reasonable test program using conventional launchers can demonstrate safety. The problem is compounded by the need to launch, at the very least, to the Moon.

Laser Propulsion offers safe, cheap disposal:

- Arbitrarily high *demonstrated* reliability:
  - Laser system can be modular and heavily "overbuilt" -- even duplicated
    - Single-stage launch -- no failures in LEO
  - Very many (e.g., $10^5$) vehicles can be test-launched
  - Emergency re-entry/recovery systems can be tested $10^5$ times too
    - Catastrophic failure probability less than one-in-a-billion
- Inherent safety even in disaster
  - Small payload size means even a worst-case accident is limited
    - Easy to crash-proof (mouse vs. elephant)
  - Inert vehicle -- can't explode, can't go "off course"
  - Of course, you do need to *find* a payload that crash lands in Mongolia...

Unlike weight- and volume-limited conventional systems, a laser launcher could potentially handle unprocessed or minimally-processed waste. This minimizes both radiation and toxic chemical hazards on the ground, and is therefore crucial to an economical system. A laser system could even be cheaper than geological disposal, because there would be less handling (separation, glassification) of waste.

Lasers can launch waste directly to any desirable disposal site -- the Lunar surface, interplanetary space, or deep space (solar escape). The required delta-V's are roughly 11 to 15 km/s, beyond the capability of any single-stage chemical rocket or proposed cannon launcher. Laser propulsion could even launch payloads directly into the Sun, at 30 km/s delta-V. The precision guidance and flexible launch direction of a laser system could allow dumping payloads into, e.g., a selected lunar crater, for future recovery if desired.

Very small laser propulsion payloads could present problems of shielding (to protect both launch-site workers and possible crash site bystanders) and safe any-angle reentry [11]. However, some problems of laser propulsion, such as launch delays due to weather, are not important as long as the total mass launched is constant and the reliability is high.

**Application 4: Manned Launch**

In the long run, the most valuable payload is always Man. Laser propulsion, because of its inherent safety, is a nearly ideal launcher for people, provided the basic requirements of a man-rated launcher can be met.

- Requirements:
  - Excellent safety -- but actually less than for nuclear disposal
    - Accident consequences are smaller; hysteria is less
  - Sufficient payload capacity
  - Low peak acceleration
    - Apollo was ~5 G's; Shuttle is ~3 G's
    - Good shock absorber required (<1 G vibration?)
      - Easy to do in a large vehicle with a high pulse rate
- Payload capacity needed is clearly less than 1 ton (a Mercury capsule):
  - Better structures, electronics available
  - Minimal life-support needed
    - Normal dock-or-reenter in ~2 hrs (1 orbit)
      - Assumes synchronized launch; 2-4 "windows" per day
    - Worst-case dock-or-reenter in ~24 hours
  - Minimal guidance system (Must have some, to prevent tumble)
  - Baggage goes up first! (Limit 1 carryon, must fit under your seat)
  - Potentially ~300 kg, but must include:
    - Person (up to 100 kg)
    - Couch
    - Air/water/power
    - Pressure shell
    - Emergency reentry system (pared to minimum mass via extensive tests)
- G-limit:
  - Drives system to long range, high $I_{sp}$
  - 1000 km range gives 5-6 G's for last few seconds @ 800 s $I_{sp}$
    - ~12 G's at 400 s
  - Thrust is constant, so acceleration peaks sharply at end of launch
    - Trivial to throttle system -- just reduce laser pulse rate
    - But good shock absorbers will be a necessity

## Acknowledgements

My thanks to Arthur Kantrowitz, Dennis Reilly, and Chris Rollins for their discussions of laser propulsion issues, and to Leik Myrabo for his innovative suggestions for laser propulsion vehicle designs. This work was supported by the Directed Energy Office of the U. S. Strategic Defense Initiative Organization, and carried out at Lawrence Livermore National Laboratory under the auspices of the U. S. Department of Energy, contract No. W-7405-ENG-48.

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[7] E.g., R. L. Forward, "Roundtrip Interstellar Travel Using Laser-Pushed Lightsails", J. Spacecraft, Vol. 21, No. 2 (Mar.-Apr. 1984).

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[9] A. Kantrowitz, personal communication.

[10] E. E. Rice, R. S. Denning, A. L. Friedlander, "Preliminary Risk Assesment for Nuclear Waste Disposal in Space" Vol. I. NASA CR-162028 (1982).

[11] The shielding requirement was pointed out at the Vision-21 Symposium by both Theodore Taylor and Frank Rom; the waste/shielding mass ratio is a function of the total mass, and is lowest (worst) for masses of 10's of kg.

C-4

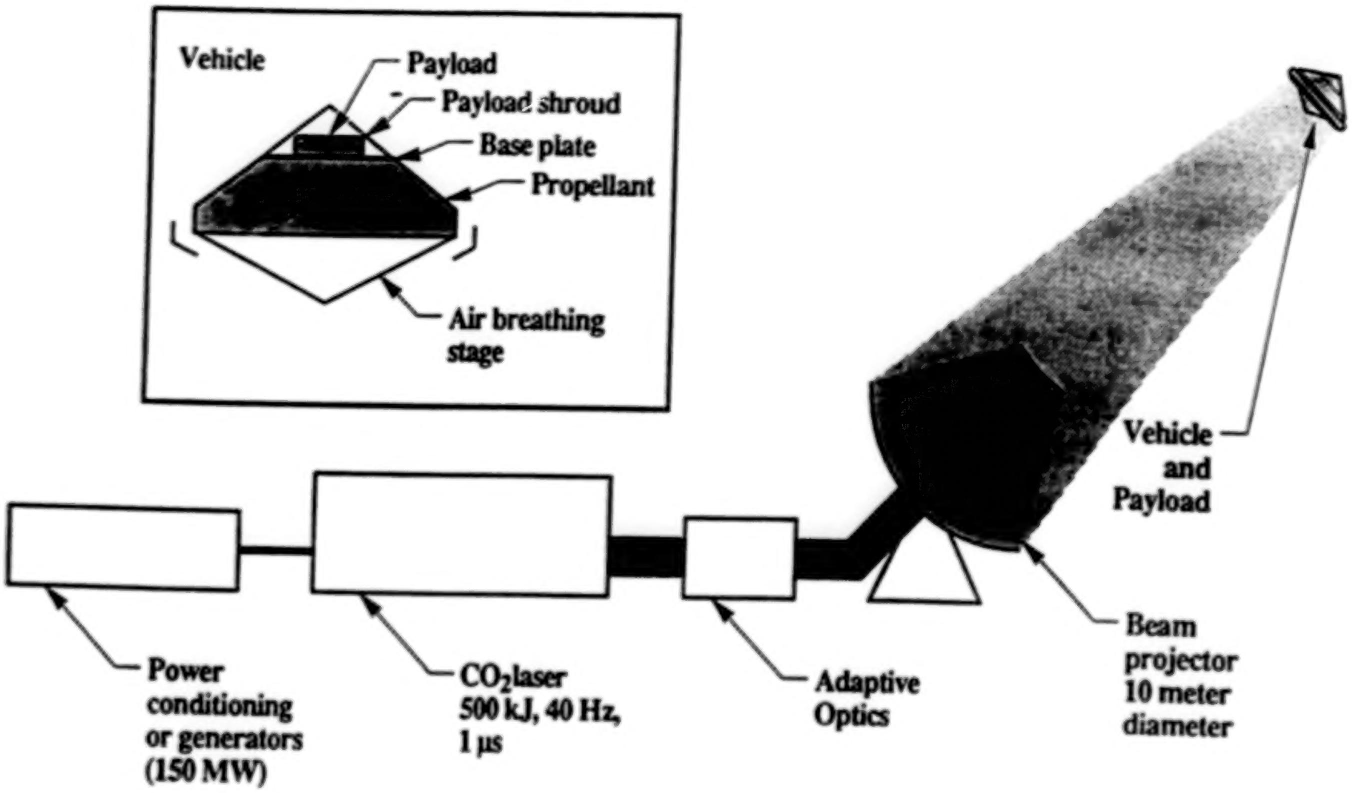
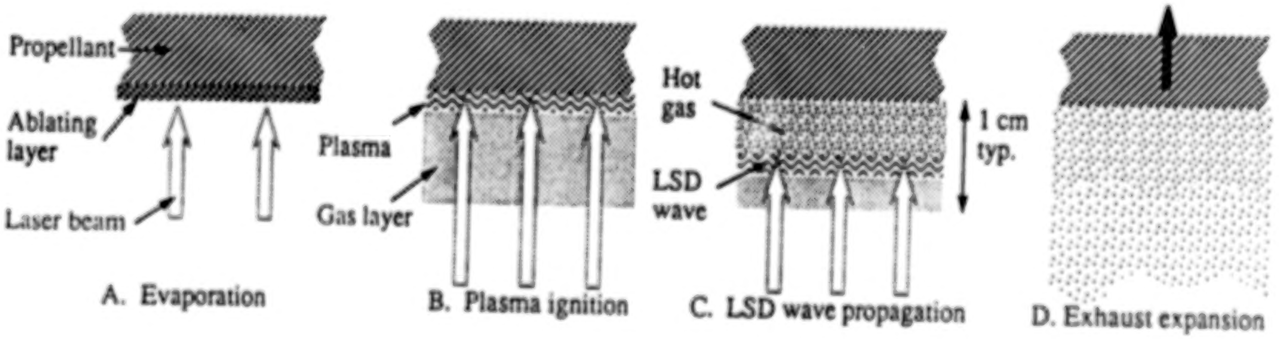

Vehicle
Payload
Payload shroud
Base plate
Propellant
Air breathing stage
Vehicle and Payload
Beam projector 10 meter diameter
Power conditioning or generators (150 MW)
$CO_2$ laser 500 kJ, 40 Hz, 1 μs
Adaptive Optics

**Figure 1:** Components of a 20 MW/20 kg Laser Launch System


Propellant
Ablating layer
Laser beam
Plasma
Gas layer
Hot gas
LSD wave
1 cm typ.

A. Evaporation B. Plasma ignition C. LSD wave propagation D. Exhaust expansion

**Figure 2:** Double-Pulse Thrust Cycle

N91-22158

# MOMENTUM HARVESTING TECHNIQUES FOR SOLAR SYSTEM TRAVEL

Alan J. Willoughby
Analex Corporation
April 1990

## ABSTRACT

Astronomers are lately estimating there are 400,000 Earth visiting asteroids larger than 100 meters in diameter! These asteroids are uniquely accessible sources of building materials, propellants, oxygen, water and minerals. They also constitute a huge momentum reserve, potentially usable for travel throughout the solar system.

To use this momentum, we must track these stealthy objects, and learn to extract the momentum we want. This paper discusses momentum harvesting by momentum transfer from asteroid to spacecraft, and by using the momentum of the extraterrestrial material to help deliver itself to our destination.

A net and tether concept is the suggested means of asteroid capture, the basic momentum exchange process. The energy damping characteristics of the tether will determine the velocity mismatch that can be tolerated, and hence the amount of momentum that can be harvested per capture. As it plays out of its reel, drag on the tether steadily accelerates the spacecraft.

This paper discusses a variety of concepts for riding and using the asteroid after capture. The hitchhiker uses momentum transfer only. The beachcomber, the caveman, the swinger, the prospector, and the rock wrecker also take advantage of raw asteroidal materials. The chemist and the hijacker go further, they process the asteroid into propellants.

Or, an "asteroid railway system" could evolve with each hijacked asteroid becoming a scheduled train. Travelers could board the space railway system assured that water, oxygen and propellants await them.

## THE OPPORTUNITY

Mother Nature has provided an abundance of interplanetary resources for those beings observant enough to see them and wise enough to use them. Earth visiting asteroids provide material wealth in a uniquely useful place, near Earth yet beyond the depths of its gravity well. Too small and too dark to be easily seen, we are discovering lately that **Earth-visiting asteroids are numerous, not rare.**

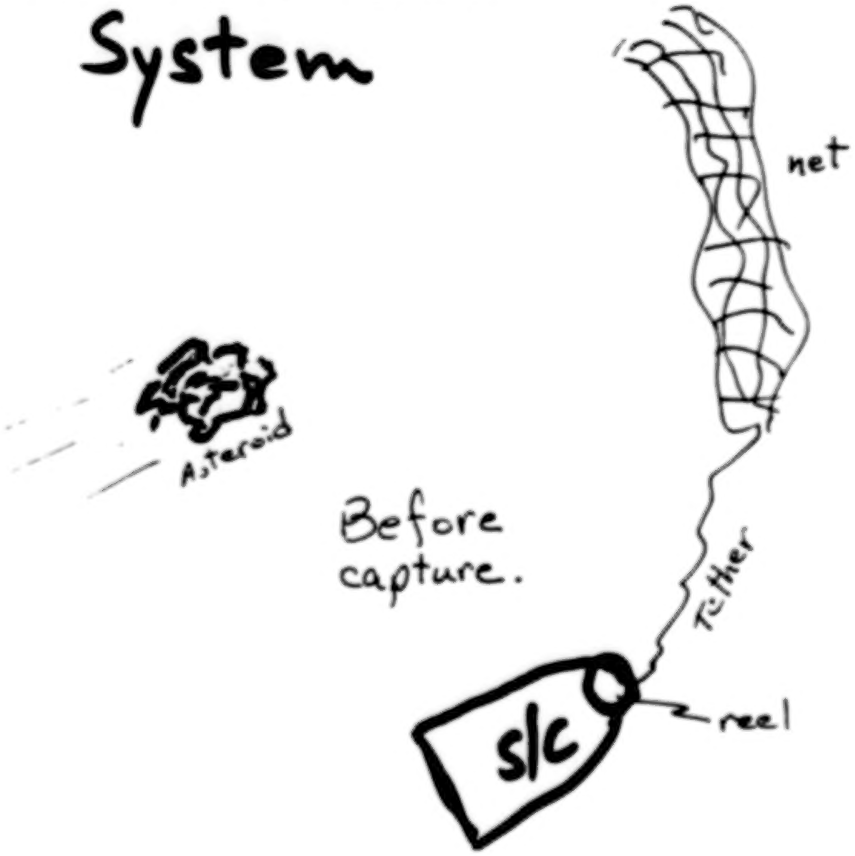
To quote the enthusiasm in The SERC Newsletter of December 1989 from the NASA/University of Arizona Space Engineering Research Center:

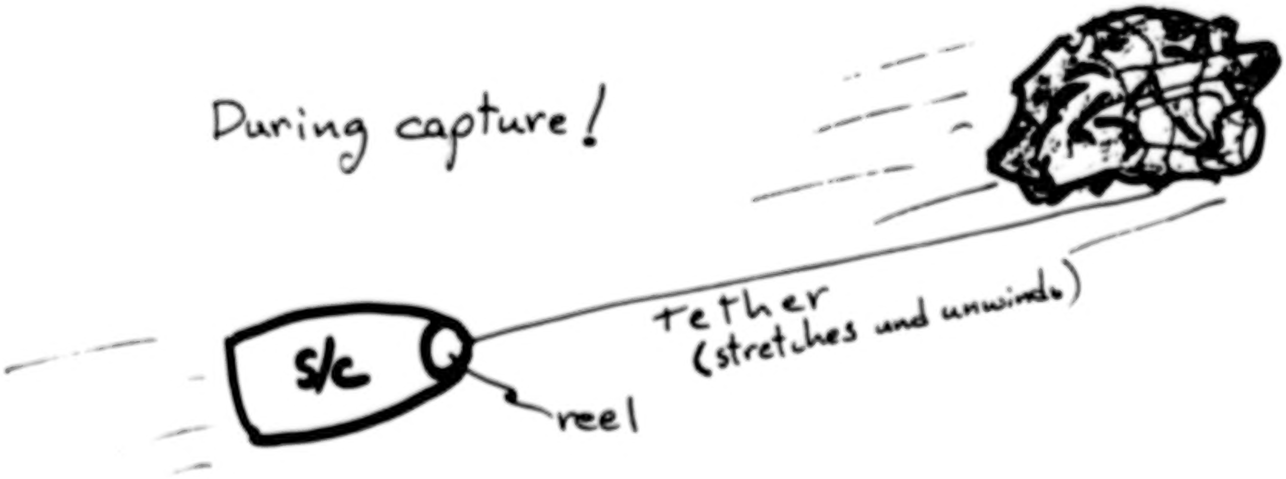
"Observations such as those made by ... the University of Arizona and ... JPL ... suggest that there are 1000 to 2000 kilometer-sized near-Earth asteroids, and roughly 400,000 larger than 100 meters in diameter. ... we can begin to build confidence that life-support, building, shielding, and propellant materials will be available where and when we need them."

And, **momentum is an asteroidal resource also worth harvesting!** There are two aspects to momentum harvesting. First, **select bulk extraterrestrial mass which already has favorable momentum** for its delivery to a desired destination. For example, select asteroids with low return delta velocities as sources of material in earth-lunar space. Second, **exchange momentum** from these fast flying natural objects to our spacecraft.

Their great numbers means there may indeed be an asteroid "going our way" where and when we need it. To really use these flying fragments of fine fortune, we must first track these stealthy objects, then learn to extract the momentum we want.

# Net & Tether System


Before capture.
net
Asteroid
Tether
s/c
reel


During capture!
s/c
reel
tether (stretches and unwinds)

John Lewis, a planetary resources expert at the University of Arizona, observes that current technology can be deployed to find and track earth crossing asteroids, as well to deflect, destroy or use them. His book, SPACE RESOURCES Breaking the Bonds of Earth, is strongly recommended. The concepts described here add the conjectured possibility of catching rides on asteroids, to further enhance the already great opportunities Lewis has identified.

## NET AND TETHER CAPTURE APPARATUS

The challenge of momentum exchange is to dilute, in time, hypervelocity collisions to the point where they become constructive interactions. A net and tether concept is suggested as a fundamental means of asteroid capture.

The net and tether isolates the collision from the spacecraft. The net must be designed to survive by virtue of low mass, great strength and prudent pre-acceleration. The tether should have large energy damping characteristics, acting as one very long sequence of dash pots, to prevent whiplash and "yo-yoing". The state-of-the-art of energy absorption technology will determine the amount of velocity mismatch that can be tolerated, and hence the amount of momentum that can be harvested in a single capture. What we can accomplish depends, quite literally, on how far we can stretch the technology.

As it plays out of its reel, drag on the tether steadily accelerates the spacecraft. It's vital to remove the initial angular momentum. If this isn't accomplished by tangential drag, then it must be completed by propulsion or by using a third body.

Many applications are possible after asteroid capture. Some options are described qualitatively.

## THE BEACHCOMBER

The beachcomber opportunistically harvests the bounty of small asteroids washed his way by the celestial tides. As he snags the small asteroids in his nets, he incrementally gathers both momentum and useful materials.

Since he gathers small stuff, the beachcomber cannot expect much help from the asteroid tracking network. But he has many mini-asteroids he can harvest, so he is not very dependent on chance. He must selectively harvest the asteroids going his way. Just as earthly beachcombers carry metal detectors to discover their best finds, the space beachcomber will likely carry his own detection and tracking system, too. It will provide him reaction time to see and catch asteroids flying in his preferred direction.

## THE HITCHHIKER

The hitchhiker hangs onto an asteroid until his desired velocity is reached. He then releases the tether system, either by severing the tether or by disengaging the entire mechanism. By using a few net-tether devices, he can hitch a series of rides.

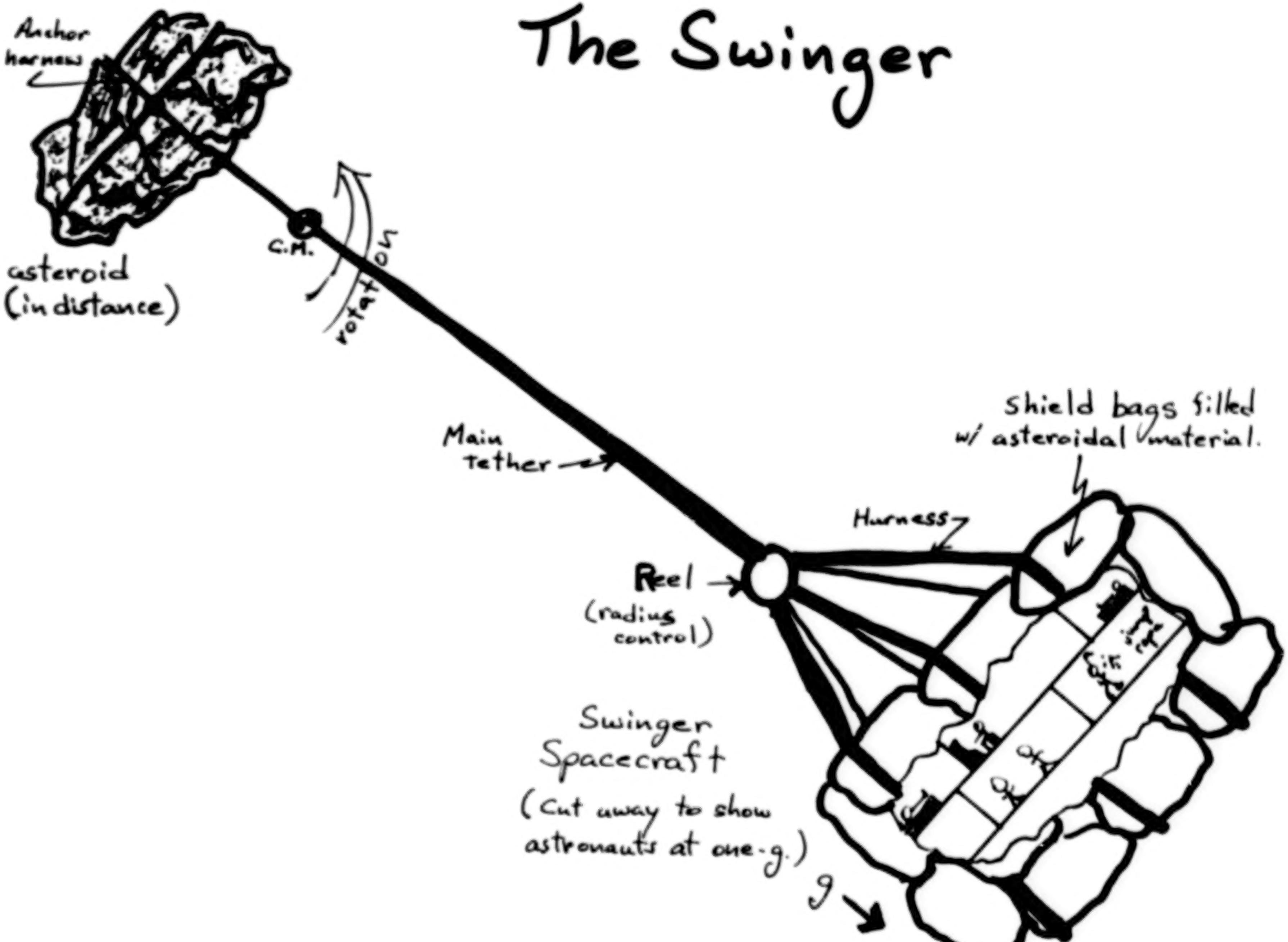
The hitchhiker uses ordinary propulsion to perform course correction, to reach his desired destination at just the right moment. Small early corrections can easily make significant changes at destination.

Prior cataloging of earth visiting asteroids or a complete active deep space tracking network will be vital to hitchhikers, so their rides out and their returns home will not be left to chance.

## THE CAVEMAN

The caveman seeks shelter from cosmic rays and solar flares. After his tether plays out and he has reached equal velocity with the asteroid, he reels in the tether and climbs aboard. It is relatively easy for him to burrow into most

# The Swinger


Anchor harness
asteroid (in distance)
C.M.
rotation
Main tether
Reel (radius control)
Harness
Shield bags filled w/ asteroidal material.
Swinger Spacecraft (Cut away to show astronauts at one-g.)
g

asteroids.

Once inside, the caveman has better protection from highly penetrating cosmic radiation than any other space traveler, and solar flares are of absolutely no consequence. The other comforts of home? They need some planning and preparation.

## THE SWINGER

The swinger also wants the benefits of gravity during his ride through the solar system. After pulling himself aboard the asteroid, he pulverizes as much material as he needs and stuffs this free shielding into expandable compartments on the exterior of his craft. The swinger then pushes off and plays out his tether. At a selected distance he stops the tether, then thrusts tangentially to create gravity.

Earthly gravity is his most logical choice, but any g-level is possible from centrifugal force. If sturdy anchor ground cannot be found, the swinger can tie up the asteroid like a package, to prevent unplanned separation. The initial asteroid spin rate and tether moment arms must be taken into account to prevent the swinger from getting wound around the axle, so to speak. Despinning the asteroid or tether play out are possible solutions.

The clever swinger can pick up more free velocity in any direction by playing with angular momentum. A small velocity at a very long radius will change to a much larger velocity as the tether is shortened. A well timed tether cut will send the swinger off on the next leg of his journey, with his free cargo of shield and raw materials.

## THE PROSPECTOR

The prospector wants big profits with little work. He gathers and sells raw asteroid materials.

The prospector knows that each earth visiting asteroid has a low delta velocity to somewhere in the inner solar system. Each such destination is one of his orbital marketplaces. His tanks are salvaged from the jetsam of others. He stuffs all the empty tanks he can tug with asteroidal matter, chunked or ground for best packing.

After selling his raw materials at the most convenient market, he buys his needed propellant right there from the processors. This propellant is now quite cheap, thanks to his supplying inexpensive feedstock.

In response to market prices, some prospectors may pass up asteroids of opportunity to wait for richer mineral-bearing asteroids heading toward high profit ports, such as Earth.

## THE CHEMIST

The chemist is a bit more enterprising than the prospector. Rather than transport heavy, bulky raw materials, she sets up propellant production plants right there on the asteroid.

These plants give her all the propellant she needs to ship her finished products to easily accessible orbital markets. These products include not only fuels and oxygen, but also water and other precious chemicals not available from the Moon. She can greatly underprice her competitors shipping from Earth, Mars or the Moon.

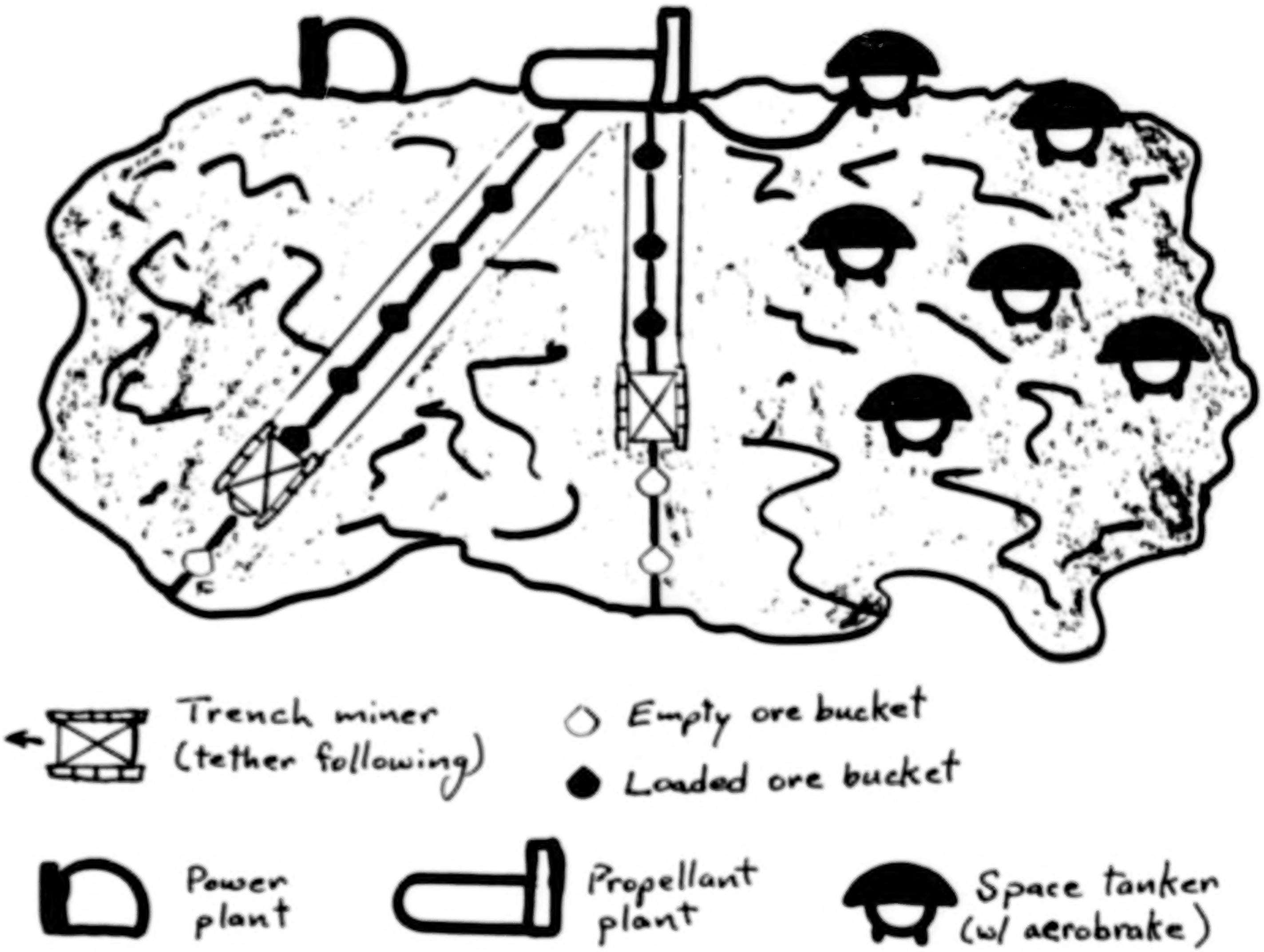
Nuclear power gives the chemist her high production rates and high profits. Automation and electric propulsion will make her future operations even more profitable.

## THE ROCK WRECKER

The rock wrecker is a gatherer and seller of raw materials, but with much grander ambitions than the prospector. She delivers huge chunks of asteroid, intact!

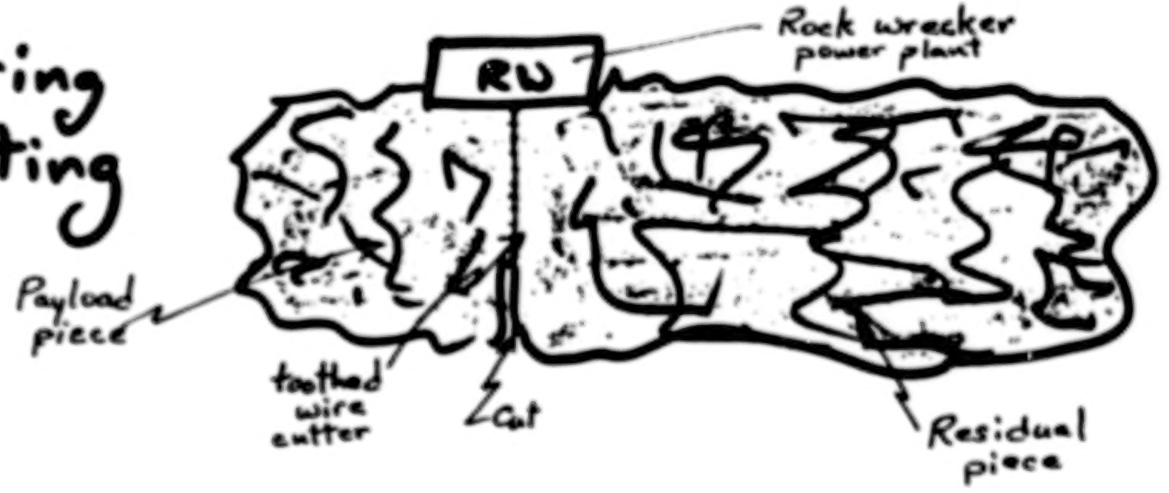
After reeling in her tether and boarding an asteroid, the rock

# The Chemist

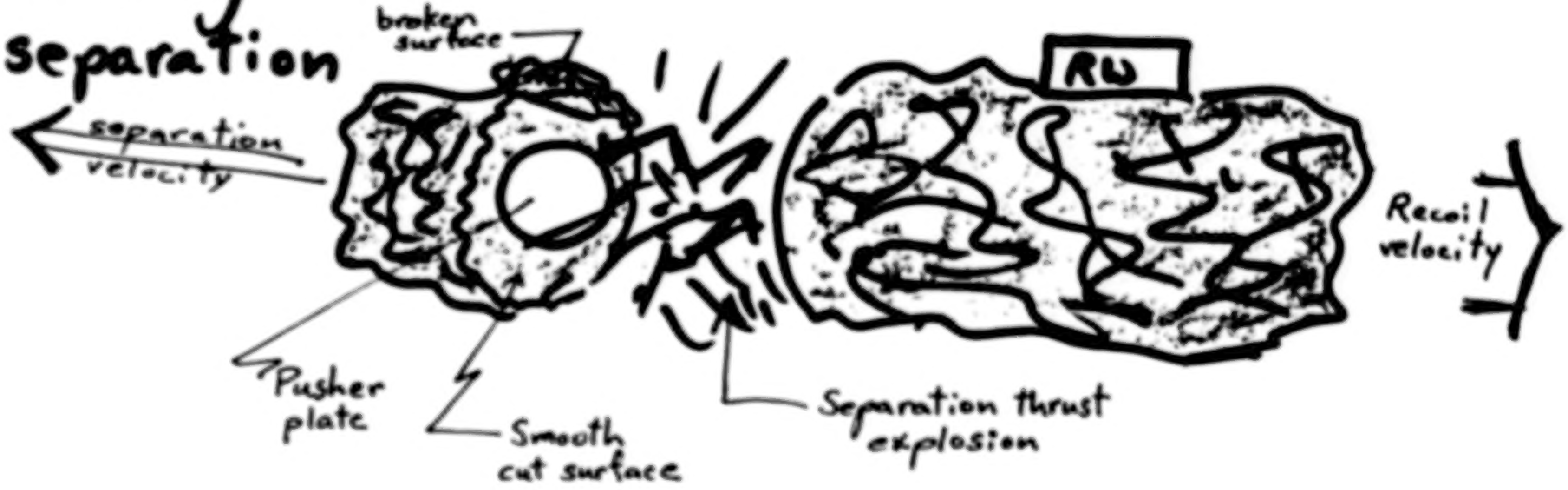

Trench miner
(tether following)
Empty ore bucket
Loaded ore bucket
Power plant
Propellant plant
Space tanker
(w/ aerobrake)

# The Rock Wrecker

**During cutting**


RW
Rock wrecker power plant
Payload piece
toothed wire cutter
Cut
Residual piece

**During separation**


broken surface
separation velocity
RW
Recoil velocity
Pusher plate
Smooth cut surface
Separation thrust explosion

wrecker sets about slicing off as large a chunk as she can deliver to market. By keeping it intact, she avoids the limitations of tankage. Through use of a long toothed wire cable, slicing is not too difficult. The challenge is geometric, rather than material toughness.

Delivering the payload piece takes clever application of Newton's third law. The residual asteroid acts as the reaction mass for conservation of momentum, thus avoiding the need for large quantities of propellant. By choosing the relative size and orientation of the pieces at separation, the rock wrecker selects just where the pieces go. The payload goes to market. The residual piece may go to a better orbit for later rock wrecking.

The pieces must not drift apart before full separation thrust is applied. One trick is to not fully sever the asteroid. The separation thrust can be a long steady push, or an explosive force. Pusher plates prevent the pieces from shattering.

### THE HIJACKER

Hijackers are not content with tanks of raw material, tanks of propellant, or even large chunks of asteroid. Hijackers take whole asteroids!

One readily envisioned hijacker is a nuclear powered propellant plant combined with a nuclear thermal rocket. Propellant would be produced steady state, then burned in short bursts to adjust the asteroid's orbit. Perihelion burns would adjust aphelion, and vice versa. Other burns could adjust eccentricity and inclination.

Multiple burns are needed at each point. Each hijacking is a long term operation, requiring many orbits and many years. Astronauts could set up the system, during a trip to Mars for example, then let the automated system do its job. The trip to set up the hijacker can be free, or even profitable, since momentum is harvested and resources can be gathered during this detour. Hijacking jobs would eliminate wasted time, and even boredom, while traveling back and forth through the solar system.

Hijacked asteroids could be destined for planetary capture (assisted by aerobraking) and orbital consumption. Asteroids in orbit would give Earth the same advantage as Mars, which has Phobos and Deimos as ready sources of orbital consumables.

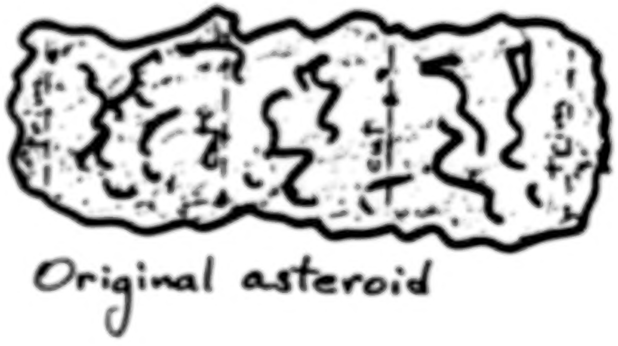
### ASTEROID RAILWAY SYSTEM

Perhaps even easier and better than total asteroid consumption may be the development of an "asteroid railway system". Each hijacked asteroid becomes a regularly scheduled "train" linking specific parts of solar system. The railway system evolves to provide more and more frequent travel opportunities as each link is added.

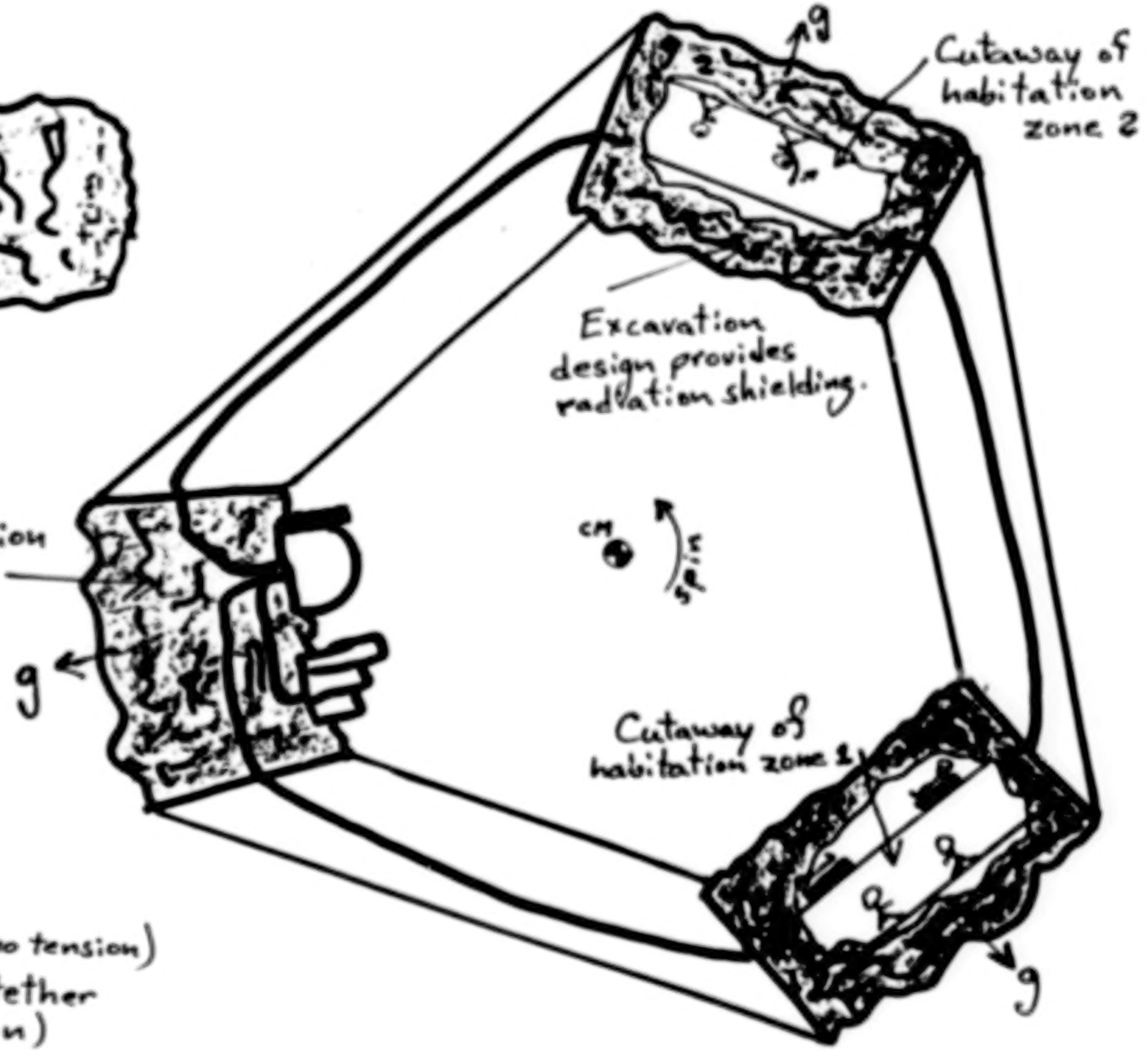
This space railway network would have the advantages of cycling spacecraft, but could be made more robust by use of natural resources, i.e. the asteroids. Space travelers could use lightweight craft to board and exit the railway system, because of assurances that shielding and supplies of water, oxygen and propellants await them. Momentum harvesting while boarding is an added benefit.

Asteroids could be improved by entrepreneurs. A luxury liner may consist of sliced asteroid pieces, rejoined by tethers, then spun up gently to provide safe, habitable compartments complete with earthly gravity. This cruiser would be a product of the hijacker, rock wrecker, caveman, chemist and swinger. The austerities of pioneer space travel would give way to creature comforts and luxuries, all with a speed and economy impossible without nature's gift of earth visiting asteroids.

# Small Cruiser on Asteroid Railway



Original asteroid


g
Cutaway of habitation zone 2
Excavation design provides radiation shielding.
CM
Spin
Production zone
g
Cutaway of habitation zone 1
g

Power plant

Propellant plant

━ Power cable (no tension)

─ Structural tether (under tension)

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Lewis, John S. and Ruth A. Lewis, SPACE RESOURCES Breaking the Bonds of Earth, Columbia University Press, New York, NY, 1987. (Available from ASPERA, Inc., 1745 North Campbell Ave., Tucson, AZ 85719.)

## ACKNOWLEDGMENTS

Thanks to Robert E. English (Distinguished Research Associate, NASA Lewis Research Center) for encouragement, consultation and several creative contributions, most notably the "swinger."

N91-22159

# THE CENTAURI PROJECT: MANNED INTERSTELLAR TRAVEL

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Houston, Tx. 77388

## ABSTRACT

The development of antimatter engines for spacecraft propulsion will allow man to expand to our nearest stellar neighbors such as the Alpha Centauri system. Compared to chemically powered rockets like the Apollo mission class which would take 50,000 years to reach the Centauri system, antimatter propulsion would reduce one way trip time to 30 years or less.

The challenges encountered by manned interstellar travel will be formidable. The craft must be a combination sub-light speed transportation system and a traveling microplanet serving an expanding population. As the population expands from the initial 100 people to ~ 300, the terraformed asteroid, enclosed by a man-made shell will allow for expansion over its surface in the fashion of a small terrestrial town. All aspects of human life - birth, death, physical, emotional and educational needs, government and law must be met by the structure, systems and institutions on board.

## INTRODUCTION

Interstellar space exploration is at once profoundly exciting and profoundly frightening. The isolation and vast distances along the paths of these starlines are beyond the understanding of today's civilizations. The Alpha Centauri star system, our Sun's closest stellar neighbor, lies 4.3 light years away from us. This is the equivalent of just over twenty-five trillion miles. A one way trip to Proxima Centauri, the closest member of this three star system, will require 30-50 years, depending on the minimum cruise velocity required (between 10 to 30% of the speed of light).[ref. 7]

If the exploration of the solar system is the Third Great Age of Discovery as suggested by Pyne [ref. 12], the exploration of star systems will herald the dawn of the Fourth Age of Discovery. Though each age ultimately deals with worlds beyond the terrestrial sphere, the discovery and exploration of stellar systems is fundamentally different from the discovery of abiotic planets within the solar system. The rift valleys, ice caps and red deserts of Mars, the volcanoes of Io and the great cliffs of Miranda are images that our civilization can understand through terrestrial counterparts. Images of stellar systems bear few if any physical, temporal or social symbols recognizable to our civilization.

It is necessary for the structure of a paper such as this to make assumptions. The time frame of this mission is placed in the latter part of the twenty-first century - when we assume that man has explored and colonized the majority of Solar Systems. This colonization effort has established a network of manufacturing plants and O'neill type colonies amongst the Jovian planetary systems.[ref. 11] Construction in space is a mature technology, as is materials processing and the use of antimatter engines for propulsion. Finally, it is assumed that non-centrifugal artificial gravity has been developed and implemented in a number of vessels and orbiting colonies.

## THE INTERSTELLAR VESSEL

The interstellar vessel for the Centauri Project will consist of an asteroid of the stony carbonaceous chondrite class, with a nominal diameter of approximately two miles. Surrounding this asteroid will be a man-made shell, best defined as a '2-manifold in 3 space' [ref. 8], which folds in on itself at each end along the primary axis. (See figures 1 & 2). This shell, approximately 60 feet thick and containing three graduated pressure levels, encloses an atmosphere that allows the crew to live on the surface of the asteroid.

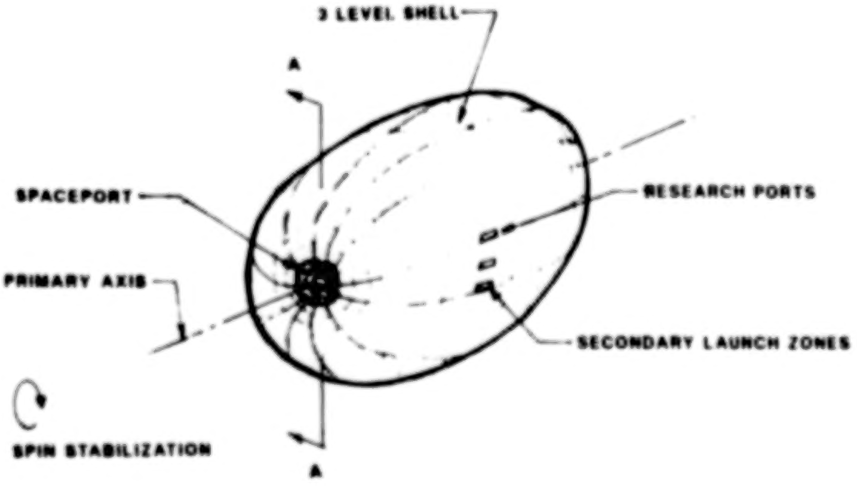

FIGURE 1: EXTERIOR VIEW OF INTERSTELLAR VEHICLE

Upon enclosure, the surface of the asteroid will be terraformed to contain a variety of Earth ecosystems: forests, farmlands, small lakes, etc. Highly automated food production techniques will be incorporated to control highly automated agriculture, fish farming, husbandry and hydroponics. An asteroid measuring slightly less than four kilometers across has a surface area of roughly 9,000 acres, satisfying Nasa's guidelines for area requirements of habitation, and food production. [ref. 2,9]

## Livability Requirements

The habitable surface of an asteroid serving as a microplanet creates an environment which will appear quite Earth-like. The importance of Earth-like living conditions becomes evident through positive effects of privacy, mobility and the reduction of the closed environment feeling. Provisions for long lines of sight, views of a horizon and large overhead clearances - all possible with this vessel style - offer enormous benefits to intellectual and emotional well being of crews on long duration missions. [ref. 6] Combining these provisions

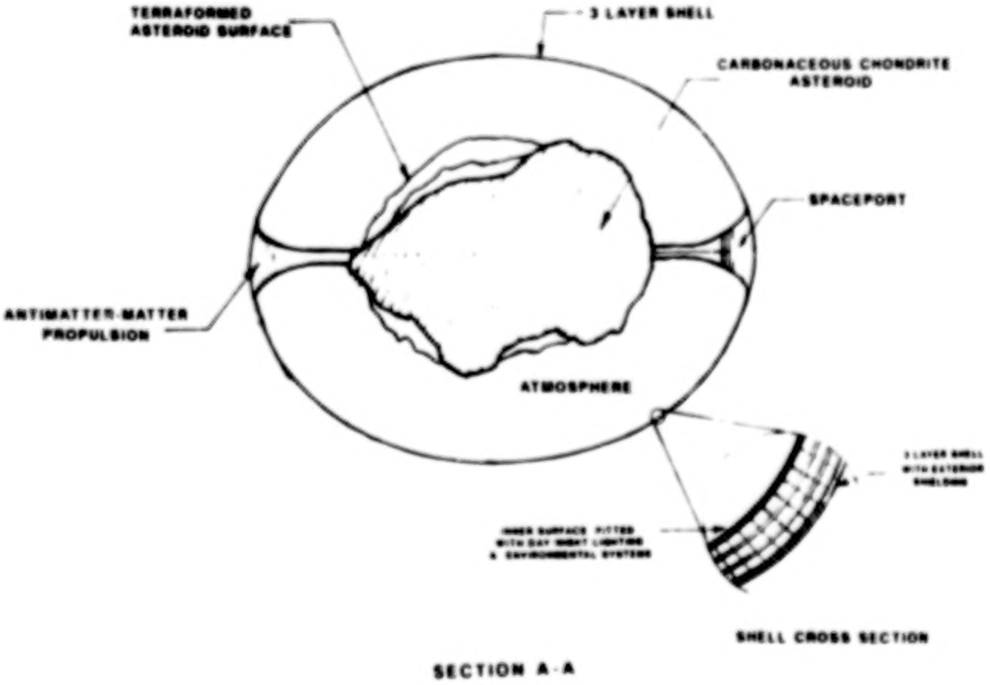
TERRAFORMED ASTEROID SURFACE
3 LAYER SHELL
CARBONACEOUS CHONDRITE ASTEROID
SPACEPORT
ANTIMATTER-MATTER PROPULSION
ATMOSPHERE
3 LAYER SHELL WITH EXTERIOR SHIELDING
INNER SURFACE FITTED WITH DAY NIGHT LIGHTING & ENVIRONMENTAL SYSTEMS
SHELL CROSS SECTION
SECTION A-A

FIGURE 2: CROSS SECTION THRU INTERSTELLAR VEHICLE

with a variability factor in the environment will provide the stimulation necessary to suppress the symptoms of Solipsism. Solipsism is a state in which a person feels that everything is a dream and nothing is real. This state of mind often occurs in the Arctic winter when darkness lasts 24 hours a day. A person feels lonely and detached, eventually becoming apathetic and indifferent. Variability will be enhanced by proper asteroid selection which will provide varied topography allowing for views of natural objects such as hills and valleys on the terraformed surface.

The ability to invoke randomized environmental patterns is crucial to the goal of allowing this vessel to simulate a microplanet. To achieve randomized patterns, a 'unpredictability factor' is included in the systems programming. The program then controls day/night illumination cycles, temperature variations and even the creation of breezy conditions. Of the randomized systems discussed, the day/night cycle - serving as a zeitgeber - is perhaps the most important. Zeitgebers are physical, temporal and social cues that help to establish rhythmicity in the sleep/wake cycle. Social cues such as daily meals, work/rest schedules and evening leisure time also reinforces the circadian rhythm, which enhances performance.

## The Shell

The shell of our vessel bears a resemblance in appearance and function to the shell of an egg. Attached to the asteroid at only two points - where the manifold folds in on itself - the shell rotates with the asteroid for spin stabilization around the primary axis. As an extension of the asteroid's terraformed surface, the shell provides livable spaces throughout selected zones of the triple level structure. These three levels, having the potential of providing habitable space will serve as laboratory, communications and scientific research spaces, as well as distribution corridors for various environmental and recycling systems.

The involuted areas at each end of the shell provide specialized areas for propulsion systems on one end and space port facilities on the other . The inward folds of this shell which reach the surface of the asteroid will serve as main traffic and systems corridors between the surface and the shell. Elevator systems within the involutions and in the shell proper will provide access to habitable shell zones as well as the activities zones housed within the involutions.

The spaceport (see Fig. 3 ), will provide the crew with the ability to launch a variety of shuttles, space probes and experiment packages, as well as accept incoming personnel and cargo vessels. As the leading edge of the vessel, a concentration of navigation and scientific decks will also be located here.

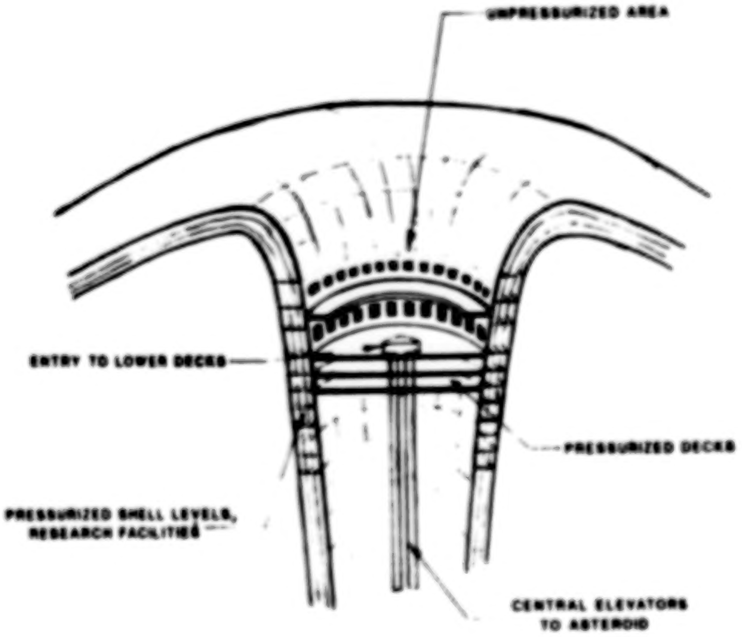
UNPRESSURIZED AREA
ENTRY TO LOWER DECKS
PRESSURIZED DECKS
PRESSURIZED SHELL LEVELS, RESEARCH FACILITIES
CENTRAL ELEVATORS TO ASTEROID

FIGURE 3: DETAIL OF SPACEPORT

### Antimatter Propulsion & Power Systems

It is highly probable that first generation interstellar vessels will use multiple energy sources to fulfill the needs of a vessel of such magnitude. Antimatter engines for propulsion will be required during the boost phase for between one to two years of constant acceleration. During this boost phase and through the mission, power requirements for light, heat and life support systems will be large, given the vessels volume. The antimatter engines may not be the best source of energy for daily power demands. Multiple fusion reactors distributed within the shell and below the surface of the asteroid may be a practical configuration.

The viability of antimatter as a propulsion source has been under investigation by many [ ref.'s 5,14,15]. When a particle and its antiparticle collide at a sufficiently low energy level, annihilation results in a high conversion of their mass into kinetic energy of other particles, photons and neutrinos. To harness and control this energy only kinetic energy and photons may be utilized. Therefore a matter - antimatter annihilation propulsion system is a device which would use these to produce thrust.

Since stable, long lived particle-antiparticles should be considered for propulsion - only electron-positron and nucleon-antinucleon annihilation reactions have potential for space propulsion. The advantage of the latter pair is the larger amount of energy released by these collisions. For this discussion, we will therefore assume that an antiproton-proton or antiproton-neutron annihilation will be used for the propulsion system, stored perhaps as magnetically levitated diamagnetic balls of antihydrogen, or a levitated ball of hydrogen with molecules of antihydrogen embedded in the crystal lattice of the ball.[ref. 12]

In general, matter-antimatter engines utilize the mixing of antimatter(antihydrogen) and matter(hydrogen) together in an annihilation chamber to interact, annihilate and exhaust a high velocity energetic plasma. Figure 4 shows one such concept of a matter-antimatter engine design.

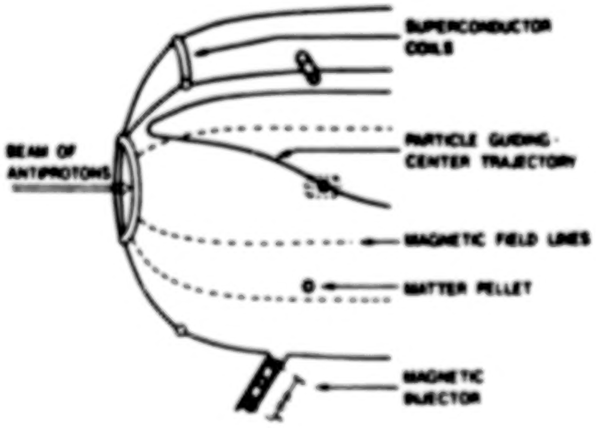

FIGURE 4: CONCEPT FOR ANTIMATTER ENGINE
(REF. 14)

# THE CREW

We will assume that this interstellar vessel is equipped with highly automated systems. Though not attaining the sophistication of a 'von Neuman probe'*[ref. 1], these systems will nonetheless be capable of self monitoring and autonomous repair. With this level of automation in power, life support, food production and other critical functions, smaller crew sizes become practical. One benefit of this automation is the time available for research and experimentation during the mission.

## Crew Size

The social dynamics of small groups, staffing requirements, and long term population growth must be considered before establishing initial crew size. Too small a group may create understaffing during a crisis situation. Social variety , necessary for individual fulfillment, is dependent on the number of available social contacts. A crew of thirty or less may be incapable of providing this social variety. On the other hand too large of a group may create control problems for the leadership and may contribute to runaway birth rates.

For this discussion, our crew size will total one hundred people: forty men, forty women and 20 children. Because we will need a young healthy crew for the start of this mission, the adults will be between 21 and 26 years old. The children will range from five to ten years old. There are positive and negative aspects regarding the inclusion of children at the beginning of this mission. On the negative side, children require a great deal of time and attention, something that may be scarce early in the mission. They must also be educated, requiring the dedication of crew members to this task.

On the positive side, these children represent the first of the next generation of adult crew members. This reduces the need for females to produce offspring almost immediately after the mission begins. It could be argued that the stresses involved in pregnancy and childbirth would outweigh those associated with raising children. The inclusion of children will also create a social structure closer to the mix of peoples on Earth. It has been speculated that this inclusion of the family structure early in this mission will provide social stability to the crew. [ref. 4]

*. The 'von Nueman probe' is a self-reproducing universal constructor, capable of making any device, given the materials and construction programs.

# THE SOCIAL ENVIRONMENT

The crew of an interstellar vessel embarking on a long duration mission will enter a microsociety which will have its own rules, benefits and hardships. Analogous to air-born seed pods of terrestrial plants, this social pod will take the elements of humanity into the galaxy. The remoteness of an interstellar crew possess special problems for mission planners, particularly in the area of organizational and management structure.

## Leadership

Almost all of the space efforts by the USA and the USSR have used military personnel to staff the missions. The use of a military organizational and management structure has proved successful in almost 100% of these missions, especially in those missions where crises has occurred. This military authority structure may not work however, for a crew made up of non military personnel.

It is probable that on a multi-decade mission, the authority structure in place at the outset of the mission may have been completely abandoned for an alternate style later. The isolation, confinement and risk of long duration missions may prove to undermine authority structures and increase demands on the leaders. In addition there is the issue of a multi-generational crew coming of age on an interstellar vessel. The challenge will be to instill a commitment to the mission.

The point is that whether a military based model or a community democracy model is used, each has advantages and disadvantages of its own. No one model will work the best for all situations. As mankind populates the Solar System, novel alternatives may evolve which may be better adaptable to interstellar travel.

## The Life Cycle

Unlike most space efforts today, the drama of the life cycle will be a predominant part of the Centauri project. Birth, aging and death will be present as will the many anomalies associated with the process. As the population expands, crime will likely exist, perhaps even murder; requiring social systems to be in place to deal with events which could effect the performance of the crew. The existence of marriage and the family unit also presupposes the existence of divorce. The practicality of divorcing a spouse in such a closed society is an unknown factor, but it will certainly require contingency planning.

Unique to this type of mission is the simple act of living out a life in space. The day to day activities and demands of mental, physical and social maturation must be dealt with in addition to responsibilities of a crew member on the mission. An aside to this life cycle is the unprecedented fact that humans will be born with lives totally disassociated from the planet Earth. For those crew members born as part of the first Centauri generation, the microplanet serving as their starship may well be all they will know of life.

Space travel, particularly long duration missions, involves the sacrifice of personal freedoms. For those individuals born aboard the Centauri vessel another basic freedom will be lost - free choice of a profession. Because the vessel and systems must be maintained, many positions must be manned for the entire length of the mission. This means that replacements must be trained for the original crew, requiring a portion of the descendants to be selected for a specific function. The process and timing of this selection may best be left to social norms developed within the society.

SUMMARY OF VESSEL PARAMETERS

| | | |
|---|---|---|
| PROJECTED SURFACE AREA OF ASTEROID: | 9000 Acres | |
| PROJECTED EXTERNAL SURFACE AREA OF SHELL: | 9x$10^7$ $m^2$ | |
| MAXIMUM HABITABLE SURFACE AREA WITHIN SHELL STRUCTURE : | 1.25x$10^8$ $m^2$ | |
| ATMOSPHERIC COMPONENTS: (1 standard atmosphere=101kPa) | (kPa) | (mmHg) |
| $O_2$ : | 22.7 (±5) | 170 |
| $N_2$ : | 26.6 | 200 |
| $CO_2$ : | < .4 | <3 |
| TOTAL PRESSURE: | 50.0 | 380 |
| WATER VAPOR : | 1.0 (± .33) | 7.5 (± 2.5) |
| TEMPERATURE: | 23° 4°C | |
| RELATIVE HUMIDITY: | 50 ± 10 percent | |
| PSEUDOGRAVITY : | .95 ± .5g | |
| AVERAGE OVERHEAD CLEARANCE: | 914 m | |
| LONGEST LINE OF SIGHT: | 2414 m. | |
| PORTION OF HABITAT HIDDEN FROM VIEW : | 90% | |
| ILLUMINATION CYCLE : | STANDARD TERRESTRIAL DAY (Variable) | |

TABLE 1:
SUMMARY OF VESSEL PARAMETERS

SUMMARY OF DESIGN CONSIDERATIONS

LONG LINES OF SIGHT
LARGE OVERHEAD CLEARANCE
VIEWS OF LARGE NATURAL OBJECTS
PORTION OF HABITAT HIDDEN FROM VIEW
WATER VISTAS: SMALL LAKES
ACOUSTICS OF CHAMBER CREATED BY SHELL
AVAILABILITY OF PRIVACY
PLACEMENT OF MANUFACTURING FACILITIES UNDERGROUND
UNPREDICTABILITY FACTOR OF THE ENVIRONMENT

TABLE 2:
SUMMARY OF DESIGN CONSIDERATIONS

ORIGINAL PAGE IS
OF POOR QUALITY

# THE MISSION

Following the enclosure of the asteroid and the establishment of an atmosphere and a terraformed surface, a pre-flight shakedown will likely be undertaken in our Solar System. Navigating the planets at low speeds for twelve months, the shakedown will allow hardware, software and crew to be checked before leaving the Solar System. Minor problems in any systems hardware could quickly be repaired and serve as hands on experience for the crew. This twelve month period will also allow crew members to become acquainted with one another and discuss social norms and mission guidelines.

## Boost and Coast Phase

Once vessel and crew have passed the shakedown portion of the mission, the antimatter engines will accelerate them to approximately thirty percent of the speed of light. Once this velocity is attained the engines will be shut down as the vessel enters the coast phase of the mission, traveling at sub-relativistic speeds for approximately three decades. After a decade of travel, one way communication time will exceed a year, limiting communication to the transfer of data on recent events or discoveries. As the vessel nears the Centauri system, one way message times will be roughly four years.

During this coast phase, the crew will have the freedom to define social norms, establish routines for government and education of the children, and refine the physical aspects of their environment. As the colony settles down to living normal lives on this microplanet early planning can begin for the eventual need to expand habitats as the population grows. In conjunction with the sociological tasks of the crew, an itinerary of specific research goals and programs will have been established by mission planners. It is assumed however, that the crew will be allowed the flexibility to develop schedules, pursue new lines of research, launch space probes and develop new mission goals and hardware as needed.

The forced autonomy of this vessel after leaving the Solar system makes freedom of schedule planning an absolute necessity. Cognitive conflicts between Earth based planners and the vessel would quickly develop, should any group based in our Solar system attempt to dictate behavior patterns to an interstellar crew. Cognitive conflicts are caused by differences in information and rules regarding situations having no single concrete solution. This will

This will be particularly true with interstellar travel where the information and experiences of the people living on this micro - planet will be vastly different from mission control back in near solar space. Indeed, it may be the case that the need for a 'mission control' as we now define it will no longer exist. Other than serving as a rational voice from a distance, there is little else that we here in the Solar System could do to influence or assist this crew once they are on their way.

Major activities of this phase of the mission will likely center around long term research efforts in astronomy, astrophysics, human sociology, particle physics, and constant analysis and refinement of the star vessel and systems. In addition to looking back at our Sun and planetary system, the research teams will also be looking ahead to probe the Centauri system to better define its structure and formalize a plan of action upon arrival. Much of this phase of the mission planning must be done by the crew itself, for they will have access to the best data as they close in on the system.

## Deceleration and Exploration Phase

As the interstellar vessel nears the appropriate distance from the Centauri star system, computers programmed by the crew will automatically begin to slow the vessel at a deceleration rate no greater than 1g. Based on years of observations during the cruise phase of the mission, an exploration plan will have been developed to best utilize the peculiarities of this three star system.

Arriving at this star system will be a crew in transition as the second generation prepares to takeover an increasing responsibility for the mission. The population of this microworld having at least doubled by arrival time, will be influencing social and leadership dynamics as well as the physical environment.

The exploration phase will afford a level of excitement for the crew after a long period of quiet research and social development. Travel within the Centauri system will allow the vessel to utilize solar energy to power various on-board systems, and perhaps even distribute natural light throughout the vessel during the daylight phase. Spaceport activity will likely increase as shuttles and experimental pods are orbited around the individual stars. If planetary bodies or asteroids are found, landers may well be deployed to explore their surfaces and collect samples.

The five to ten year stay within the Centauri system will allow the crew to draw on the natural resources there to replenish stocks of raw materials. The presence of solid bodies in the system would allow mining operations to provide elements facing exhaustion from the vessels asteroid or stores.

## CONCLUSION

Given the proper assumptions, interstellar flight appears feasible. At least it is no less feasible than any of the thousands of explorational sorties undertaken by mankind over the centuries. What is unique and somewhat astounding is the level of complexity and the vastness of the distances.

While most of us want interstellar flight to be successful, we realize that even with quantum technological advances, travel to the stars will never be easy. Risks will abound within technology and within man himself. Structural decay of the vessel, fire, epidemic diseases and collisions with uncharted members of interstellar space would put the mission at risk. [ref. 13] Equally threatening is the prospect of this isolated microsociety developing along a perverse social path.

The exploration efforts of these missions into areas previously uninhabited or visited by mankind is part of the continuing moral drama of discovery. Mankind places himself in a vulnerable position as he takes with him news of his own existence to places that are likely to be abiotic and insentient. Undeniably, part of the risk of this moral drama is the chance of mankind losing its anthropocentricity with the discovery of other civilizations amongst the stars. Should intelligent life be found - though it is inconceivable that it will be in the Centauri system - assimilation of that information into our cultures could have devastating effects on the status quo of institutions and the collective intelligence of mankind.

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N91-22160

# SPECULATIONS AND INQUIRIES REGARDING THE POSSIBILITIES FOR AND LIMITATIONS TO PRACTICAL INTERSTELLAR TRAVEL

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## ABSTRACT

The existence of superluminal phenomena have now been independently confirmed by physicists working in several different laboratories, most notably by the team of Alain Aspect in Paris. The two major variants of these experiments are described and their implications for superluminal communication and superluminal travel are discussed. It is noted that while the original suggestion for these experiments is due in part to Albert Einstein (Einstein, Rosen, and Podolsky, 1935), their recent empirical validation presents a significant anomaly within the theoretcal framework of the special theory of relativity, although they are predicted within the framework of quantum mechanics. How a newly emerging paradigm broadly encompassing the empirical sciences, and informed by both the social sciences and general systems theory may resolve this theoretical crisis is discussed. With the impasse to further elaboration of these effects for possible superluminal applications removed, the discussion concludes with a research proposal.

## INTRODUCTION

Until recently even the possibility of interstellar space travel has been limited by the result of the special theory of relativity due to Albert Einstein that the velocity of light "cannot be exceeded by any form of propulsion that relies on the expulsion of mass to obtain reactive thrust... morover, every scientific experiment, designed within the last half-century to test Einstein's hypotheses concerning relativity, has consistently added verification to his postulates" (ref.15).

In the brief course of this paper, I will attempt to state not only the observations which lead to the conclusion that practical interstellar travel ("practical" in the sense that travel times will be at least on the same order of magnitude as the multi-month peregrinations of the sailing ships of the Great Age of Exploraton) is at least now thinkable, but also the process by which such a "possibility" may proceed to "practicality." Scientist-science fiction author Arthur C. Clarke has made the observation that every great idea, invention, or discovery comes about through a three-step process as shown in Figure 1. Dubbed Clarke's Law, the experts, as illustrated above, have already amply supplied the first step.

**Figure 1. Clarke's Law**

**From the viewpoint of experts, any great idea, invention, or discovery comes about in a three-step characterization process:**

1. **"It's impossible!"..."It can't work!"... "It can't be!"**
2. **"It's impractical!"..."It won't work!"..."It'll never make you any money."**
3. **"I thought it was a great idea all along!"**

But these same experts evidently will not allow themselves to be cast in the roles of historical curmudgeons. British Interplanetary Society fellow James Strong as one such expert quoted above in nearly the same breath stated that "to be so positive that it was impossible- after a mere century of industrial progress- is surely defeatist, and most men would be more guarded in their statements. There is always room for speculation concerning the future, but none for evasion." (ref.13). To quote the admonition of Hamlet, "There are more things in heaven and earth, Horatio, than are dreamt of in your philosophy." Einstein himself said "Imagination is more important than knowledge."

Now, perhaps ironically, but more likely consistent with Einstein's latter observation, a ludicrous prediction made by quantum mechanics that Einstein elucidated in the 1935 paper with Rosen and Podolsky to demonstrate the incompleteness of quantum theory has turned out to be true.

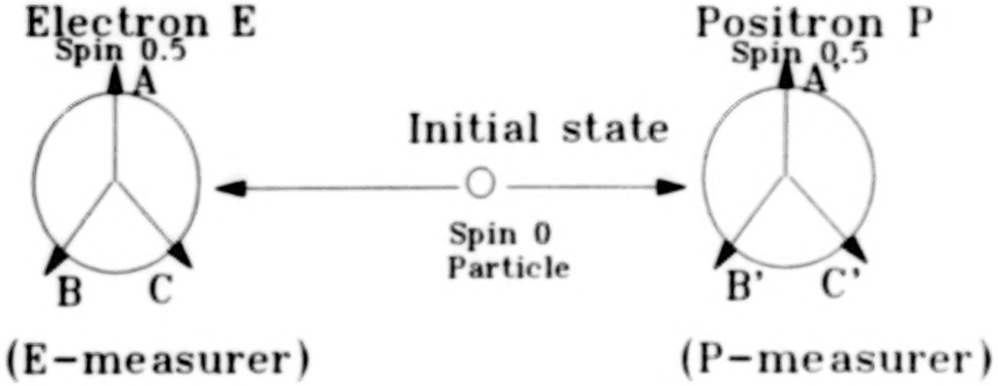
## THE EINSTEIN-ROSEN-PODOLSKY (ERP) EXPERIMENTS

The original challenge to quantum theory devised by ERP was a thought experiment that relied on the conservation of momentum of two interacting elementary particles to show that the position and monentum of one of the particles could be determined exactly by measuring the momentum and position of the other particle even if they had already separated by a large distance (ref. 5) This result is required by the putative conditions of the experiment, which was to be conducted in such a way as to avoid any interaction with other particles or systems. As momentum, like energy, could neither be created nor destroyed, the position and momentum of the particle "in London" could be instantaneously determined by measurement of these properties of the particle "in New York." But the Heisenberg uncertainty principle, which, like the conservation of momentum law, had already been experimentally verified, stated that the position and momentum of a single particle could not be ascertained without uncertainty. But most distressing was the result that the principle of local causality- that distant events cannot instantaneously influence local objects without mediation- was also violated! According to physicist Heinz Pagels writing about ERP 50 years later, "This finding startled most physicists, because they held the principle of local causality sacred" (ref.14).

As the conditions necessary to isolate the particles in the ERP thought experiment from other influences would prove to be difficult, physicists such as David Bohm (ref. 2) and J.S. Bell devised other, practical experiments that nonetheless had the property in common with the original thought experiment that a conservation principle would allow the state of one, remote part of the system to be determined "instantaneously" by the measurement of the state of another part of the system separated by some sensible distance from the first. (Note that it is important to bear in mind as we describe these experiments that they all are appied to the "microworld" of elementary particles such as fermions and photons). The two major types of experiments that have been proposed to date involve 1) decay of a spin-zero particle into two spin-one-half particles, viz., an electron and a positron (see ref. 14); 2) decay of "positronium," an atom consisting of a single electron bound to a positron (positive electron), into two photons that travel in opposite directions (see refs. 2,3,8,14). (In the discussion of all such experiments it is important to bear in mind that while we talk about individual particles, the observations are actually being made on macrosopic agglomerations of the source particles and the resulting decay products, and that the actual occurrences of decays happen in the "chaotic" or random manner typical of all radioactive decay processes).

Figure 2 illustrates ERP experiments of the first kind. At the

**Figure 2. An ERP experiment involving pair production. A spin-zero particle decays into an electron E and a positron P. Measurement of the spin of any one of the particles fixes the spin state of the other instantaneously.**



time of decay, both positrons and electrons fly off in opposite direction and spinning with their axes of spin oriented more-or-less randomly. Instruments can be set up in such a way in advance of any series of spontaneous decays as to determine the number of respective particles spinning in any of the directions A, B, C for the electron observations and A', B', C' for the positron. The directions A',B',C' are to be made parallel to A, B, C as shown and in the same plane. When the measurements are conducted, sometimes the electron-measurer will register YES whenever the spin is in the A, or B or C direction and NO when the spin is NOT in the A, B, or C direction. Similarly, the positron-measurer counts up his YES's and NO's for his A',B', C' settings.

Now quantum theory predicts, according to Penrose (ref. 14), that 1) whenever the A, B, or C measurement is YES, the corresponding A', B', or C' measurement is always NO, and vice versa, i.e, the results by the two measurers always disagree; 2) whenever the dials for the spin directions are spun and set at random and independently of one another, then the two measurers are equally likely to agree as disagree. Penrose goes on to prove logically that the results cannot be explained in terms of any set of conditions hidden from observation whereby the electron and positron spins are prepared in advance, as the conservation condition stated for this experiment (i.e, opposing spins) leads to a false prediction (at best a 5 to 4 agreement/disagreement ratio) when condition 2) is imposed. Hence there is no set of prepared answers which can produce the quantum mechanical probabilities.

Penrose states that the above experiments have not actually been performed, but that the second type using the polarization of pairs of photons has. Here the conservation principle states that the opposing photons must be plane-polarized in the same direction whenever they are measured. He quoted the work of Alain Aspect (1986) and his colleagues in Paris as having performed the "most accurate and convincing of the experimental results"( ref. 14). Aspect added the additional feature that the "decision" about which direction to measure the polarization in was only made after the photons were emitted. Thus, if we think of some influence traveling from one photon detector to the one on the opposite side, signalling the direction in which it intends to make the measurement so that the opposing photon can "align itself" in the same direction, then the effects must be able to travel faster than light!

However, all these researchers are quick to point out that there is no known way to actually set the direction of spin or polarization of the electron/positron or photon, respectively or to predict in advance how a particular particle or photon will be oriented- only that when A is "UP" then A' is "DOWN" or that when photon A is found to be polarized at 60 degrees, then photon A' must also be polarized at 60 degrees. Within the framework of current quantum mechanics, Penrose quotes Ghirardi, Rimini, and Webber 1980 as having made a general demonstration that such putative superluminal influences can't be used for signalling.

We have thus seemingly come round to where we started, with no superluminal communication, let alone superluminal transportation possible. Have we merely generated "a lot of sound and fury, signifying nothing..." as MacBeth lamented? I think not. In the next section I will state why the situation is still better than before the revlelation of the EPR experiments.

## DIRECTIONS FOR RESEARCH

Thomas S. Kuhn over 25 years ago wrote his now-famous "Structure of Scientific Revolutions" in which he concluded from his historical study of major scientific "revolutions" that when major anomalies occur while practitioners are working within a given scientific body of knowledge or "paradigm", "something's got to give". Either the

Kuhn's characterization of this progression.

**Figure 3. T.S. Kuhn's Structure of Scientific Revolutions**

**Stage 1. Normal science: puzzle-solving**

**Stage 2. Anomaly and emergence of scientific discoveries**

**Stage 3. Crisis and the emergence of scientific theories**

**Stage 4. Resolution of crisis and change of world-view**

From what we have described above, we are well into Stage 2, and we have evidence that we have already moved into stage 3 with respect to the quantum mechanical/relativistic paradigm. References 1, 2, 3, 6, 8, 10, 13, 14 are all major scholarly works that both grapple with the anomalies stated above and engage in major philosophical discussions of the history, personalities, motivations, and metaphorical content of the paradigm in question. In this brief space I can only mention these treatises, but I wish to bring out two major conclusions from these works that seem implicit but are not stated in any one place.

First, that the observations that lead to the anomalies, whether simply "thought" experiments or actual observations, are "real" and that therefore either quantum mechanics, special relativity, or both are fundamentally limited and must be corrected or replaced with a new paradigm that explains and/or predicts all existing data properly.

Second, that "a process of metaphor" is under way now that involves an intensive search for familiar objects, images, and concepts that can serve as the bases for a new model or set of models that will explain these phenomena. Psychologist Julian Jaynes can be credited with the realization that the "history of thought" and intellectual development is a process by which familiar phenomena (which he calls "metaphiers") are sifted through to give meaning to the unfamiliar–or anomalous– which he refers to as the metaphrands. (Thus a metaphor always is composed of two parts– the metaphier:"the familiar", and the metaphrand: "the unfamiliar") (ref. 7).

I hereby suggest that a conscious search for the appropriate "metaphiers" will be the speediest way to resolve the anomalies, and to arrive at either a new paradigm or a re-vamped version of the old. I am confident that such a devleopment will remove the impasse to further research into superluminal phenomena and allow the concommitant technology to develop. Finally, I list in Figure 4 an agenda for research:

**Figure 4. Proposed Areas of Research for Superluminal Communication and Transportation**

**1. Identify a quantum system whose decay phenomena can be externally influenced. Such a system could be used for superluminal communication.**

**2. Re-conduct the Michelson-Morley experiment at higher levels of sensitivity both on earth and in the space environment to determine the presence of "luminiferous" and even "super-luminiferous" media"**

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N91-22161

# EXPLORING THE NOTION OF SPACE COUPLING PROPULSION

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## ABSTRACT

All existing methods of space propulsion are based on expelling a reaction mass (propellant) to induce motion. Alternatively, "space coupling propulsion" refers to speculations about reacting with space-time itself to generate propulsive forces. Conceivably, the resulting increases in payload, range, and velocity would constitute a breakthrough in space propulsion.

Such speculations are still considered science fiction for a number of reasons: (1) It appears to violate conservation of momentum. (2) No reactive media appear to exist in space. And (3) No "Grand Unification Theories" exist to link gravity, an acceleration field, to other phenomena of nature such as electrodynamics.

This paper focuses on the rationale behind these objections. Various methods to either satisfy or explore these issues are presented along with secondary considerations. It is found that it may be useful to consider alternative conventions of science to further explore speculations of space coupling propulsion.

## INTRODUCTION

Space coupling propulsion is a term offered here to collectively discuss those concepts that consider interacting with the "fabric", or structure, of space-time itself to produce propulsive forces. These concepts have been grouped together because they raise similar issues and unknowns. Some of the more familiar concepts within this category include "anti-gravity", "graviton rockets", and "propellantless propulsion". "Anti-gravity" refers to the negation, control, or generation of gravitational forces, or more generally the concept where a vehicle can induce its own acceleration field (reference 1). "Graviton rocket" refers to the concept of expelling gravitons or gravitational waves to produce reaction forces. "Propellantless propulsion" is a more generic term that refers to any concept that does not need on-board reaction masses, and thus extends beyond the category of coupling to the more conventional concepts of light sails, magnetic sails, or electrodynamic tethers.

In much the same way that pioneering rocketry was inspired by the science fiction of its day (reference 2), these concepts and the awareness of their potential benefits were probably inspired by more recent science fiction. Science fiction images of vehicles that levitate off the Earth or that travel interstellar distances with ease suggest propulsion that does not require propellant, or more specifically, that can induce acceleration fields at will. Without the burden of carrying propellant, payload capacities could dramatically increase, ranges would no longer be limited by propellant supply, and there could even be higher vehicle velocities from continuous acceleration. In addition to such propulsion breakthroughs, the ability to produce acceleration fields could provide artificial gravity to enable crews to endure long voyages. Additional spin-offs could be speculated, but it is enough to say that a discovery on this frontier would constitute an enormous breakthrough.

Presently, such wishful thinking remains within the realm of science fiction for several reasons: Primarily, the notion of producing motion without a conventional reaction mass appears to violate conservation of momentum. In all known forms of propulsion, something acts as the reaction mass. Rockets expel propellant, and aircraft push against the air. Space coupling propulsion can conserve momentum by various means, but implying that some substance in space acts as the reaction mass evokes another objection: There is nothing in the vacuum of space to react against. Finally, speculations about creating a local acceleration field, similar to gravity, evokes another objection: There are no "Grand Unification Theories" (GUTs) linking the fundamental forces of nature to enable controlling gravity via intermediary phenomena, such as electrodynamics.

Because this subject is still speculation rather than engineering, space technologists are not pursuing it. Scientists are pursuing the underlying fundamentals, but that work is not targeted toward space propulsion applications. The main bodies of science that apply most closely to this subject are General Relativity, Cosmology, and Particle Physics. General Relativity deals with gravity and space-time (reference 3), including experiments aimed at detecting gravitational waves (reference 4). Cosmology, which deals with the origin of the universe and the structure of space-time, has combined with Particle Physics in pursuit of GUTs. That quest is largely based on exploring the correlations between the fundamental forces of nature at high energies, like those that existed during the "Big Bang" origin of the universe (reference 5). These approaches are making progress, but they are oriented toward general understanding rather than applications to space propulsion.

Not having any GUTs, however, should not preclude the exploration of space coupling propulsion. The lack of rigorous scientific and engineering theories should not discourage qualitative speculations about space coupling propulsion. By making "what-if" speculations (assuming that it is indeed possible to propel against space-time), while considering the science issues, various ways can be speculated for satisfying the issues, or at least identifying the unknowns within the sciences. Quantitative validation of these ideas or even a detailed identification of the unknowns are beyond the scope of a single paper. This paper is meant primarily to suggest the range of possibilities which could spur further discussion and investigation.

Motivated by the revolutionary benefits, inspired by the science fiction, and challenged by the speculative nature, this paper explores the notion of propelling a vehicle against the structure of space-time. By exploring this notion from a "what-if?" perspective, rather than "what-with?" (engineering tools/methods) we might stimulate thought-provoking explorations, might help shape the tools of science to be more applicable to the perspective of space propulsion, or might even reveal more readily obtainable solutions.

## CONSERVATION OF MOMENTUM

The primary reflexive response to the notion of space coupling propulsion is concern over conservation of momentum. Newtonian mechanics requires that momentum be conserved, and propulsion without propellant appears to violate this law. In the case of conventional propulsion, conservation is satisfied because the expelled propellants possess equal and opposite momentum to the vehicle. Space coupling propulsion appears to violate this law because the reaction mass is not readily apparent. Conservation of momentum can be satisfied in various ways that do not require having an on-board supply of reaction mass. These include: conservation by using the contents of space as the reaction mass, conservation by expelling non-mass momentum, conservation by negative mass, and conservation by coupling to distant masses via the intervening space. Several of these treatments, most notably interacting with the contents of space and coupling to distant masses, evoke secondary issues.

### Conservation Using the Contents of Space:

Rather than using an on-board reaction mass, momentum can be conserved by using the matter that is available in space in much the same way that aircraft propellers react against the medium of air. Space, however, is commonly thought to be empty which is another major barrier to the notion of space coupling propulsion. Space is not empty, however. Space contains interstellar matter, magnetic fields, star light, Cosmic Microwave Background radiation, and subtle substructures of space like Zero Point Energy and the virtual sea of pair creation/annihilation. And, underlying all of these media, is the "structure" of inertial frames which may also constitute a reactive medium.

The more familiar contents of space, matter, light, and magnetic fields, are probably too feeble to be an adequate reactive media. Methods have been proposed that use these media for propulsion, namely solar sails (references 1,6) and an "Interstellar Ramjet" (references 1,6,7), but these methods do not constitute genuine space coupling propulsion.

A less obvious candidate for a reactive media in space, which was discovered in 1964 (reference 3) and is being studied today by the COBE space craft (reference 8), is the Cosmic Microwave Background radiation. Presumed to be a remnant from the Big Bang, this background radiation permeates all space and appears to be coincident with the mean rest frame of the galaxies surrounding earth, and provides a phenomena by which velocities relative to that frame can be measured (directional doppler shifts). Such features invite using this background as a medium to possibly react against, but, like interstellar matter, it is very feeble (4x10^-34 g/cm^3) (Reference 3). Although not promising as a direct reactive medium, it may one-day provide a useful reference for deep space navigation.

A more fundamental category space coupling media is the substructures of space. These include Zero Point Energy (also known as the vacuum fluctuations of the electromagnetic field) and the sea of virtual pair creation/annihilation. Zero Point Energy is the absolute minimum energy of a harmonic oscillator at its ground state. This means that even in the vacuum of space, there is a non-zero energy of electromagnetic oscillations (reference 3). The sea of virtual pairs refers to the quantum mechanical possibility that particle pairs (matter-antimatter) are spontaneously produced and reconverged throughout space. Usually they are low energy photons, but could occasionally be electron/ positron pairs (reference 1). Some concepts for reacting against these medium have been speculated and may be candidates for Space Coupling Propulsion (references 1,9).

Perhaps the most likely media for genuine space coupling propulsion are inertial frames themselves. Inertial frames are the fundamental frameworks against which the laws of motion are described, and as such, have some physical significance beyond just mathematical entities. The nature of this physical significance and the correlation to other phenomena is not fully understood. Imagining inertial frames as a candidate reactive media is difficult because inertial frames are used as a reference for observing interactions, rather than as a participant of interactions. The utility of inertial frames will be discussed later in this paper under the heading; "Conservation Using Coupling to Distant Masses".

Perhaps one way to consider interacting with inertial frames is to use the previously described contents of space, in particular, the Cosmic Microwave Background radiation and Zero Point Energy. Both of these phenomena are coincident with inertial frames, and perhaps are fundamentally linked to some "structural" property that may some-day provide an indirect means to reactively couple to inertial frames themselves.

**Conservation Using Non-Mass Momentum:**

Another way to satisfy conservation of momentum is to consider that some non-mass momentum is expelled from the vehicle, such as photons, gravitons, or hypothetical "space waves" (figure 1). Assuming that it were possible to focus all this expelled radiation along a single direction, the general equation relating power (P), force (F), and velocity (v), $P = F \times v$ could be used to indicate the potential force per radiated power. The velocity term refers to the radiation velocity, and in the case of photons and gravitons, this is the speed of light. Entering the speed of light into this equation translates to a rather feeble force per radiated power: 3.3x10^-9 Newtons/Watt.

This force/power equation assumes, however, that the radiation has zero rest mass. Gravitons have been speculated as being a more promising candidate of energy expellant because they might not have zero rest mass and because they are related to mass/acceleration phenomena. Gravitons are quantized gravitational waves analogous to the way that photons are quantized electromagnetic waves. Unfortunately, gravitons are still just theoretical entities, and no methods have yet been proposed for using gravitons for propulsion.

Another avenue for exploring this non-mass momentum theme would be to look for alternative forms of "space waves" that either have a non-zero rest mass, or have much lower propagation velocities than light. Perhaps oscillations in the "structure" of inertial frames may constitute these hypothetical space waves.

(NOTE: The graphic device employed in figures 1-3 is space-time "fabric", where the height of the "fabric" is proportional to gravitational potential. Gradients or "hills" in this graphic fabric represent acceleration fields and, analogous to real hills, induce motion in the "down hill" direction.)

$$E/c = (mv)_{vehicle}$$

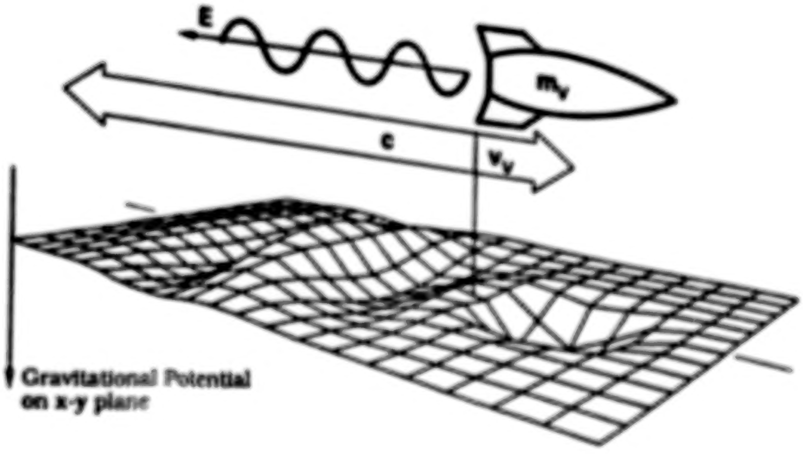
E
$m_v$
c
$v_v$
Gravitational Potential on x-y plane

Fig. 1. Conservation of momentum by expelling non-mass momentum such as hypothetical "space waves".

$$(-m)(-v) = (+m)(+v)$$

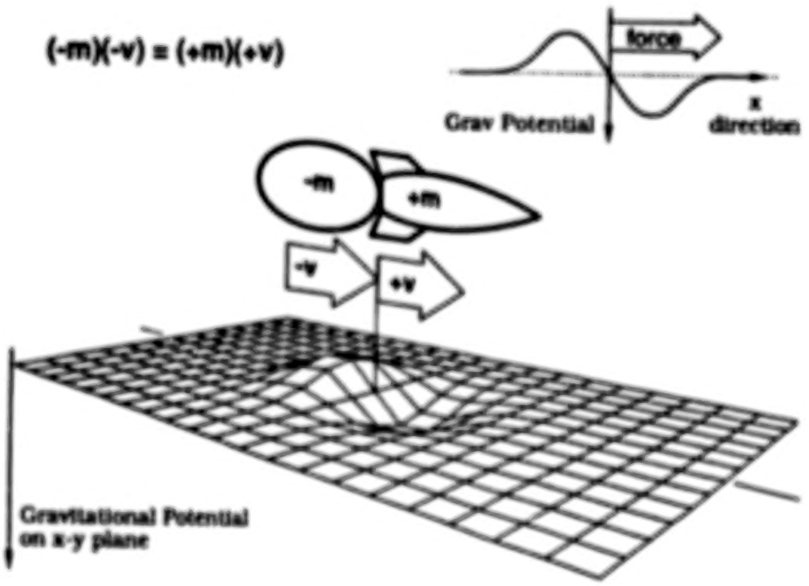
force
Grav Potential
x direction
-m
+m
-v
+v
Gravitational Potential on x-y plane

Fig. 2. Conservation of momentum by using negative mass.

**Conservation Using Negative Mass:**

An imaginative means of conserving momentum is to create a condition where the total mass, and hence momentum, is always zero. This treatment uses hypothetical "negative mass". If equal amounts of positive and negative mass were placed side by side, they would both accelerate along the vector pointing from negative toward positive matter because of the interactive properties of negative and positive mass (figure 2). This negative matter propulsion concept does indeed satisfy the laws of motion (reference 10). The weakness of this negative matter scheme, aside from the problem of obtaining and handling negative matter, is whether or not the laws of motion would still be satisfied if unequal proportions of negative and normal mass were used. Conservation may still hold with unequal masses if the concept of coupling to distant masses is considered. This concept is discussed next.

**Conservation Using Coupling To Distant Masses:**

Perhaps the most fundamental and broad-sweeping concept for space coupling propulsion is the concept where a vehicle produces its own acceleration field to push against some "structure" of space. To satisfy conservation of momentum with this concept, it is necessary to speculate that the reaction force is imparted onto distant masses via this space structure in much the same way that gravity attracts distant masses. Momentum is conserved by the equal and opposite momentum imparted to the space/matter system (Figure 3). This requires the perspective that matter is somehow connected to space, and that space has a degree of "stiffness" to transmit force to distant matter. This perspective is difficult to conceptualize, and evokes secondary issues that are discussed next.

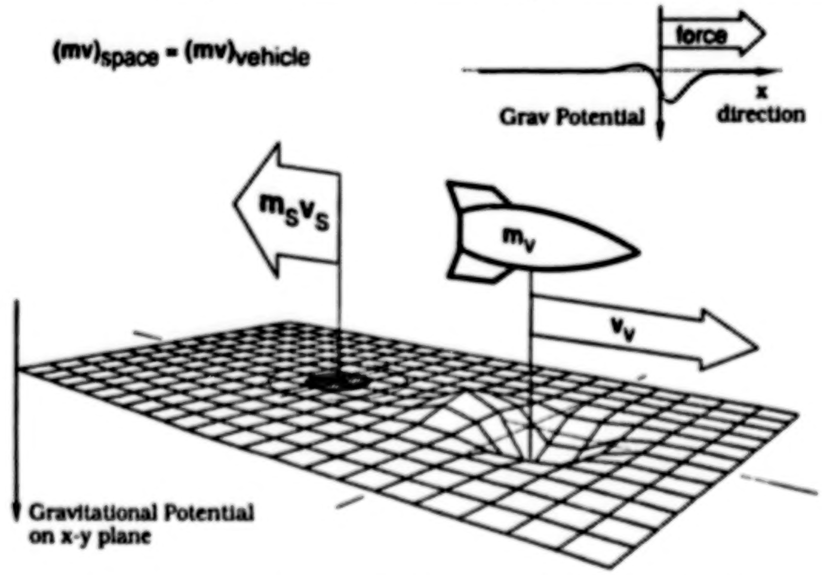
$(mv)_{space} = (mv)_{vehicle}$
force
Grav Potential
x direction
$m_S v_S$
$m_v$
$v_v$
Gravitational Potential on x-y plane

Fig. 3. Conservation of momentum by coupling to distant masses via coupling to the structure of space.

Mass is known to "connect" to space in two ways; gravity and Mach's Principle. Gravity is the field phenomena related to the presence of mass, where this field causes an attraction with other distant masses. Mach's Principle relates the presence of matter to the definition of inertial frames. Mach theorized that inertial frames exist only because of the presence of matter (reference 3). Additionally, as indicated by inertial drag, a given point in space may actually be a composite of inertial frames, each of which is somehow "connected" to its source mass. Another interesting point is that gravity and inertial frames are related. Gravitational fields are, in essence, accelerated inertial frames, and alternatively, unaccelerated inertial frames are gravitationally flat (the gravitational potential across the space is constant).

To further consider reacting against distant masses, it is useful to indulge in some alternative perspectives of these known phenomena. For example, it is useful to consider that inertial frames and their "connection" to their source masses provide *the* structures for reactive coupling. This implies that inertial frames would have a property for referencing position, in addition to referencing acceleration, to allow position relative to the frame's source mass to be uniquely defined. This is unconventional because inertial frames are thought to provide only a reference for measuring accelerations, not velocity or position. Additionally, it is useful to assume that this position property has some characteristic "stiffness" that allows forces to be imparted across space to the source masses. Such considerations evoke the notions of the proverbial "aether" and the theoretically defunct "absolute reference frame". The similarity between these views and those unpopular notions is not exact, and hence, should not prejudice indulgence in these perspectives.

Continuing with these speculative perspectives, forces could be induced relative to inertial frames if it were possible for a vehicle to alter its gravitational field distribution or its connectivity to its own inertial frame. By redistributing its own gravitational field, it could, in effect, create a local asymmetric acceleration field. The reaction forces would be imparted to the "stiff" inertial frames and subsequently to their source matter (figure 3). This is similar to the special case in the concept of negative mass propulsion where there is more normal mass than negative mass. In this case the non-zero momentum of the vehicle would be balanced by the equal and opposite momentum of the inertial space and its associated matter.

An issue related to these speculations, whose investigation may provide clues to the "structure" of inertial frames, is the proportionality of the imparted forces. If inertial spaces are pushed against, do the frames' source matter move in unison (Case A, figure 4), or do they move proportionally (Case B, figure 4)? What is this proportionality based on; the distance from that point and/or the magnitude of the source mass? One speculation to quantitatively explore this proportionality is to assume that the proportionality coefficient at a given point in space is simply the gravitational potential of the source mass at that point in space. These speculations and questions have yet to be fully explored.

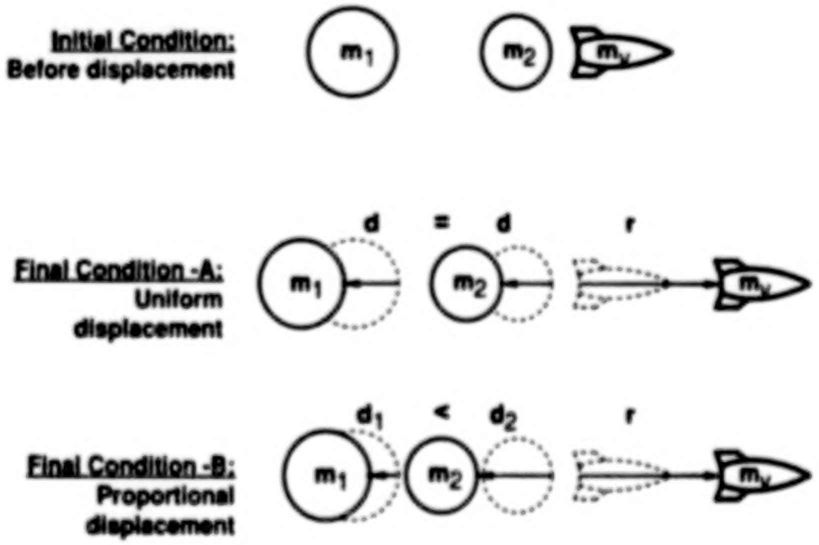

Fig. 4. Questioning the proportionality of coupling to distant masses.

The concept of coupling to distant masses requires some unconventional perspectives on the structure of space, particularly with respect to the definition of inertial frames and their relation to matter. Further investigations of this coupling possibility would likely require further indulgence and refinement of these unconventional perspectives, including exploration of the proportionality issue and the relation to Mach's Principle.

## SEARCHING FOR A FORCE INDUCING TOOL PHENOMENA

Having addressed the issues of conserving momentum and the contents of space, and having identified the desirability of inducing a localized acceleration field onto an inertial frame, the next issue is to identify candidate mechanisms to create this acceleration effect. Acceleration fields imply gravity, and hence, the target mechanism is to discover some means to alter gravity. This evokes the last major reflexive response to the notion of space coupling propulsion: There are no known ways to practically manipulate the phenomena of gravity. There are two avenues to respond to this issue. The first avenue is direct manipulation of gravity by the motion of masses. The second avenue is to induce gravitational forces via an intermediary phenomena, such as electrodynamics, which evokes the need for a "Grand Unification Theory". Although science has not yet provided such a theoretical mechanism, there are several different approaches toward discovering a useful connection between gravity and other phenomena. All these approaches offer different applicability or viability for space coupling propulsion and are described next.

### Inducing Accelerations by Motion of Masses:

Several concepts exist that consider inducing force or local accelerations by the motion of nearby masses. In general, these concepts are either impractical because of the enormous mass densities and speeds required, or are of doubtful viability because of uncertain physics.

1. **General Relativity Based Gravity Devices:** An impractical, but theoretically sound method to create acceleration forces is based on "magnetic gravity". General relativity provides the possibility of an analogous phenomena to gravity that magnetism is to electricity. Unfortunately, in order to produce appreciable forces with these conceptual devices, ultra-dense masses (densities approximately that of a white dwarf) must be moved at relativistic speeds along strictly defined paths (reference 1).

2. **Gyroscopic Antigravity Machines:** On a more speculative side, devices have been designed and patented (reference 11), that claim to produce gravity negating forces by gyroscopic motion. These devices are variations on a theme of converting angular momentum into linear force; a scheme which violates conservation of linear momentum. One example of this is a "Laithwaite Engine" (reference 12) which gives the appearance of providing upward forces by the upward swing of its gyroscopes once the device begins to rotate. This motion, however, is not a propulsive force, but rather a torque that makes the gyros change orientation in order to conserve angular momentum. This device and others like it do not hold much promise as propulsion devices, but are excellent instructional tools for understanding conservation of angular momentum.

3. **Anomalous Gyroscopic Measurements:** Recently, another gyroscopic device has been reported to produce reductions in weight proportional to rotational motion (reference 13). This report is not a proposed anti-gravity device, but rather an observation of an unexplained result. A gyroscope weighing on the order of 150 grams and with a vertical spin axis, was found to have weight reductions on the order of milligrams when rotated in the right-hand direction, and no weight change when rotated in the left hand direction. This is probably just an experiential error, but being such a peculiar observation, it is worthy of note.

### Inducing Acceleration Effects via Intermediary Phenomena:

In addition to the perspective of inducing forces from the simple motion of matter, there is the perspective of using some intermediary phenomena to induce effects. This means finding some controllable phenomena that is related with the phenomena of gravity, and using this control phenomena to indirectly induce gravitational effects. An example of this intermediary principle is the way that microwaves (electrodynamics) are used to induce molecular vibrations (heat). With respect to space coupling propulsion, the prime intermediary phenomena is electrodynamics. Various approaches to correlate gravity to other phenomena are briefly reviewed below and include: (1) General Relativity's connection between inertial frames and gravity as referenced by electrodynamics, (2) Gravity as an index of refraction for electrodynamics, (3) Gravity as a Zero Point Energy effect, and (4) The hypercharge force.

1. **General Relativity, Conventional Correlations:** Although gravity is known to effect electrodynamics (gravitational fields bend the path of light), General Relativity has not provided a gravity/electrodynamic tool applicable for space coupling propulsion. Instead, General Relativity uses electrodynamics (specifically the speed of light) as the reference for describing how gravity relates to inertial frames. For example, in the basic equation governing the relation between distance (d), time (t), and the phenomena of light, $d = t \times c$, the speed of light (c) is the reference constant, and space and time are the variables that "warp" relative to gravity (reference 3). Although this perspective has proven its usefulness, it may not be optimum for the perspective of space coupling propulsion.

2. **Index of Refraction and Gravity:** An alternative approach to describe the same natural observations is to treat the speed of light as the variable that gets "warped" in the presence of gravity. Basically, this perspective takes the form of relating the index of refraction of light to gravitational potential (references 14, 15). In the case of space coupling propulsion, it may be more useful to consider distance as "stiff" and the speed of light as the variable with respect to gravity. This approach allows considering electrodynamics as the intermediary mechanism rather than as the reference. To date, no proposed mechanism based on such perspectives have been reported, but this may be an interesting avenue for further exploration.

3. **Gravity and Zero Point Energy:** An interesting alternative approach to relating gravity and electrodynamics is the theory that gravity is an induced effect associated with Zero Point Energy fluctuations of space (reference 16). Various methods that use Zero Point Energy for propulsion have been proposed (references 1, 9), but no concept has been proposed that takes advantage of these correlations to induce asymmetric gravity fields. This approach also merits additional consideration.

4. **Fifth Force, Hypercharge Force:** Another interesting perspective linking gravity to some other more manageable phenomena, is the "hypercharge force" concept. In a reanalysis of the experiment that demonstrated that all masses, independent of composition, accelerate uniformly in a gravitational field, it was found that there may be a correlation between gravitational acceleration and a sub-atomic characteristic called hypercharge or baryon number (reference 17). This correlation has yet to be fully proven or disproven, but either way, it does not hold much promise as a candidate mechanism for space coupling propulsion. The differences in gravitational attraction by hypercharge are negligible (delta-g/g approximately 10^-7) (reference 17).

## CONCLUDING SUMMARY

"Space coupling propulsion" refers to the category of propulsion concepts that involve some means of coupling to the structure of space-time itself to produce propulsive forces. Such speculations are enticing because of the enormous benefits that could result. Unfortunately, such concepts are also considered science fiction. Even though these notions are still fiction, avenues for advancement exist. This paper examined the reasons behind the "science fiction" conclusion, and, based on the unknowns within those reasons, identified a variety of avenues for making progress on this potentially breakthrough subject.

The primary reasons that space coupling propulsion is considered science fiction are: (1) It appears to violate conservation of momentum, (2) There appears to be nothing in space to act as a reactive medium, and (3) There are no "Grand Unification Theories" which link the phenomena of gravity, an acceleration field, to manageable phenomena of nature, such as electrodynamics.

Conservation of momentum can be satisfied by using the media in space as the reaction mass, expelling non-mass momentum, using negative mass, or by coupling to distant masses via the structure of inertial frames. None of these methods are readily available, but the most promising pursuit may be to fundamentally explore coupling to distant masses via the structure of inertial frames, perhaps by some interaction with Zero Point Energy or the Cosmic Microwave Background radiation.

The contents of space that are candidates for reactive media include: interstellar matter, magnetic fields, starlight, Cosmic Microwave Background radiation, and the substructures of space such as Zero Point Energy, virtual pair creation/annihilation, or inertial frames themselves. None of these contents appear to be substantial enough to constitute an adequate reactive medium, but may be useful tools in the search for more fundamental structures of space. Perhaps the most promising direct medium for space coupling propulsion is Zero Point Energy, and the most promising indirect candidates are to use Zero Point Energy or the Cosmic Microwave Background radiation as intermediary phenomena to explore the structures of inertial frames.

With respect to searching for theories that link the phenomena of gravity to some intermediary phenomena, the possible avenues include: conventional General Relativity, exploring the notion of the speed of light as a variable relative to gravity, exploring the notion of gravity as a Zero Point Energy force, and exploring the use of the hypercharge force. None of these avenues presently provide a mechanism to alter gravity by practical means, but all are worthy of further investigation. Promising avenues could be the notion of the speed of light as a variable relative to gravitational potential, and the notion of gravity as a Zero Point Energy effect.

Although no methods yet exist to enable genuine space coupling propulsion, there are many unknowns and unexplored avenues that may one day lead to a breakthrough discovery on this frontier. These avenues are not always obvious nor do they promise high chances of success, but the potential benefits are enormous. So long as speculations can be offered, the opportunity to translate them into testable concepts exists, and within such activities may spring new awareness and closer avenues toward discovering the breakthrough potential of space coupling propulsion.

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VOL. 23, NO. 1, JAN.-FEB. 1986 J. SPACECRAFT

# Tethers and Asteroids for Artificial Gravity Assist in the Solar System

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**Planetary missions have benefited greatly from the gravity assist mechanism where a planetary flyby can boost or otherwise modify a spacecraft trajectory to accomplish specific goals. The multiplanet encounters of Voyager 2, for example, were accomplished using this process. The Galileo mission will utilize over a dozen flybys of the Galilean moons to perform a complete scientific investigation of the Jupiter system. Can asteroids be used in a manner similar to gravity assist? Their gravitational pull is too weak to provide the required bending of the trajectory, but this turning can be done by means of a tether. For example, the spacecraft may release a 100-km tether that will attach itself to an asteroid it approaches. The spacecraft then will be forced to turn in a long arc, which can be terminated upon release of the tether when the proper vector is obtained. The primary limitation of using this process will be tether strength which, with today's technology, will not allow relative velocities to exceed 1-3 km/s. This and other limits are investigated, as well as some mission possibilities using this method. Methods of tether/asteroid attachment and release will be discussed as well.**

## Introduction

"GRAVITY assist" is the term given to the effective use of the gravitational field of a massive body to deliberately modify the trajectory of a flyby spacecraft. For example, the two Voyagers[1] utilized the gravity assist of Jupiter to give them additional energy to continue their flight to Saturn. Jupiter not only provided a needed velocity boost so that it could reach the orbit of Saturn, but also turned the trajectories through just the angles needed so that the spacecraft would intercept the planet. Voyager 1 is headed out of the solar system, but Voyager 2, with a gravity assist by Saturn, will encounter Uranus in 1986, and then, again via gravity assist, will encounter Neptune in 1989. Similar gravity assist missions were Mariner 10, which flew by Venus before going to Mercury,[2] and Pioneer 11, which used a Jupiter gravity assist to lob it out of the ecliptic plane and across the solar system to Saturn.[3]

It is the ability of a massive body to bend a spacecraft trajectory in a near-collision approach that is essential to the gravity assist process. Jupiter is very massive, and will bend the spacecraft trajectory through a large angle of the order of 180 deg. In contrast, the asteroids have such low surface gravity that flybys of them are nearly rectilinear. If, however, in the course of a flyby, a spacecraft can be attached to an asteroid with a tether, then the spacecraft can swing around the asteroid through a large angle to accomplish the same type of trajectory change as gravity assist from a massive planet. Here, more about benefits than about means will be discussed, hoping to stimulate the process leading to the utilization of asteroids in a mode similar to gravity assist.

## Dynamics and Limitations of Gravity Assist

A gravity assist is kinematically equivalent to an elastic collision of two bodies, which produces a momentum exchange between them. It is an example of a "soft" collision, as compared to a hard collision involving actual surface impact. In the case of Voyager 2, in a gravitational encounter with Jupiter, an observer on Jupiter will see the spacecraft travel a hyperbolic path, and the spacecraft's outgoing speed will equal its incoming speed. However, a momentum increase (or decrease) will be seen by a heliocentric observer. Thus in a direct flyby of Jupiter, Voyager 2 experienced a velocity increase of several kilometers per second, permitting it to fly out to Saturn. On the other hand, a retrograde flyby of Jupiter will be needed for the Starprobe spacecraft to lower its sun relative velocity and cause it to fall to within 4 solar radii of the sun.[4]

Although gravity assist by the planets and the large planetary moons (such as the moons of Jupiter for the Galileo mission)[5] is a useful technique for expanding our capability to explore the solar system, the assisting planet or moon must be at the right place at the right time. Therefore, launch opportunities are restricted; favorable dates may be years apart. In an extreme case, the Voyager 2-type mission to the four giant planets will not be available again for 175 years.[6]

Knowing the mechanism, value, and limitations of planetary gravity assist, is there an alternate means of producing the same effect with the smaller but more numerous asteroids or comets? At the current level of space operations the answer is no, but with the development of tethers, which is now in the infant stage, it may be possible in the future. Since tethers are so new in space applications, some examples that are being seriously considered will be given. The realization that some applications have already been assigned Shuttle flight target dates may remove a somewhat science fiction aura that has surrounded the tether concepts.

Considering that there are thousands of asteroids greater than 1 km in diameter, the opportunities for utilizing gravity assist through soft collisions will expand by orders of magnitude. More asteroids will be discovered and smaller ones will be even more numerous. Those as small as 10 m in diameter will weigh over 1000 metric tons and could also be effectively used.

## Some Proposed Tether Applications

In 1988, it is planned to conduct some Shuttle-based tether experiments from orbit at a 200-km altitude in space.[7] In the first experiment, a 200-kg satellite made by an Italian team will be deployed 30 km above the Orbiter, connected to the Shuttle by an electrodynamic tether. Measurements will be made of the electric power generated as the tether moves through the geomagnetic field, and reciprocally of the thrust developed as a current is passed through the tether.

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Presented as Paper 84-2056 at the AIAA Astrodynamics Conference, Seattle, WA, Aug. 21-22, 1984; submitted Sept. 20, 1984; revision received June 12, 1985. Copyright © American Institute of Aeronautics and Astronautics, Inc., 1985. All rights reserved.

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On a second Shuttle launch, the same Italian spacecraft with different instruments will be suspended on a tether 100 km below the Orbiter to measure the atmospheric properties in a region where the Orbiter itself cannot fly. This experiment will analyze the aerodynamic and aerothermal interactions in a density regime where there will be some departure from free molecule flow. This tether has a circular cross section 2 mm in diameter with an inner core of Kevlar 49 and an overcoating of Teflon to protect the Kevlar from interaction with atomic oxygen and from solar uv exposure. In both flights, the tether dynamics will be studied and controlled as the tether and satellite are deployed into their gravity-gradient-stabilized position, and again as the satellite is reeled back into the Orbiter cargo bay.

In another proposed application,[8] tethers are used to assist in the transfer of payloads from low-Earth orbit (LEO) to geosychronous orbit (GEO). A Shuttle payload in LEO is deployed upward on a long tether. By taking advantage of the deployment dynamics, one can arrange that at the minimum altitude, the payload, swinging in an arc about the Shuttle payload center of mass, is moving so that the swinging velocity is in the direction of the orbital motion. At that point the payload is released. It moves in a new orbit with a higher perigee than the Shuttle, and a much higher apogee due to the velocity from both the swinging motion and the Shuttle angular velocity. At apogee it is caught by a tether lowered from a station in circular orbit. Since the payload usually will have a lowered velocity than the station, it will revolve in a circle about the station while constrained by the tether. The two masses can remain in this rotating configuration until the payload is released at its highest point to attain a yet higher orbit. Another station at a still higher altitude can repeat the catch-and-release process so that the payload eventually reaches GEO. We have computed that two or three stations would be needed if the tethers are to be made of an existing material such as Kevlar.

Upon closer examination, it can be seen that this method of momentum transfer, where the station loses the momentum that the payload gains, is a soft collision similar to a gravity assist. In this case, the station takes the place of the planet.

### Tether Strength Requirements

Assuming, at present, that some means have been developed for attaching a tether to an asteroid during a flyby, it is possible to determine the tether strength requirements as a simple function of the relative velocity ($V_r$) and the payload-to-tether-mass ratio. In these calculations, the asteroid is considered as an anchor point only, and its gravity gradient effect on the tension in the tether is neglected.

The spacecraft, of mass $M$ and velocity of approach $V_r$, is swung in a circular arc of length $L$ about the asteroid. At the spacecraft end, the tension is $T_L = M\omega^2 L$, where $\omega = V_r/L$ is the angular velocity of revolution. This represents a boundary condition on the system. If the tether has a constant cross-sectional area $A$ and mass density $\rho$, the differential equation for the tension $T$ as a function of radius $r$ is

$$\frac{\mathrm{d}T}{\mathrm{d}r} = -A\rho\omega^2 r \tag{1}$$

and the stress in the tether is $S = T/A$. The solution to this equation with the boundary condition above is

$$SA = T = M\omega^2 L + \frac{\rho A\omega^2}{2}(L^2 - r^2) \tag{2}$$

We will choose the cross-sectional area $A$ to make the stress at the origin (where the stress is greatest) to be the safe working stress $S_0$ of the tether material. We can then obtain the tether mass $m = \rho AL$. After some algebraic manipulation, we can find the spacecraft-to-tether mass ratio as

$$\frac{M}{m} = \left(\frac{V_c}{V_r}\right)^2 - \frac{1}{2} \tag{3a}$$

$$= \delta\left(\frac{C_L}{V_r}\right)^2 - \frac{1}{2} \tag{3b}$$

where the characteristic velocity is calculated as $V_c = \sqrt{S_0/\rho}$. In the second equality, the stress $S_0$ has been replaced by $S_0 = \delta E$, where $E$ is Young's modulus and $\delta$ is the safe working strain, and the fact that the longitudinal sound velocity is $C_L = \sqrt{E/\rho}$ has been used.

This equation shows that as the spacecraft mass approaches zero, there is an upper limit to the velocity that can be constrained by the tether, namely, $V_{\max} = \sqrt{2\delta}C_L$. This result is remarkable in that this limit does not depend upon the spacecraft mass or tether length. It is an intrinsic property of the tether material. For Kevlar 49, $C_L = 10$ km/s, and a good value for the working strain is $\delta = 0.01$. (Actually, the breaking strain is $\delta_B = 0.02$, therefore, we have an adequate, but not generous, safety factor of 2 in the working strain.) Then, the characteristic velocity is $V_c = \sqrt{\delta}C_L = 1$ km/s, and the maximum spacecraft velocity is $V_{\max} = 1.4$ km/s.

Equation (3b) places limitations on the achievable relative velocities for a given material and a given spacecraft-to-tether mass ratio M/m. A plot of this relation for Kevlar and stronger materials is given in Fig. 1.

This velocity limitation may be circumvented in two ways. First, from Eq. (1), since the tension decreases as the distance from the center of rotation increases, it is possible to decrease the cross-sectional area accordingly. The solution is an exponentially tapered tether (see Ref. 9, for example). Unfortunately, the tether mass required increases rapidly with velocities larger than $V_c$. For example, for $V_0 = 2V_c$, the tether-to-spacecraft mass ratio is 17.7. Thus, flyby velocities should be restricted to the characteristic velocity or less, except for situations where the tether is reused extensively.

A higher velocity may be achieved, however, through control of tether tension by paying out or reeling in the tether. It has been assumed that the spacecraft-asteroid tether attachment will occur when the velocity vector is exactly perpendicular to the radius vector between the two. Normally, however, there will be a radial component of velocity that the tether system must handle. If this component is outward, then the tether must be payed out to avoid the tether tension exceeding some maximum. If the component is inward, then the tether should be reeled in to ensure rotation of the spacecraft. Higher velocity than the limit may be handled by paying out the tether when the maximum tension would otherwise be exceeded. This cannot go on indefinitely, therefore, at some point the tether must be detached. Higher velocities, then, will limit the turning angle available for artificial gravity assist.

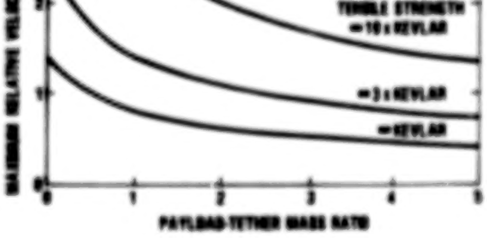

**Fig. 1 Velocity limitations for various payload-tether mass ratios and tether strength.**

## Methods of Tether Attachment to Asteroids

The material strength of the asteroid surfaces will not be known in detail when a tether assist mission is designed, therefore, the methods by which tethers are attached should be largely independent of surface characteristics. For small bodies, say 10 m in diameter, the entire asteroid could be surrounded by a fishnet-type structure made of flat Kevlar tape which when drawn tight does not place a concentrated load on any surface portion. As a single numerical example for orientation purposes, consider a Kevlar net engaging the 10-m-diam asteroid with tapes 0.01-g/cm$^2$ thick that cover 3% of the asteroid surface. This net will have a mass of 1 kg only, yet will be able to sustain a force of $10^5$ N applied to its drawstring. A spacecraft of 1000 kg traveling at 1 km/s past the asteroid, and held by a 100-km tether will generate a centrifugal force of only $10^4$ N, which could be held safely by the net.

For large bodies with strong surface structures, the tether end could be fastened to a ground penetrator that would anchor into the asteroid. The penetrator would be left behind when the tether is released to let the spacecraft fly off.

For large bodies of about 1 km in diameter with a highly brecciated surface that would have little mechanical strength, there may be nothing worthwhile for the tether to hold onto, and the fishnet required to englobe the asteroid would be too massive. It is suggested that the tether end have a plow-shaped device which the spaceraft drags along the asteroid surface. The plow exerts a force on the tether, due not to the strength of the surface it breaks up, but to the inertia of the material it displaces. Preliminary calculations show that appropriate tensions can be sustained with plow masses substantially smaller than the spacecraft mass. However, some active control system is necessary to cushion the shock if the plow hits a strong surface feature.

### Mission Capabilities for Artificial Gravity Assist

The velocity limitations just derived place some restrictions on the general use of artificial gravity assist. For example, none of the Jupiter flyby missions mentioned earlier could have been accomplished by this alternate method, since the relative velocity ($V$ infinity) exceeded 6 km/s in all cases. Because of this velocity limitation, each application must be examined carefully. For example, in an asteroid belt tour it is quite likely that a series of hops could be made with less than 1 km/s velocity difference in each. Furthermore, the tether method might well be aided by some rocket propulsion to reduce the velocity difference, since in any event propulsion would be necessary to achieve a close enough approach to use a tether.

Given the velocity limitations for soft collisions imposed by tether strength, it is possible to compute the orbit change available using this technique. Assuming a circular orbit for the asteroid (eccentric orbits with the same major axis give similar results), a soft collision with it, using a tether, will allow departure in any direction from the asteroid. The most favorable departure direction, to enlarge the orbit, is in the direction of the asteroid's orbital motion about the sun. The aphelion increase in terms of the radius of the initial orbit and the relative velocity is given in Fig. 2.

For an asteroid in the Earth's orbit, for example, having an orbit radius of 1 a.u., about 3 km/s are required to reach out to Mars orbit which is at about 1.5 a.u. A velocity of 1 km/s will only extend about 0.15 a.u. from the Earth's circular orbit.

Applying Fig. 2 to larger orbits, this technique is more effective. A tethered swing around an asteroid in Mars' orbit will extend the spacecraft aphelion out to 2.5 a.u., assuming a relative velocity of 3 km/s. This is still not as effective as a gravitational assist by Mars itself. At one phase in the Galileo mission design, for example, a close flyby of Mars was proposed to boost the spacecraft out to Jupiter, which is at 5.2 a.u.[10]

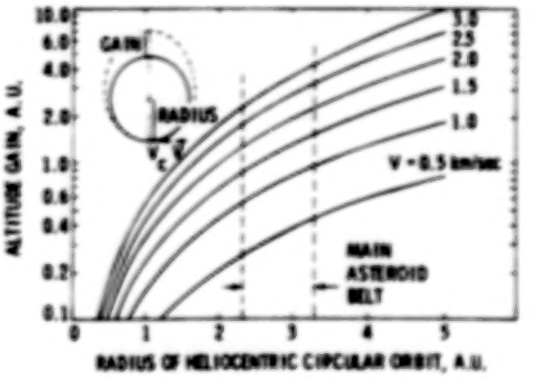

Fig. 2 Aphelion altitude gain for velocity applied to circular orbits.

The effectiveness of tether-asteroid assist improves considerably at the main asteroid belt and beyond. At Jupiter's orbit, for example, a tethered spin around with a relative velocity of 2 km/s will extend a spacecraft orbit to 10 a.u., or out to Saturn. Near Jupiter thre are many small bodies, which are possibly available for a tethered artificial gravity assist. Attending the planet itself, in addition to the four large Galilean moons, there are eight other smaller satellites orbiting at large distances. Their sizes are estimated to range from 1 to 50 km in diameter. Furthermore, as a result of Jupiter's influence, over a dozen known asteroids cluster about the two Trojan points located in Jupiter's orbit 60 deg ahead and behind Jupiter itself. Very likely there are many much smaller undiscovered trojans oscillating about these two stable points. In the future, when missions are flown to these asteroid groups, perhaps the soft collision techniques presented here may be of some benefit.

## Mission Applications

No specific missions using this technique have been calculated in detail, but some mission applications will be described to illustrate its potential. Some applications show how this tether assist method may ease the propulsion requirements of previously considered missions, and some show how the method may open up new mission possibilities.

### Mars Missions

A Hohmann transfer from Earth orbit to Mars orbit requires velocities of about 3-4 km/s. Using an intermediate tether-asteroid assist between Earth and Mars can reduce the propulsion requirements by about 50 %. The relative velocity requirements at the asteroid would be about 1.3 km/s, requiring a spacecraft-to-tether mass ratio of 0.1 for Kevlar (Fig. 1) or 1.2 for a tether three times stronger. An Aten-type asteroid, which has an aphelion less than that of Mars, and whose orbit is nearly in the ecliptic, would be a suitable intermediary. It is estimated that about 2500 asteroids with diameters greater than 1 km are in suitable near-Earth orbits. Probably, there are enough of these bodies so that the Earth-asteroid-Mars phasing problem would disappear, since at any launch time one of them would be in a proper position to accommodate a tethered assist.

### Outer Planet Missions

Transfer velocities from Earth to Jupiter orbits and beyond are much greater: 9 km/s compared with 3 km/s to reach Mars. A Mars true gravity assist to Jupiter can reduce this transfer requirement to about 6 km/s. Instead of Mars, a suitable asteroid in the main belt could be used with the tether artificial gravity assist method. Phasing would not be a problem, as it would be for Mars itself, since many of these asteroids are fairly evenly distributed in near-circular orbits, and are close to the ecliptic.

An even more advantageous method would be to transfer to Mars' orbit with about 4 km/s, and perform a Mars gravity assist into the asteroid belt. Next, use tether assists with several main belt asteroids in succession to gain the velocity required to reach out to Jupiter. By using Mars, lower relative velocities of the spacecraft with each asteroid will be needed, and hence a lighter tether may be used than with a single intermediary asteroid. We believe that in most of these applications all or a major part of the tether is reuseable. This is one advantage of using a tether compared with using rocket propulsion and expending fuel.

Similar scenarios for tether assist missions may be developed for the other outer planets. It should be remembered, however, that Jupiter remains the most powerful source for gravity assist in the solar system.

### Main Belt Asteroid Missions

The asteroid belt itself is the natural place for tether assist missions. As mentioned for the Jupiter mission, the spacecraft may utilize a Mars gravity assist to get from Earth into the asteroid belt. One can then imagine a spacecraft collecting samples of asteroid material at the same time it is performing a tether assist to fly on to another asteroid. After a tour of a number of asteroids, the process could be reversed by performing a gravity assist at Mars to return to Earth with the asteroid samples collected.

Main belt asteroid tours have been seriously considered using low-thrust rocket propulsion.[11] Successive rendezvous with from 4 to 8 asteroids would take up to 10 years. Although penetrators were suggested for in situ measurements, sample returns were not considered. Perhaps the ideal spacecraft to explore the asteroids in the main belt would use both low thrust and tether assist. With thrust, midcourse corrections could be made and the relative velocities at the asteroids could also be reduced to values where conventional materials would be adequate for a tether assist.

It is not known how many small asteroids (but still large enough for a tether assist) there may be in the main belt, since Earth-based telescopes cannot detect bodies smaller than about 1 km in diameter at that distance. There are probably more than a billion greater than 10 m in diameter with a mass greater than 1000 tons each, adequate for our method. In that case, the complete mission need not be preplanned based on knowledge of the position and orbits of selected asteroids that it should encounter. Instead, a spacecraft, thrust into the asteroid belt, could be capable of detecting 10-m asteroids at an adequate distance; for example, with passive optical sensors, backed up by ranging lasers once an object is detected. Then it would be determined whether the spacecraft can maneuver into position for a close flyby and perhaps a tether assist. In this manner, successive hops could be made with relatively little propulsion yet adding up to a considerable total velocity increment. No detailed analysis has been done on this unique mission as yet.

Finally, tethered assists may be valuable in the far future for possible economic utilization of asteroidal materials in space. It may, for example, be necessary to return asteroidal mass to the vicinity of the Earth or Moon on a continuing basis. Rather than expendable propellants, a set of permanent reuseable tether stations on a string of asteroids could provide the means to transport the mined material back to Earth.

## Conclusions

An alternate method of producing gravity assist using asteroids has been presented. Successful development of this technique will depend on many factors, some of the more important being: higher-strength tether material, a method of attaching and releasing a tether with an asteroid, tether dynamics control, and development of a navigation system to achieve the required accuracies for tether attachment and release.

Even when these problems have been solved, actual use of the system will be heavily mission-dependent. Tradeoff studies will be required to decide whether the tether system or conventional rocket propulsion or some combination of both is optimum for the mission goals. Tethers appear to have significant merit in missions where they can be reused several times. For highly repetitive use they may be the only practical devices.

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# ACHIEVABLE SPACE ELEVATORS FOR SPACE TRANSPORTATION AND STARSHIP ACCELERATION

N91-22162

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## ABSTRACT

Space elevator concepts for low-cost space launches are reviewed. Previous concepts suffered from requirements for ultra-high-strength materials, dynamically unstable systems, or from danger of collision with space debris. The use of magnetic grain streams, first proposed by Benoit Lebon, solves these problems. Magnetic grain streams can support short space elevators for lifting payloads cheaply into Earth orbit, overcoming the material strength problem in building space elevators. Alternatively, the stream could support an international spaceport circling the Earth daily tens of miles above the equator, accessible to advanced aircraft. Mars could be equipped with a similar grain stream, using material from its moons Phobos and Deimos. Grain-stream arcs about the sun could be used for fast launches to the outer planets and for accelerating starships to near lightspeed for interstellar reconnaisance. Grain streams are essentially impervious to collisions, and could reduce the cost of space transportation by an order of magnitude.

## INTRODUCTION

The major obstacle to rapid space development is the high cost of launching payloads into Earth orbit. Current launch costs are more than $3000 per kilogram, and rocket vehicles such as NASP, Sänger, or the Advanced Launch System will still cost $500 per kilogram. The prospects for space enterprise and settlement are not good unless these high launch costs are reduced significantly.

Over the past thirty years, several concepts have been developed for launching large payloads into Earth orbit cheaply using "space elevators." These structures can be supported by either static force balance or by the dynamics of moving masses. Their construction faces several technical difficulties, however. Enormous masses must be launched into orbit by conventional rockets to start the systems, extremely high-strength materials must be used to build the structures, and collision with natural meteoroids or man-made debris would shorten their lifetimes.

This paper discusses a complete space transportation system encompassing Earth-to-orbit launching of large quantities of material, easy transportation without rocket propellant between Earth orbit and the moon or Mars, solar launching facilities for high-speed transportation to the outer solar system, and the acceleration of near-lightspeed interstellar spacecraft. The concepts are based on the use of a stream of magnetic grains acting as the current-carrying element of an electric motor. Spaceships equipped with superconducting solenoids to generate superhigh magnetic fields could interact with the magnetic grain stream to propel themselves at high accelerations throughout the solar system. Similar grain streams at the destinations could decelerate the spaceships without requiring on-board rocket propellants. The result would be smaller, more capable spaceships. An even more important advantage would be the drastic reduction in the cost per kilogram of Earth-orbit launches.

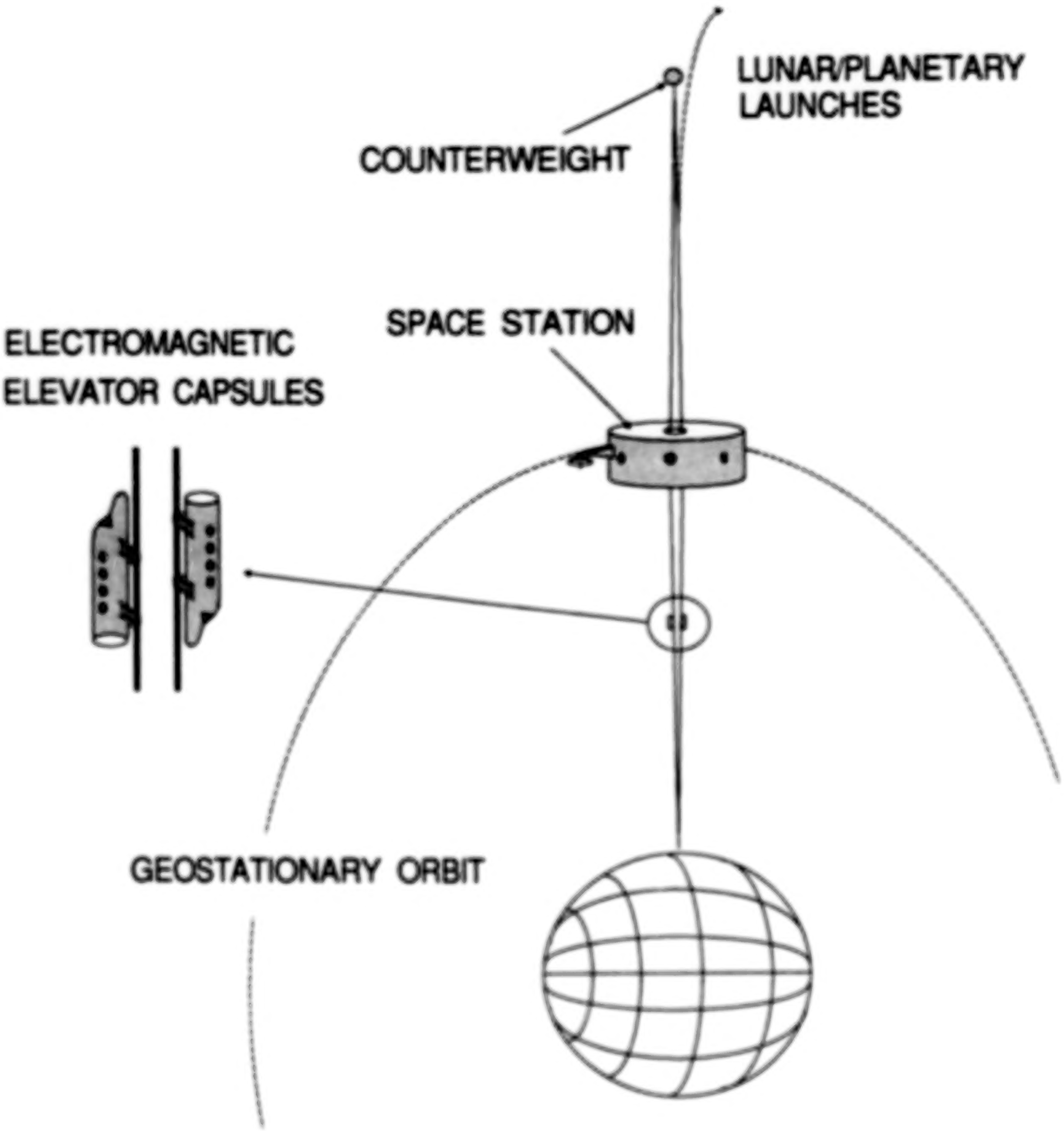
LUNAR/PLANETARY LAUNCHES
COUNTERWEIGHT
SPACE STATION
ELECTROMAGNETIC ELEVATOR CAPSULES
GEOSTATIONARY ORBIT

FIGURE 1. THE STATIC SPACE ELEVATOR

The classical concept for a space elevator is the statically balanced structure centered on the geostationary orbit and extended both upward and downward in balance until the lower end touches the equator; the upper end is then counterweighted to put the entire structure into tension, allowing it to lift payloads from the Earth into geostationary orbit. The concept was invented independently by Yuri Artsutanov (ref. 1.), John Isaacs, et al. (ref 2), and Jerome Pearson (ref. 3). Artsutanov proposed the launching of Earth-escape launches from the upper tower, and Pearson proposed the two-way traffic of electromagnetic vehicles from the ground to the geostationary space complex without external power. The concept requires an enormous amount of high-strength material and is vulnerable to severing by collision with debris.

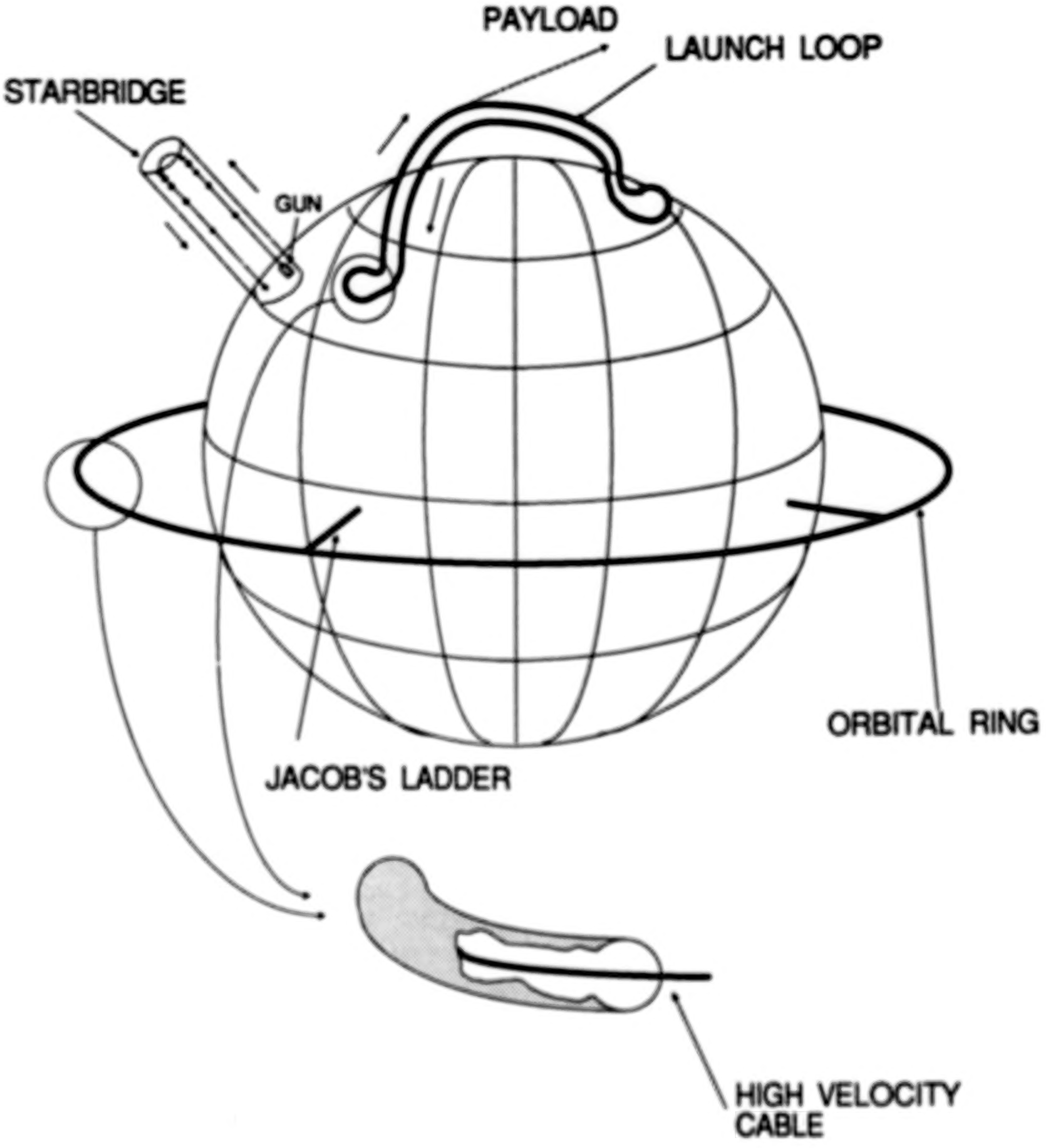

FIGURE 2. DYNAMIC SPACE ELEVATOR

To overcome the material strength and mass requirements of the static space elevator, several concepts were proposed to support space elevators by the dynamic forces of mass in motion. Paul Birch (ref. 4) proposed an orbital ring of conducting material moving faster than orbital velocity inside a thin torus circling the Earth. The moving cable produces an upward force on the torus, allowing the torus to support "Jacob's ladders" the few hundred kilometers down to the surface. Keith Lofstrom (ref. 5) and Birch proposed a similar moving conductor in a shorter "launch loop" that could accelerate electromagnetic vehicles to orbital velocity along the loop. Rod Hyde (ref. 6) proposed a purely vertical "starbridge" supported by the reaction force from accelerating downward a series of conducting rings fired upward from the base. These dynamic concepts solve the material strength problems of the space elevator, but they depend on the fail-safe operation of fundamentally unstable dynamic systems.

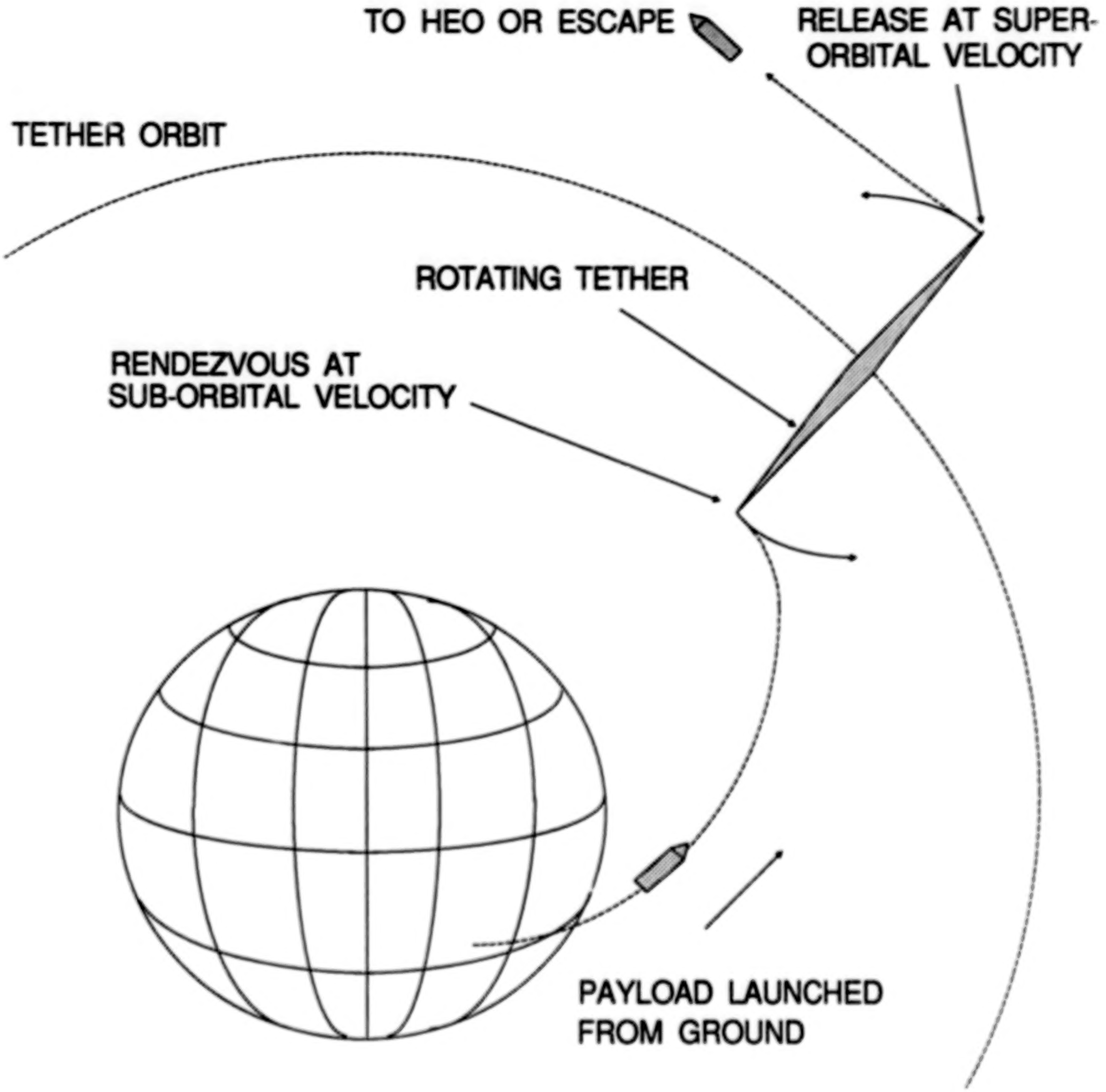
TO HEO OR ESCAPE
RELEASE AT SUPER-ORBITAL VELOCITY
TETHER ORBIT
ROTATING TETHER
RENDEZVOUS AT SUB-ORBITAL VELOCITY
PAYLOAD LAUNCHED FROM GROUND

FIGURE 3. THE ROLLING SATELLITE

Yet a third way to produce the space elevator is to set the static space elevator into rotation, and catch and launch payloads from the rapidly moving ends of this "rolling satellite." A rotating tether in orbit could provide most of the orbital energy needed for a payload launched from the ground by catching it on the lower end; the tether could then launch the payload into a higher orbit or to Earth-escape by releasing it from the upper end, half a tether rotation later. If payloads returning from higher orbits are also caught and released for Earth entry, no net energy is required by the rotating tether. Artsutanov originated the concept in 1969 (ref. 7); it was invented independently by Hans Moravec (ref. 8) and put into a practical form by Pearson (ref. 9). The concept faces the problems of high-speed rendezvous and tether dynamics.

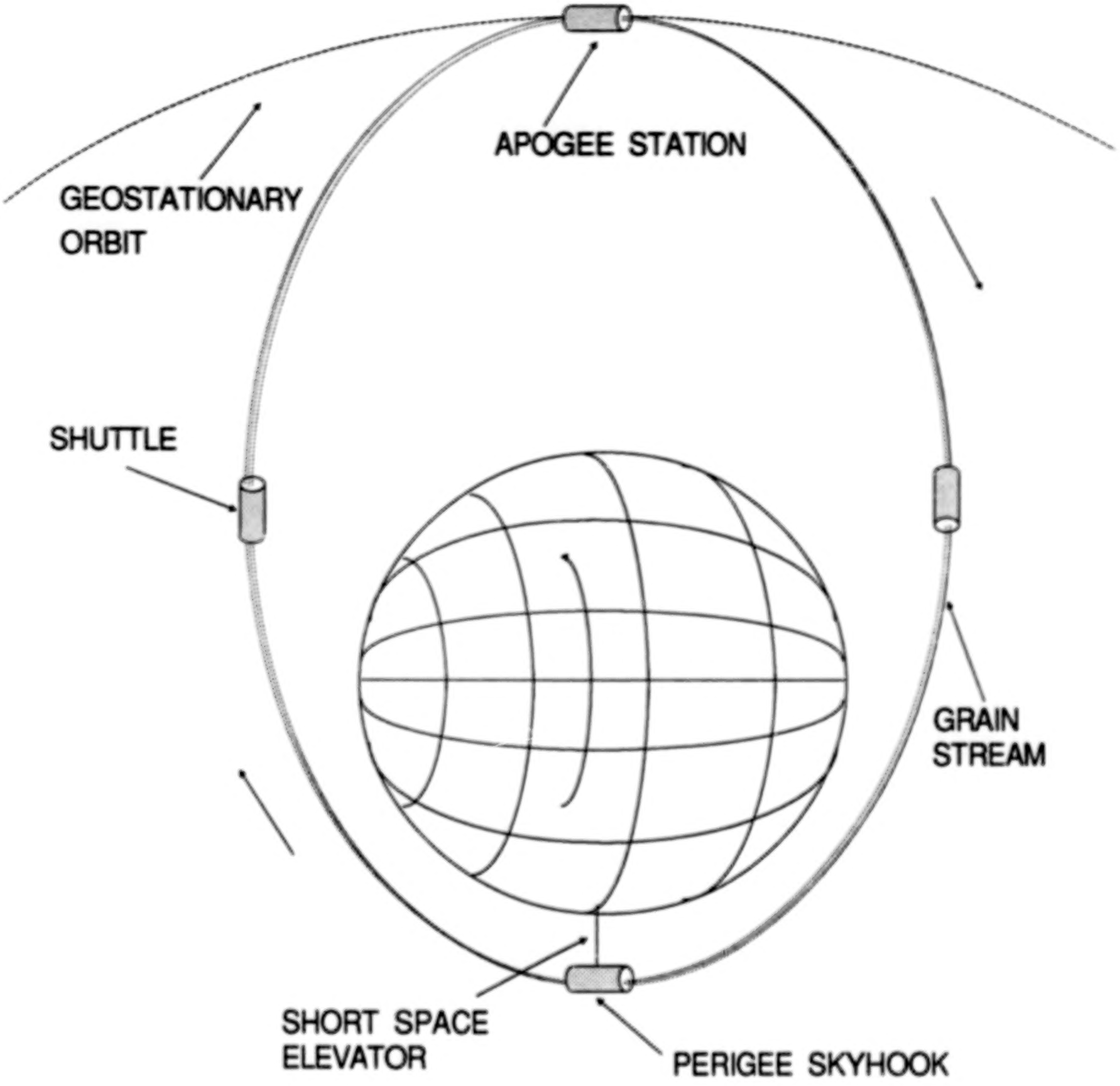

FIGURE 4. THE GRAIN-STREAM SPACE ELEVATOR

A space elevator concept that overcomes all the problems of previous attempts was proposed by Benoit Lebon in 1986 (ref. 10, 11) and elaborated upon in 1987 (ref. 12). The concept is similar to Birch's orbital ring, but it uses a much less massive stream of magnetic grains and dispenses with the massive enclosing torus. The grain stream is kept in place by shepherding solenoids that move slowly along the stream. This shepherding action is done naturally by the gravitational fields of small satellites on thin rings about Saturn and the other giant planets. The solenoidal vehicles would have a much easier job because of the far stronger magnetic forces available. A grain stream in elliptical orbit could support a large space station in geostationary orbit and a short space elevator from a perigee skyhook. The system has the advantages that it could be built in stages, increasing the stream density to carry higher payloads, and that it is impervious to collision. The short space elevator could be almost entirely in the atmosphere, where debris could not penetrate. The magnetic grains could be obtained from the moon or from a nearby asteroid.

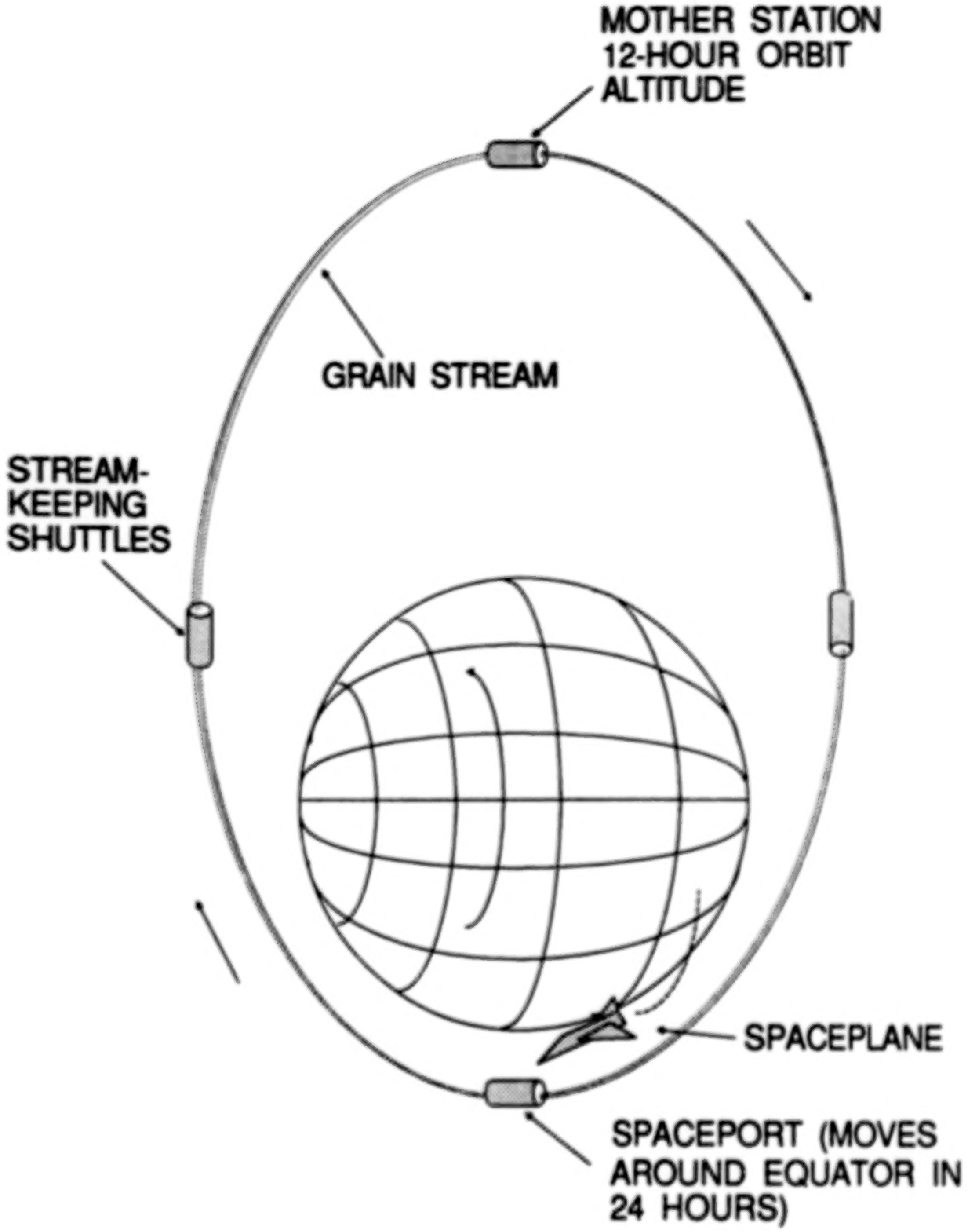
MOTHER STATION
12-HOUR ORBIT
ALTITUDE
GRAIN STREAM
STREAM-
KEEPING
SHUTTLES
SPACEPLANE
SPACEPORT (MOVES
AROUND EQUATOR IN
24 HOURS)

FIGURE 5. THE INTERNATIONAL "SPACE TELPHER" SYSTEM

In 1988, Lebon (ref. 13) proposed an alternate grain-stream launch system that is even more advantageous than the original. He replaced the short space elevator with a "space telpher," or travelling spaceport supported by the grain stream and moving completely around the equator once per day. This spaceport could be at less than 100 kilometers altitude, allowing advanced aircraft from all nations to reach the spaceport daily. Once there, grain-stream shuttles could move to the mother station at 12-hour orbital altitude and be launched to Earth escape. In a sense, this is a space elevator without the elevator, and it would not be tied down to one location on the equator. By not having to support an enormous elevator, the grain stream could be smaller, making the entire system simpler and cheaper.

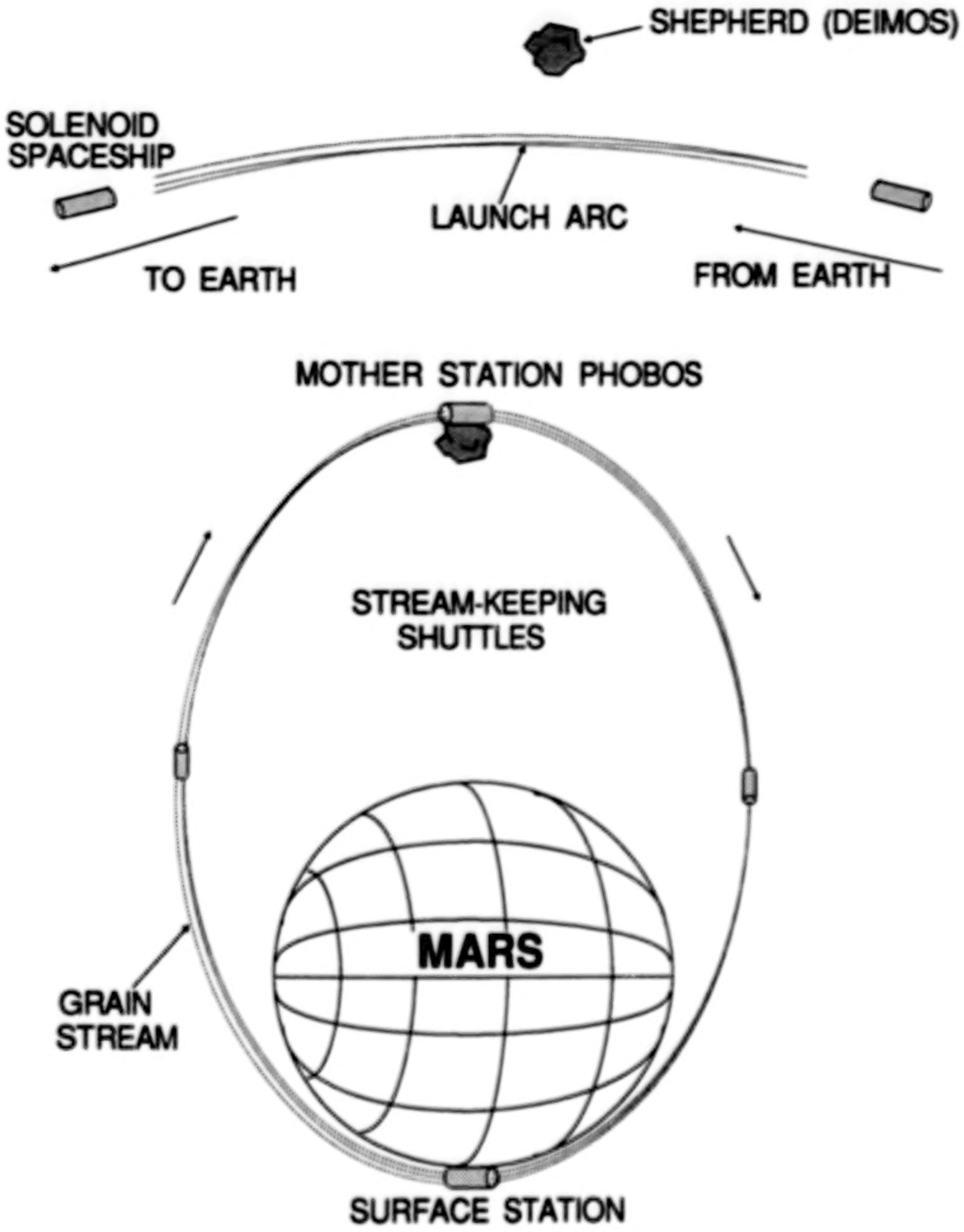
SHEPHERD (DEIMOS)
SOLENOID SPACESHIP
LAUNCH ARC
TO EARTH
FROM EARTH
MOTHER STATION PHOBOS
STREAM-KEEPING SHUTTLES
MARS
GRAIN STREAM
SURFACE STATION

FIGURE 6. MARTIAN RING TRANSPORTATION SYSTEM

Once it is cheap and simple to launch cargo and people into Earth orbit, we will be able to develop Mars as a sister planet, using a similar grain-stream transportation system. Because of the low atmospheric pressure on Mars, the lower station could be right on the surface, with the mother station being at "arestationary" altitude. The Martian moon Phobos could provide the raw material for the grain stream, and its remnant could be moved to form the mother station. A grain-stream ring arc could be created near the orbit of the outer Martian moon Deimos, perhaps partially shepherded by Deimos itself, in addition to streamkeeping shuttles. This arc could accelerate solenoidal spaceships bound for Earth and decelerate spaceships arriving from Earth. A similar launch arc about the Earth would complete a two-planet transportation system.

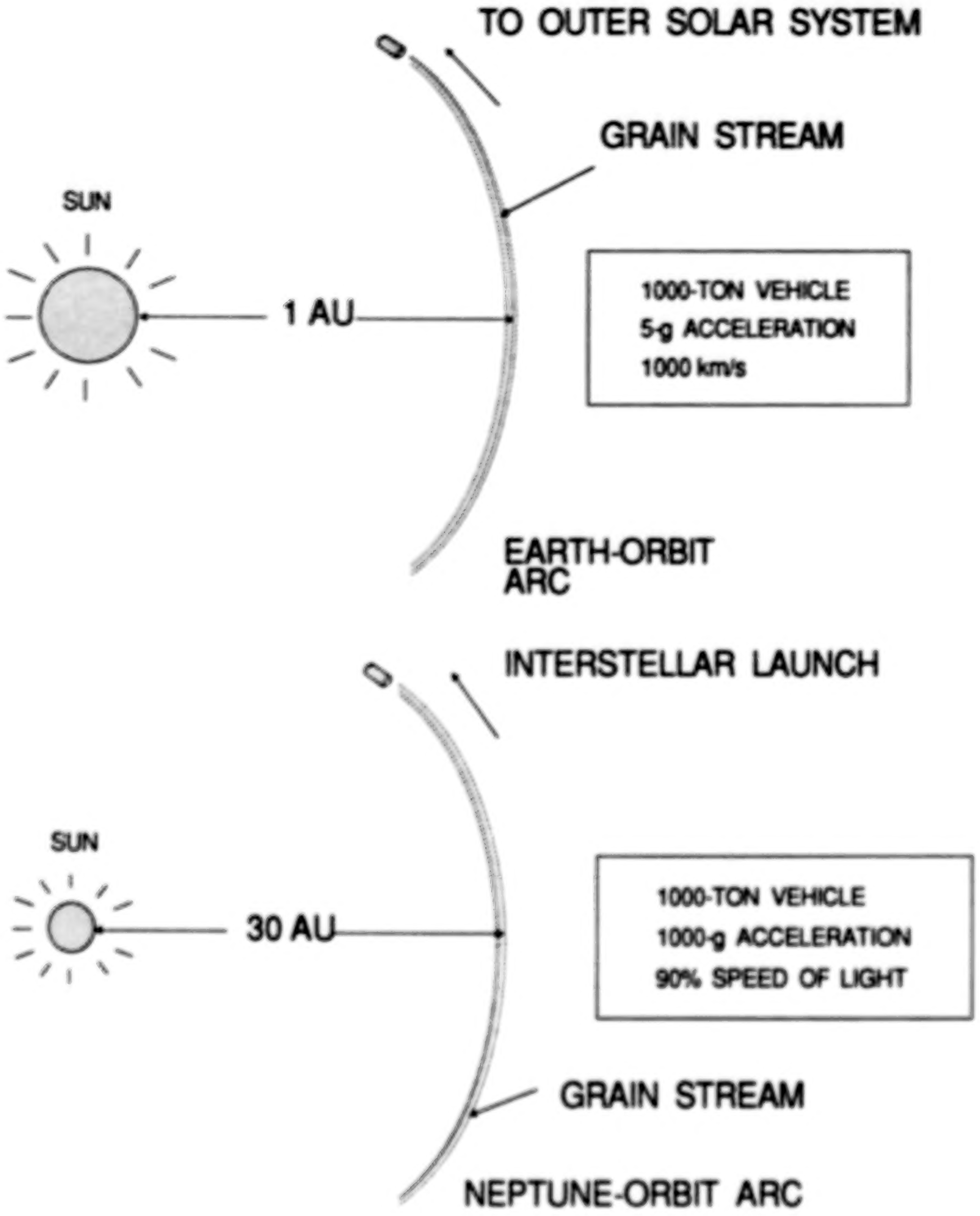
TO OUTER SOLAR SYSTEM
GRAIN STREAM
SUN
1 AU
1000-TON VEHICLE
5-g ACCELERATION
1000 km/s
EARTH-ORBIT ARC
INTERSTELLAR LAUNCH
SUN
30 AU
1000-TON VEHICLE
1000-g ACCELERATION
90% SPEED OF LIGHT
GRAIN STREAM
NEPTUNE-ORBIT ARC

FIGURE 7. SOLAR RUNWAYS

The partial grain stream concept was originated by Lebon (ref. 11) as a method for achieving extremely high launch velocities. He proposed creating a partial ring in solar orbit at the Earth's distance from the sun, and using it to accelerate solenoidal spaceships to about 1000 km/s in order to reach the outer solar system in just a few months. The acceleration is limited by the magnetic field achievable on the spacecraft. With a field of 0.06 tesla, a 1000-ton spacecraft could be accelerated to 1000 km/s at 5 g in less than 7% of the circumference of the Earth's orbit. A larger "solar runway" could be placed about Neptune's distance from the sun for interstellar launches. A 15-tesla solenoid could accelerate a 1000-ton vehicle at 1000 g to 90% the speed of light from such an arc. Much more modest fields would suffice to launch fast interstellar probes for the initial reconnaissance of nearby planetary systems in a few decades. Ring orbit perturbations could be minimized by launching vehicles in opposite directions along the grain-stream arc.

## CONCLUSIONS

The ancient dream of the space elevator is now possible. Magnetic grain streams and high-magnetic-field solenoidal spacecraft could solve the problem of the high cost of space launching and result in rapid space colonization and industrialization. This form of space transportation could also greatly reduce the travel time to the outer solar system, bringing the resources of the outer planets into our realm. Finally, grain-stream arcs in solar orbit could be the launching ramps for fast reconnaissance missions to the nearby stars that could provide data on extra-solar planetary systems within a decade or two of launch.

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N91-22163

# Reactionless Propulsion Using Tethers

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NASA Lewis Research Center

## Abstract

A orbiting tethered satellite can propel itself by reaction against the gravitational gradient, with expenditure of energy but with no use of on-board reaction mass. Energy can be added to the orbit by pumping the tether length in the same way as pumping a swing. Examples of tether propulsion in orbit without use of reaction mass are discussed, including: (1) using tether extension to reposition a satellite in orbit without fuel expenditure by extending a mass on the end of a tether; (2) using a tether for eccentricity pumping to add energy to the orbit for boosting and orbital transfer, and (3) length modulation of a spinning tether to transfer angular momentum between the orbit and tether spin, thus allowing changes in orbital angular momentum.

## 1. Introduction

A tether is a long, flexible cable which connects one part of a satellite with another. Although quite simple, many very interesting things can be done in space using tethers [1-3]. In the equilibrium configuration, as shown in figure 1, the tether is oriented radially outward, with a tension on the tether due to the gravitational gradient (or "tidal") force.

The effective acceleration due to the gravity gradient a distance $z$ from the center of mass (CM) is, to first order:

$$a_{eff} = 3\, g_e r_e^2 z / r_o^3, \tag{1}$$

where $g_e$ is the gravity at the Earth's surface, $r_o$ is the orbital radius and $r_e$ is the radius of the earth.

Most analyses of tether orbits assume that the center of mass of a tethered satellite system remains in the original orbit; *i.e.*, that the angular velocity of the tethered satellite does not change as the tether is extended or retracted. This is true only to the first order approximation in tether length. Briefly, the mass that extends outward experiences an increase in centrifugal force that increases linearly with distance, but the mass that extends inward experiences gravity that increases faster than linearly. Thus, as the tether is unreeled, the center of mass of the orbit is pulled inward. To conserve angular momentum, the angular velocity of the orbit increases.

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The effect that a tethered satellite can extend across the gravity gradient can be used for propulsion. The following analysis (sections 2 and 3) follows my calculations from reference 4. Similar results to section 3 are also derived in reference 5.

## 2. Orbital Repositioning of a Satellite

In the following section we assume a tether of negligible mass in circular orbit. The extension of the analysis to tethers of non-negligible mass is straightforward.

Consider a satellite of mass $m_i$ consisting of two pieces of mass $m_1=m_2=m_i/2$ connected by a tether. The initial orbit is assumed to be circular, with an angular velocity $\omega_o$ and an initial orbital radius (measured from the Earth's center) $r_o$. With the tether at initial length zero, the orbit has initial angular momentum

$$L_i = m_i \omega_o r_o^2. \tag{2}$$

Now assume that the tether is extended to length $x$ in each direction from the CM, as shown in figure 1. The total length is $2x$. Note that energy decreases, since in deploying a tether work is done by the effective tidal force. Angular momentum is still conserved,

$$L = m_1 \omega r_1^2 + m_2 \omega r_2^2, \tag{3}$$

where $r_{cm}$ is the orbital radius of the CM, and $r_1=r_{cm}-x$ and $r_2=r_{cm}+x$. The inward tension on the low end of the tether must equal the outward tension on the high end of the tether. If we expand to second order in $x$, then set equation (2) equal to equation (3) to solve for $\omega$ and $r_{cm}$ as a function of tether extension $x$, we find the center of mass drops,

$$r_{cm} = r_o - 5\,\frac{x^2}{r_o} \tag{4}$$

and the orbital period P increases as the tether extends:

$$P = P_o[1 - 9(\frac{x}{r_o})^2]. \tag{5}$$

For example, a GEO satellite consisting of two equal masses on a 1000 km long tether will have a period faster than that of an untethered satellite by 0.44° per day.

Inclusion of higher order terms results in an increase in the effect.

If the two masses are allowed to differ, the orbital period change is proportional to $m_1 m_2/(m_1+m_2)$, which is maximum when the two masses are equal.

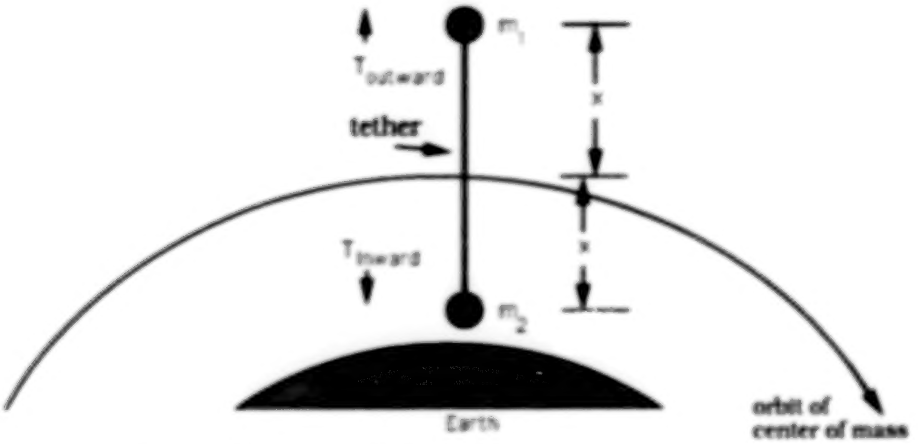

$T_{outward}$
tether
$T_{inward}$
$m_1$
$m_2$
x
x
Earth
orbit of center of mass

*Figure 1.* **Tether orbit and definitions**

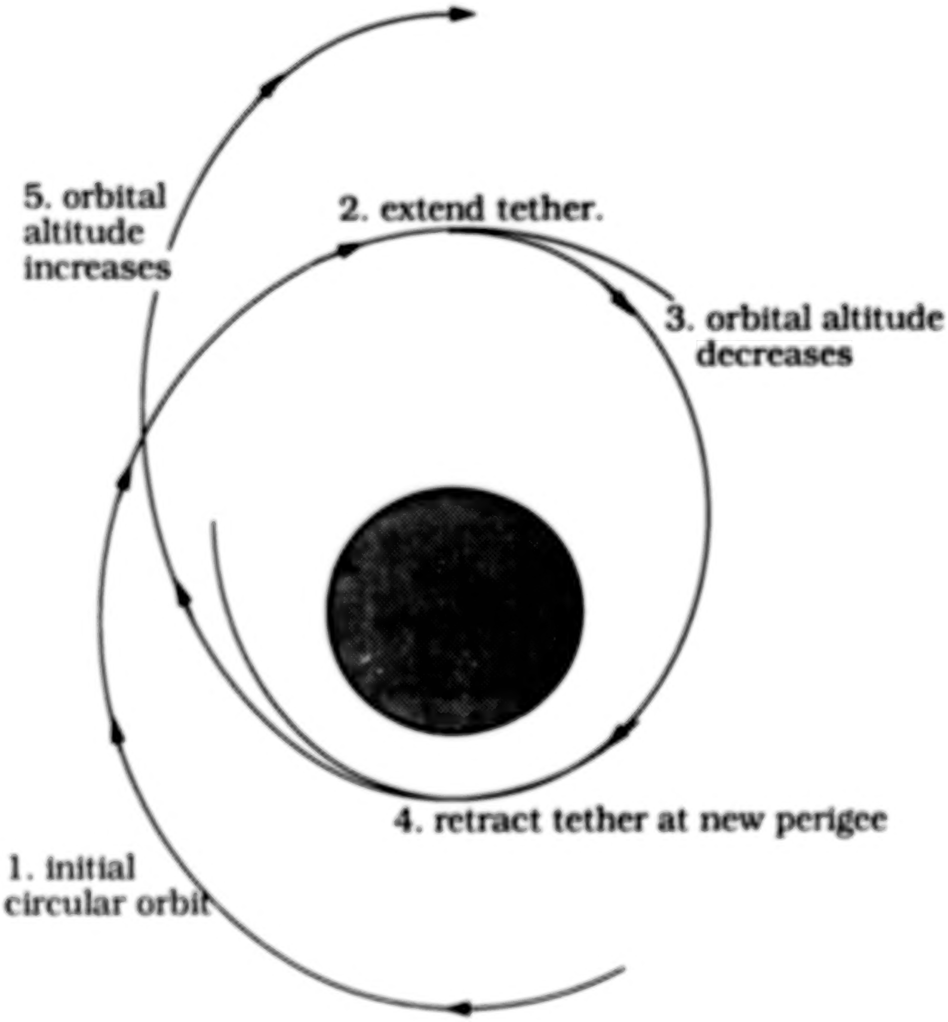

5. orbital altitude increases
2. extend tether.
3. orbital altitude decreases
4. retract tether at new perigee
1. initial circular orbit

***Figure 2.* Tether length variation causes eccentricity change from initially circular orbit (eccentricity greatly exaggerated)**

## 3. Orbital Propulsion by Eccentricity Pumping

The preceeding analysis has assumed equilibrium conditions, *i.e.*, that the orbit remains circular during the extension and deployment of the tether. This assumption is true only if the tether is deployed or retracted over a time greater than an orbital period. Faster deployment will result in dynamic changes to the orbital eccentricity. This is shown in figure 2, where an initially circular orbit is altered to an eccentric orbit by modulation of the tether length. Continuing the length modulation allows eccentricity to be continuously increased (or decreased). This effect can be used as a means for orbital propulsion that does not require expenditure of reaction mass. In the process energy is added to the orbit (from a power source on board the spacecraft), while the orbital angular momentum is constant. Particular applications are injection of a spacecraft into an escape orbit from an initially circular orbit, and use of the process for transfer orbits, *e.g.*, LEO to GEO.

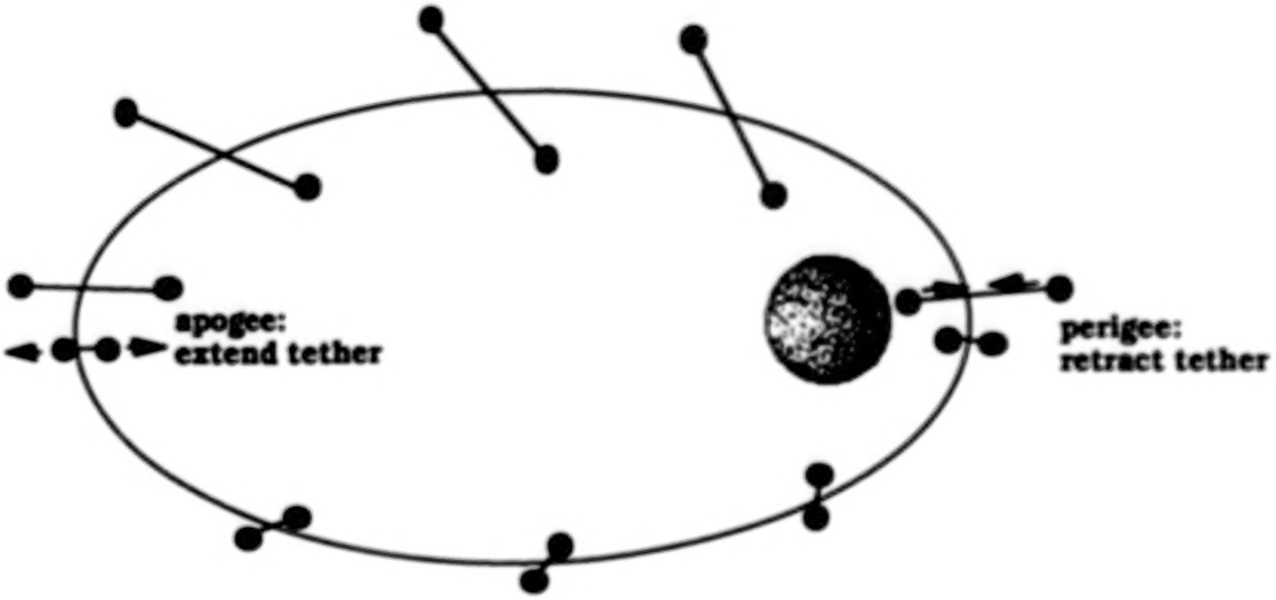
The method is straightforward. A mass is deployed away from the spacecraft on the end of a tether. The stable configuration is with the tether oriented radially from the central body. The tether is mounted on a reel with a motor which can pull it in or let it out. The method of orbit pumping consists of pulling the tether in at perigee (more generally, periapsis) and letting the tether out at apogee. Since gravitational gradient (tidal) forces are to first order proportional to the inverse radial distance cubed, more mechanical work is done against the tidal force in pulling the tether in than is returned when the tether is let back out. Thus, energy is added to the orbit. Since the orbital angular momentum is unchanged, the eccentricity $\epsilon$ of the orbit increases. This is shown in schematic in figure 3.

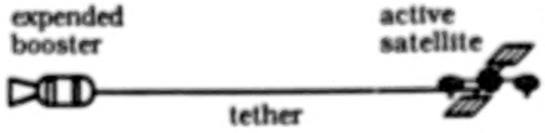
As an aside, it may be noted that this process is essentially the same as the process of adding energy to a playground swing by "pumping" [6].

Contrary to expectations, the eccentricity pumping process is most effective when the orbit is nearly circular. Although the amount of energy available per orbit decreases as the orbit becomes nearly circular, the sensitivity of the eccentricity $\epsilon$ to small changes in energy increases as $1/\epsilon$, and this factor dominates over the decrease in energy. For a perfectly circular orbit, higher order terms contribute as well.

The reverse process, circularizing an eccentric orbit by removal of energy by a viscoelastic tether, has been discussed in detail by Columbo *et al.* [7]. This is equivalent to tidal damping, a natural phenomenon that accounts for the fact that most of the moons in the solar system have nearly circular orbits.

Eccentricity pumping can only be done if the initial orbit is high enough that the minimum perigee does not impact the primary. For pumping from an initial circular orbit at distance a to escape this implies that $a_o \geq 2\ r_e$. In general, the minimum perigee must also be high enough not to intersect the atmosphere. This corresponds to an initial orbital radius of ~13,150 km, or a minimum initial orbital altitude of 6,575 km.


apogee:
extend tether
perigee:
retract tether


expended
booster
active
satellite
tether

**Tether detail**

***Figure 3.*** **Eccentricity pumping (schematic). Tether is retracted at perigee of orbit, extended at apogee. Since the gravity gradient is higher at perigee than at apogee, more energy is input to the system during retraction than is recovered during extension of the tether, and so net work is done. This work results in an increase of orbital energy.**

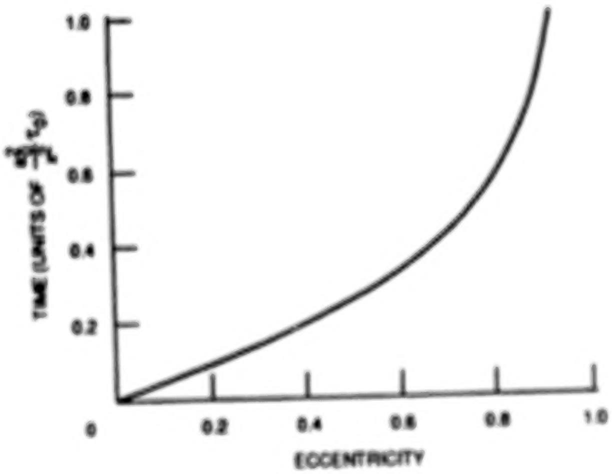


**Figure 4. Time required to reach a given eccentricity.**

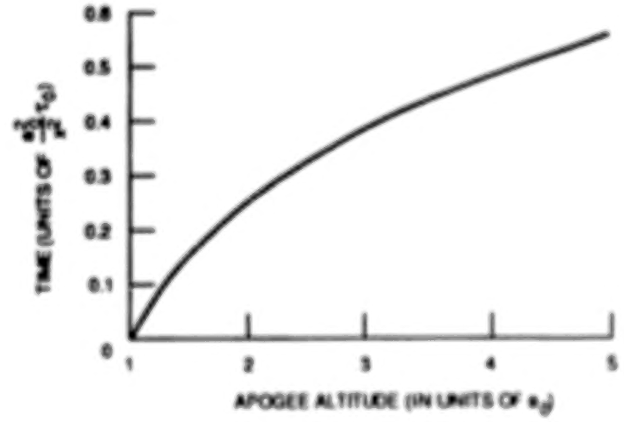


**Figure 5. Time required to reach a given apogee altitude (in units of initial altitude**

Again, assume that the tether itself is of negligible mass. The case of a massive tether can be straightforwardly calculated by integration over the mass distribution. We also assume that the masses on each end of the tether are equal, the case which for fixed tether length maximizes the effect. Extrapolation to unequal masses is straightforward. Assume equal masses m/2 extended on the ends of a tether of full length d (*i.e.*, half-length $z$). The mechanical energy stored in the tether is:

$$E = \frac{3}{8}\, m g_e \frac{r_e^2}{r_o^3} d^2 \tag{6}$$

Now assume the orbit is elliptical, with eccentricity $\epsilon$. The orbital energy is

$$E = -m g_e\, \frac{r_e^2(1-\epsilon^2)}{2a_o} \tag{7}$$

The amount of energy required to retract the tether at perigee minus the amount recovered in extending the tether at perigee is:

$$\frac{\Delta E}{orbit} = \frac{3}{8}\, m g_e d^2\, \frac{r_e^2}{a_o^3}\, \epsilon(6+\epsilon^2) \tag{8}$$

In the real case, the tether length d will not reeled in all the way to zero length. An effective value of d can be used, $d_{eff}^2 = d_{max}^2 - d_{min}^2$.

The sensitivity of eccentricity to changes in energy is

$$\frac{d\epsilon}{dE} = \frac{a_o}{m g_e r_e^2 \epsilon} \tag{9}$$

The orbital period, expressed in terms of the orbital period of the initial circular orbit, is

$$\tau = \tau_o(1-\epsilon^2)^{-\frac{3}{2}} \tag{10}$$

The average power required is:

$$\frac{dE}{dt} = \frac{3}{8} m g_e\, \frac{r_e^2}{a_o^3 \tau_o}\, d^2\, \epsilon(6+\epsilon^2)(1-\epsilon^2) \tag{11}$$

The function of $\epsilon$ has a maximum of 2.45 at $\epsilon$=0.6. In practical units, this is an average specific power of

$$\hat{P} = (280w/kg)\ a_o^{-9/2} L^3\ \epsilon(2.45 + 0.41\epsilon^2)(1-\epsilon^2) \qquad (12)$$

where $a_o$ is the initial semimajor axis in multiples of the earth radius $r_e$ and L is the tether length in thousands of kilometers. Note that the function of $\epsilon$ has been normalized to a maximum value of 1. Since for most applications d will be << 1000 km and $a_o$ will be >~1.5 $r_e$, this specific power is well within achievable levels. The peak power levels required will be higher, and depend on how fast the tether is reeled in.

Then the rate of change of eccentricity is

$$\frac{d\epsilon}{dt} = \frac{d\epsilon}{dE}\frac{dE}{dt} = \frac{3}{8\tau_o}\frac{d^2}{a_o^2}\,(6+\epsilon^2)(1-\epsilon^2)^{\frac{3}{2}} \qquad (13)$$

To find the time required to reach a given eccentricity, this expression is inverted and integrated. The integral can be done exactly,

$$t(\epsilon) = \frac{8}{3}\tau_o\frac{a_o^2}{d^2}\ \frac{1}{7\sqrt{42}}\ tan^{-1}\left[\frac{7\epsilon}{\sqrt{42(1-\epsilon^2)}}\right] + \frac{\epsilon}{7\sqrt{1-\epsilon^2}} \qquad (14)$$

Figure 4 shows the time to reach a given eccentricity. (Here time is plotted in the units of $a_o^2/d^2\tau_o$, which is simply the initial orbital period scaled by the squared tether length. Initial orbital period is 90 minutes for LEO). Escape is approached asymptotically.

It is also of interest to look at the time required to reach a given apogee altitude; this is shown in figure 5. Again time is in units of the scaled orbital period, and altitude is given as a multiple of the initial orbital radius (measured from the center of the Earth). At long times the altitude increases almost linearly with time.

As an example, consider the case where eccentricity pumping is used to move from LEO to geosynchronous transfer orbit. $a_{GEO}$=6.63 $r_e$, so $a_{LEO-GEO}$ = 3.82 $r_e$, and $\epsilon_{LEO-GEO}$ = 0.738. The minimum perigee requirement leads to a minimum initial orbital radius of $a_o$ = 1.74 $r_e$, i.e., initial orbital altitude 4700 km and initial orbital period $\tau_o \sim$ 200 min. From figure 6, the time needed is 0.43 $\tau_o(\frac{a_o}{d})^2$.

For a 500 km tether length, $(\frac{a_o}{d})^2$ ~500, and the orbital pumping process takes 725 hours, or about 31 days.

A efficient technique for the apogee kick would be to continue eccentricity pumping until the apogee is well past GEO, perform the apogee kick, then use eccentricity pumping in reverse to circularize the orbit. The amount of velocity change ΔV required to give the orbit sufficient angular momentum to attain circular orbit at GEO is inversely proportional to the distance. Thus, in theory, the ΔV needed for apogee kick could be made arbitrarily low by pumping the apogee to a high enough initial value, although this would take a long time.

The decrease in eccentricity rate with increasing eccentricity is due to several factors, one of them being that the energy transfer per orbit is proportional to the square of the tether length over the orbital semimajor axis. As the eccentricity, and thus the semimajor axis, increases, the rate decreases. Since the tether is retracted at perigee, the effect can be eliminated by increasing the tether length as the apogee distance increases. (As tether length increases, this will require initiating the retraction slightly before perigee.) Note that since the total stress due to tidal force goes as $d^2/r^3$, the maximum stress at apogee decreases despite the increased tether length.

The requirement for a minimum initial altitude comes from the necessity that the minimum perigee of the orbit not intercept the atmosphere. This requirement can be alleviated if the pumping maneuver is combined with an incremental $\Delta V$ at each apogee to increase the angular momentum just sufficiently to keep the perigee from decreasing.

## 4. Propulsion Using a Spinning Tether

The calculations in the preceeding section have all been done assuming that the tether is not spinning. This is not required by physics, but is a practical consideration due to the limits of real material strength. For a spinning tether, centrifigal stresses can very rapidly become extremely large. The constraint to avoid spinning the tether during retraction will put a limit on the maximum modulation $d_{eff}/d$ possible.

If the tether *is* allowed to spin, however, a vastly more effective propulsion method which is not limited by orbital angular momentum is possible. This is shown in schematic in figure 6. For example, to increase the orbital altitude, the tether is extended while horizontal, and retracted when vertical. Since in retraction work is done against the gravitational gradient, the orbital energy increases. The orbital angular momentum also increases, and the spin of the tether increases (if the spin is opposite to the orbital direction) or decreases (if the spin is the same as the orbit). Effectively, angular momentum is transferred from the orbit to the tether.

The amount of energy transferred per spin is nearly independent of the spin rate, and thus the higher the spin rate of the tether, the faster (in principle) the orbital energy can be changed. The rate of orbital change is limited only by the power source and the materials strength. If the energy source had sufficiently high power, it would even be possible to propel past escape velocity.

Orbital plane changes are also possible (although slightly more difficult to illustrate). For plane changes the tether spin axis is optimally in the plane of the orbit, and again the tether length is modulated in phase with the orbit.

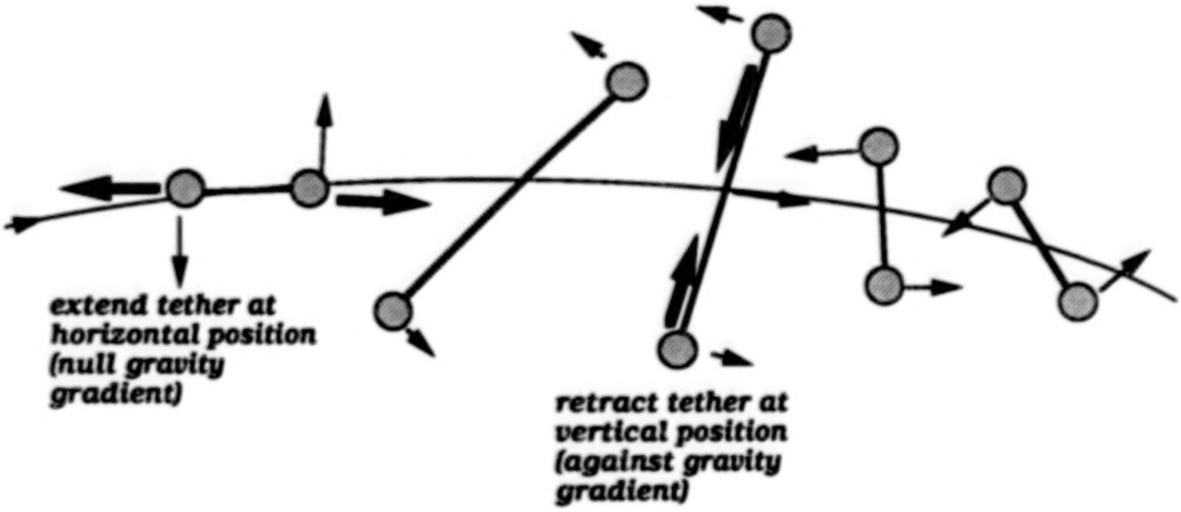

extend tether at horizontal position (null gravity gradient)
retract tether at vertical position (against gravity gradient)

*Figure 6.* **Orbital propulsion using a spinning tether (orbital motion here is clockwise; tether spin counterclockwise).**

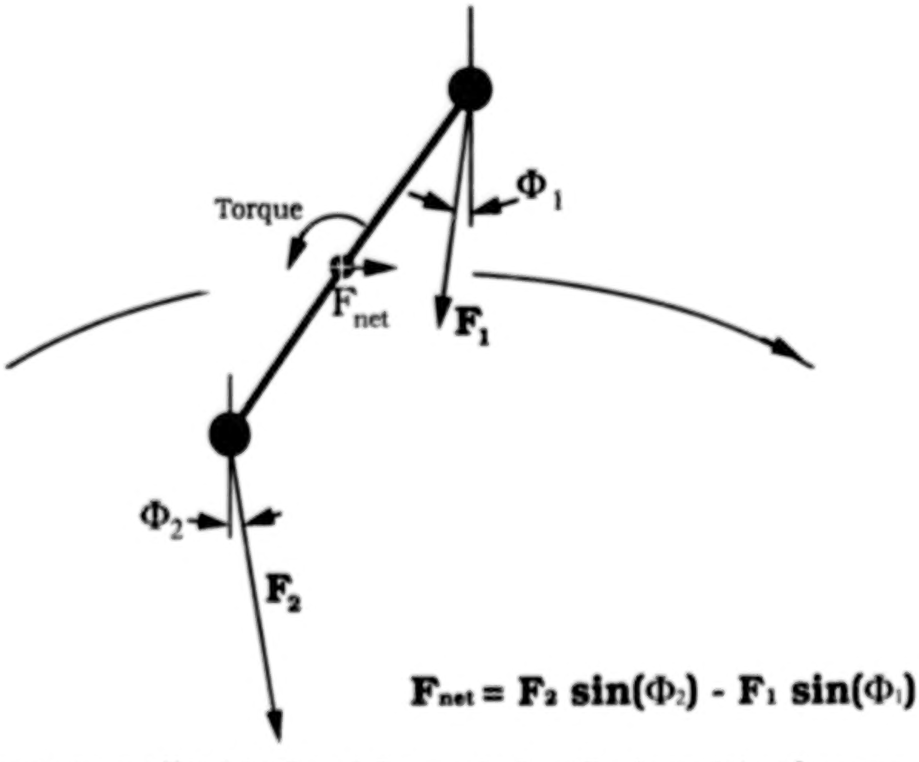

Torque
$\Phi_1$
$F_{net}$
$\mathbf{F_1}$
$\Phi_2$
$\mathbf{F_2}$
$\mathbf{F_{net} = F_2 \sin(\Phi_2) - F_1 \sin(\Phi_1)}$

*Figure 7.* **Forces on a tilted tether (shown in locally inertial reference frame). Since $F_2 > F_1$ and $\phi_2 > \phi_1$, there is a net side force on the center of mass of the tether as well as a torque.**

An alternative view of this propulsion system, showing the origin of the forces, is shown in figure 7. When the tether is not aligned vertically, there is a net side force on the tether due to the fact that the gravity on the two end masses is not the same. In general for a spinning tether this side force averages to zero. However, by (for example) increasing the tether length when the tether is angled to the left of vertical, and decreasing the tether length when it is to the right of vertical (*i.e.*, modulating the length) the average can be made non-zero. This side force is then used for propulsion.

For real materials the amount of angular momentum which can be stored in the tether is limited. An untapered material can rotate at a maximum tip speed which is characteristic of the material, $v_t = \sqrt{[(\textit{breaking stress})/\textit{density}]}$. This value $v_t$ is slightly under 2 km/sec for the best currently existing fiber. Defect-free fibers of high-strength materials–silicon carbide, diamond–have theoretically much better values, and ten times this value, 20 km/sec, is not unreasonable to expect in the long term. For constant tip speed, angular momentum increases linearly with tether length, and so the effectiveness increases with tether length.

As an example, suppose a spinning tether is used for propulsion from LEO to escape. Angular momentum at LEO is (6500 km)(7.9 km/sec) or about 50,000 $km^2$/sec. Angular momentum at escape is $km^2$/sec. The difference, about 20,000 $km^2$/sec, must be taken up in tether spin. At a maximum $V_t$ of 2 km/sec, the tether length required will be 10,000 km. If 20 km/sec $V_t$ could be achieved, the required tether length is only 1000 km, a tether length which is not unreasonable to expect to be achievable in the long term.

Alternatively, if angular momentum can be transfered to some external sink, this may not be a limitation. The obvious choice is to transfer momentum to the Earth's magnetic field via a magnetic torquer, such as is used in many satellites for orientation control. This could be done by a method as simple as driving an alternating current along the length of the tether and using the v x B potential to drive energy through a load (appropriately this load would be the tether winch motor, allowing the energy put into tether spin to be recovered). This then becomes conceptually similar to electrodynamic tether propulsion (see, for example, discussions in references 1-3]), except that tether spin velocity is substituted for orbital velocity, and since required the current is AC, no return path along the space plasma is required.

## 5. Conclusions

A tethered satellite system can extend significant distances across the gravitational gradient of the body it is orbiting. This effect can be made use of, using the gravity gradient itself for propulsion. Several applications are discussed. These applications are noteworthy as examples of raising an orbit "by its bootstraps" by pulling against the gravity gradient.

Of course, these propulsion systems are not reactionless in the physics sense: Newton's law of conservation of momentum is not violated, since momentum is transferred to the Earth (or primary) by the gravitational attraction. However, they *are* reactionless in a real, engineering sense, in that no propellant is expended as reaction mass. If a limitless energy source is available, such as a solar power system, the tether system can maneuver completely in Earth orbit.

## 6. Acknowledgments

This work was done while the author was a NASA/NRC Resident Research Associate at NASA Lewis. I would like to acknowledge the discussions, suggestions, and support of Frank Hrach at NASA, and of Dr. Robert Forward at Forward, Unlimited, who contributed significantly to this work.

I would also like to note that use of tethers for propulsion by eccentricity change of an orbit has been proposed independently, and earlier, by Manuel Martinez-Sanchez and Sarah A. Gavit of MIT [5].

## 7. References

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N91-22164

# EXPLORATION OF PLANETESIMALS BY A TRIPARTITE TETHERED SPACECRAFT

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## ABSTRACT

Asteroids and comets exert such a small gravitational force that it is not practical to survey them from orbit. One must instead continuously accelerate using maneuvering rockets to move around the surface. A space exploration craft in three parts connected by lightweight cables can survey asteroids and comets, and deploy landers, without requiring the large thrusters and the continuous depletion of fuel required by a single craft.

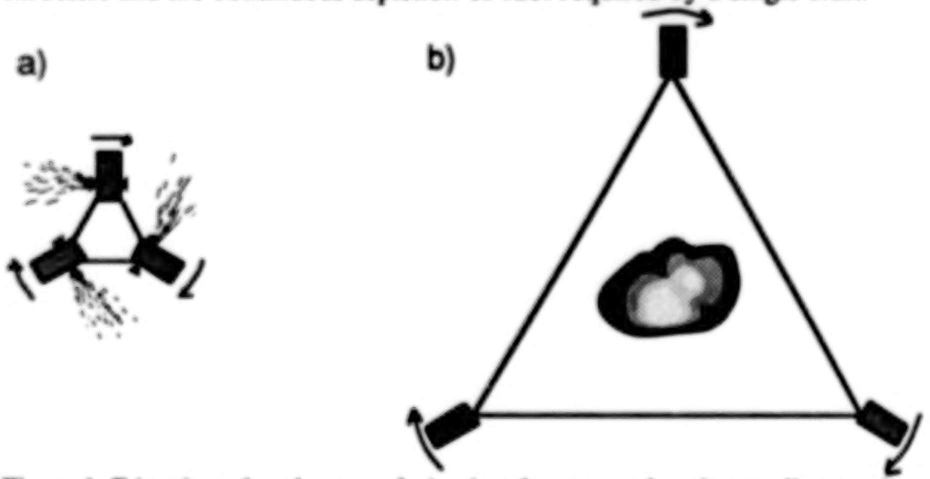

Figure 1: Tripartite tethered spacecraft a) using thrusters and paying out line to spin-up from single unit to b) spinning configuration up to 100 km across able to indefinitely survey the surface of a planetesimal.

The spacecraft is deployed by spinning up from a compact configuration using low thrust jets, and then maintains surveying orbit without any major expenditure of energy.

The triangular tether arrangement is stable, but care must be taken in changing orbits and with deploying and recovering samplers, as can be demonstrated with a simple simulation. Even 100 km long tethers occupy a low payload fraction.

## BACKGROUND

An important goal of our space program is to understand the origins and evolution of our solar system. The planetesimals — the comets and asteroids, which litter the system are some of its oldest, least modified pieces, and so it is important to study them and retrieve samples.

Though they are tiny relative to the planets there are still some sizeable pieces (ref. 1); more than 500 main belt asteroids are known with diameters from 40 up to 600 km. Near

Earth, most are small objects with diameters up to 10 km. The largest by far is 433 Eros with dimensions 18x36 km. There are believed to be thousands of earth approaching asteroids with diameters nearer 1 km, and many times that number in the Main Belt. The size of comets are less well known; their solid nucleus is obscured by evaporating gases and glowing plasma. The recent interception and investigation of comet Halley, one of the biggest, showed its core to be potato shaped 16 km X 9 km but its density is only 100 to 300 $kg/m^3$ (ref. 2).

The gravitational attraction of even the largest asteroids is miniscule, and that of the comets, with their low density, is completely insignificant. A 10 km diameter spherical asteroid (they are actually very irregular), with density $7x10^3$ $kg/m^3$ and mass $3.7x10^{15}$ kg has a gravitational attraction $10^{-3}$ that of Earth's at its surface. Orbital velocity at the surface of such a rock is only 7 m/sec, and that of a 2 km diameter comet with density $10^2$ $kg/m^3$ is about 0.1 m/sec.

These numbers are so low that it is impractical to use gravity to orbit around such a body while studying it. Surveying planetesimals requires the continuous use of thrusters. That energy requirement is a severe problem in spacecraft design, especially for comets which should be studied for a large fraction of an orbit to observe their structural and dynamical evolution as they approach the Sun.

## PRESENT SURVEY PLANS

### Configuration

The Comet Nucleus Sample Return Mission (ref. 3) is not yet well defined, but envisions a space craft of around 1000 kg (600 kG body, two 150 kg sampler/landers, and a return capsule). The samplers will return cores from the comet, while their lander will monitor the surface activity for an extended period. The objective is to return the data and some unmodified comet matrix to Earth for study.

### Procedure

The exploration strategy after rendezvous is: global characterization from a distance of 200 km for six days, one day to transfer to a 50 km distance above a specific surface for high resolution pictures, then transfer to 100 km distance to wait for landing instructions. On a landing command, move in to 10 km, wait until the proper surface is underneath, and use thrusters to maintain the spacecraft in a forced synchronous orbit with that patch of ground. From this position, the spacecraft launches the sampler/lander and takes high resolution pictures. After recovery of the sampler, the spacecraft retreats to 100 km to await decision on a second landing site.

### Limitations

The constant maneuvering described above is made necessary by the dust and gas hazards near an active comet nucleus. "To provide the additional thrust required for the 'forced-synchronous' orbit, it will almost certainly be necessary to provide an auxiliary chemical propulsion system on the spacecraft." This propulsion system does not yet exist, and is one of the critical technologies required to enable CNSR (ref. 4).

# TETHERED ORBIT SURVEY

## Configuration

A radical approach to this maneuvering problem is to divide the spacecraft into three parts, tie them together with thin fibers, and spin them. Once this tripartite spacecraft is spinning fast enough for stability, no further energy is required to maintain their orbit. It can be maneuvered over the target comet and monitor it from all sides for extended periods with no further expenditure of maneuvering energy.

These three parts need not be the same mass or have the same instruments; a configuration consisting of two module types is

A — overall control, Earth communications, return pod, maneuvering thruster, attitude jets, spin-up thruster
B — local control, sampler/lander, photographic equipment, local communications equipment, attitude jets, spin-up thruster
B' — local control, sampler/lander, photographic equipment, local communications equipment, attitude jets, spin-up thruster

While the spacecraft is together, the attitude jets would combine to give 6 orthogonal pairs of orienting jets. When separate, the pair of jets on each unit would orient the spin-up thruster on each unit as required for spin-up, spin-down, or transverse maneuvering.

The units would be connected with high strength fibers. Studies of tethered probes for low earth orbit or space shuttle experiments have suggested a variety of materials. Kevlar 29 have been suggested by a number of researchers both bare (ref. 5), and metal coated for protection and communication purposes (ref. 6). A polyethylene (Spectra 1000) was preferred by one for its temperature stability(ref. 7). 100 kg of Spectra 1000 would be sufficient to provide three 100 km tethers with 850 N breaking strength. That is orders of magnitude greater than needed for this system. Orbiting 1/hr with a tether length of 100 km (an altitude of 57.7 km) requires an acceleration $< 0.02$ N, or .002 of Earth's gravity. If each part of the spacecraft weighs 500 kg that only requires a tension of 0.5 N in the 100 km fibers. The fibers have a strength of $3\text{x}10^9\ \text{N/m}^2 = 30\ \text{N/(0.1mm)}^2$. This 0.1 mm dia fiber has a safety margin of 100 X! Three 100 km lengths weigh only 3 kg (polyethylene). A thicker fiber may be desirable to limit stretchiness. These fibers have a tensile modulus of $1.7\text{x}10^{11}\ \text{N/m}^2 = 1.7\text{x}10^3\ \text{N/(0.1mm)}^2$, so a 0.5 N force stretches the fiber by .03%, or 30 m in 100 km.

## Procedure

This tripartite spacecraft would be launched and proceed to the rendezvous as a single unit. On rendezvous with the comet, the parts would spin up to a low speed and then undock so that the centrifugal force would help control the attitude of each unit. The attitude jets on each unit would align the thrusters in the plane of rotation and they would then spin-up the system further while the reeling mechanisms paid out line until they reached the desired configuration — 100 km separation rotating once per hour, for instance. To limit bouncing, the thrusters and reels would combine to maintain a constant tension on the lengthening tether. The reels don't need any significant power for this maneuver. Once the desired condition is reached, the attitude jets reorient the thrusters so they can maneuver the spinning system around the comet. They can then study the comet indefinitely without using thruster power.

If a closer look is desired, the thrusters are aligned in the plane of rotation again, but opposite to the rotation direction, and the thrusters are fired to spin-down while the reels take in tether; again, the two actions combined to maintain a constant tension on the tether. In this case, the reels need power.

If a sample is required, one of the sampler/landers is launched from as close to the surface as possible. It is not necessary to remain in forced synchronous orbit because one of the three nodes will always be visible to and in communication with the lander. The system will orbit asymmetrically because of the weight loss (about 150 kg out of 500 kg); the lighter module farther from the center of rotation. On recovery of the sampler, the weights and hence the orbiting configuration will change again.

When the mission is over, the thrusters are aligned to spin-down and the tether is reeled in. Collected samples are transferred to the return pod, and then the A module separates from the other two modules and returns home.

## Dynamics

Algorithms for stably paying out tether have been investigated for leashed experiments in Earth orbit (ref. 8), though only up to lengths of about 20 km. That dynamics problem is difficult because leashed experiments have to come to the end of their tether with zero velocity and acceleration simultaneously, or the experiment slowly bounces all the way back to the experimenter. It is not so hard for a rotation triangular system. We have set up a simple simulation program to model its kinematic behavior. We found it very stable if, during spin-up, one lets the fiber pay out at a rate so that the tension in the tethers remains about constant. Errors only cause some jello-like wobbling around the equilibrium triangle. One prevents that by turning on and off the thrusters slowly relative to the time required for a wave to travel the length of the tether. That time seems to be a few minutes for 100 km tethers(ref. 9).

Release and recapture of the sampler/lander modules causes an abrupt change in the mass of the modules, and could cause instability problems. Our simulation shows the need to use thrusters in a radial direction to compensate for mass loss and to smooth out the transition.

## Energetics

For stated conditions, need to accelerate 1000 kg to about 300 km/hr. That is 1/2 x $10^7$ N-sec per spin-up. If spin-up and spin-down are done maintaining a constant tension in the tether, the reels would require no net power; they would be electric generators during spin-up and use their stored energy to reel in the tether during spin-down. Motor, generator, and battery inefficiencies would cause some net power drain. That power could be provided by solar cells.

## Limitations

Tether malfunction would abort the mission. Impact by a micrometeoroid, breaking a tether is the most likely cause, but the probability may be hard to estimate; there is only sparse data on the micrometeoroid density outside the Earth's moon orbit. Making the tether of a braided tow reduces that problem (ref. 10). There is also substantial ionized gas — presumably corrosive, in the vicinity of comet which could weaken the tethers.

Mechanical problem in the reels would also be fatal. Designs have been studied for and will be tested in low Earth orbit (ref. 11).

## SUMMARY

We have suggested a tripartite spacecraft connected by long lightweight tethers as an alternate approach to surveying microgravity planetesimals — comets and asteroids. The mass of the tethers is an small fraction of the total spacecraft mass. A configuration and exploration plan have been sketched out. This configuration is stable during spin up and spin down operations, but there may be problems stabilizing it during release and recapture of sampler/lander modules. In addition, there may be a problem insuring that the tethers are not broken by micrometeoroids.

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N91-22165

# ON SPACE-BASED S.E.T.I.

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## ABSTRACT

Space-based antenna systems for the search of signals from extra-terrestrial intelligence were first proposed in the mid-seventies. Tentative performance specifications for systems that might be built in the early part of the twenty-first century were established. Preliminary studies of system design and mission profile led to the consideration of the Sun-Earth collinear transterrestrial libration point (SEL2) as the ideal operational location for the system. Moreover, compatibility with current contingency plans for technology development in the geostationary orbit suggests that fabrication and assembly of major components of space-based antenna systems that might be built in the early part of the next century would most likely take place in that orbit. Consequently, deployment of these components to the operational location at SEL2 would require, first of all, an orbit transfer at geosynchronous altitude to the ecliptic plane, and then a transfer from the resulting ecliptic geosynchronous departure orbit to the heliocentric operational orbit at SEL2. Because both major components (dish and shield) of the antenna system would have structural configurations (of the shallow-shell and flat-plate type, respectively) that are associated with high values of the area-to-mass ratio, the possibility of component deployment in the ecliptic plane by means of solar sailing was early recognized. A literature review of papers and other publications dealing with Earth-orbiting and interplanetary solar-sailing missions was then conducted; however, relevant information concerning suitable attitude-control laws for solar-sailing flight to SEL2, and associated flight times, could not be obtained. For this reason, independent studies of the ecliptic solar-sailing transfer problem from the geosynchronous departure orbit to SEL2 have been conducted in recent years in the Department of Mechanical Engineering of the University of Hawaii at Manoa. These studies were based on a relatively simple mathematical model describing attitude-controlled spacecraft motion in the ecliptic plane as governed by solar and terrestrial gravitational attractions together with the solar radiation pressure. The resulting equations of motion have been integrated numerically for a relevant range of values of spacecraft area-to-mass ratio (as obtained from preliminary estimates for dish and shield established in earlier work) and for an appropriate spacecraft attitude-control law known to lead to Earth escape (as obtained from perusal of the solar-sailing literature). Experimentation with varying initial conditions in the departure orbit, and with attitude-control law modification after having achieved Earth escape, has established the feasibility of component deployment by means of solar sailing. Details are given in the paper.

# ON SPACE-BASED S.E.T.I.

## Space-Based Antenna Systems for the Search of Signals from Extraterrestrial Intelligence

### INTRODUCTION

Space-based antenna systems for the search of signals from extraterrestrial intelligence were first proposed in the mid-seventies [1,2]. A study to determine their comparative cost-effectiveness relative to Earth- and Moon-based systems was conducted by the Stanford Research Institute (now SRI International) on behalf of the SETI Program Office of the NASA Ames Research Center at that time [3]. It was concluded that, for the very large antenna systems that would be required for small assumed values of the number of transmitting civilizations, an appropriate base location in space might well turn out to be cost-effective. For this reason, tentative performance specifications for SETI space-based radiotelescopes were established at Ames during that period [4]. Their authors envisaged a three-stage development program resulting in an ultimate antenna system the receiving element of which would consist of a 3 km aperture spherical reflector dish having a surface accuracy of ± 1 mm and a nominal maximum operating frequency of 15 GHz. Other components would include up to three free-floating feed modules equipped with sophisticated laser-ranging and attitude-control systems, as well as a 6 km diameter RFI (radio-frequency interference) shield for protection against Earth- or Earth-orbit based electromagnetic emissions. Maximum sky coverage for a minimum of dish movement would be obtained by moving the feed modules across the reflector dish. The system would operate in lunar orbit at one of the two Earth-Moon equilateral libration points. It would always be pointing away from the Earth, thereby sweeping 360 degrees of sky once each lunar period.

### OPERATIONAL ORBIT

More recent studies of the SETI space-based antenna system have tended to invalidate the concept of a lunar operational orbit [5]. At the lunar distance and beyond, thermal gradients together with the solar radiation pressure would be the dominant environmental factors influencing the structural response of Earth-orbiting radiotelescopes of the size envisaged. It was postulated that unattenuated thermal loads on the reflector dish structure would so much degrade the shape accuracy of the reflecting surface that ensuing demands on the control system required for maintaining this accuracy would become excessive. Consequently, protection of the reflector dish structure against the solar thermal radiation would also be required. For operation in the lunar orbit this would imply a requirement of two shields: one located between the dish and Earth providing RFI protection, and one located between the dish and the Sun providing thermal protection. The dynamic complexity of a two-shield orbiting antenna system was considered so undesirable that a search was initiated for locations in the solar system at which a single shield would simultaneously and continuously provide both RFI and thermal protection for the dish. Elementary considerations suggest that the only location for which this would be the case lies on the Sun-Earth line at a distance from Earth compatible with heliocentrical orbital motion in phase with it. Thus, the optimal location in the solar system for a SETI space-based antenna system as envisaged appears to be the collinear transterrestrial Sun-Earth libration point (here called SEL2). Even though this location, at 1,500,000 km away from Earth, is known to be dynamically unstable, it would be an excellent operational location for the following reasons: a single shield there interposed between the dish and Earth would provide almost continuous and complete RFI as well as thermal protection (the exception consisting of occasional interference from emissions generated on the outer planets), power requirements for system station-keeping at the optimum location along the Sun-Earth line would be minimal [6], and a direct line of communication with Earth could always be maintained.

COMPONENT DEPLOYMENT It is clear that the enormous size of the ultimate antenna system envisaged in the Ames specifications would make it necessary to plan for fabrication and assembly of its major components (the RFI/thermal shield and reflector dish structures) in the highest possible Earth orbit. The higher this orbit, the smaller the gravity-gradient torques to which these structures would be exposed during fabrication and assembly; and with gravitational loads negligible at the operational location (SEL2), the optimum location for fabrication and assembly would seem to be situated at an orbital altitude where the solar radiation pressure loads on shield and dish are dominant. Although the lowest such altitudes lie well below SEL2, they are also situated at much higher locations than those for which construction facilities and personnel may be expected to become available in the foreseeable future. Recent work in preliminary technology planning for space solar power stations postulates that the large structural components of these stations would be fabricated and assembled in the geostationary orbit; a concept that envisages the availability of construction facilities and personnel at 36,000 km above the Earth during the early part of the next century. It has therefore been assumed that fabrication and assembly of the major structural components of a SETI space-based antenna system that might be built at some time during the next century would take place in the geostationary orbit. Consequently, deployment of these components to the operational location at SEL2 would require, first of all, an orbit transfer (at geosynchronous altitude) to the ecliptic plane, and then a transfer from the resulting ecliptic geosynchronous departure orbit to the heliocentric operational orbit at SEL2. Because the two largest components (shield and dish) would have structural configurations (of the flat-plate and shallow-shell type, respectively) that are associated with high values of the area-to-mass ratio, and because the ecliptic transfer problem would involve regions of space in which the influence of the solar radiation pressure on component motion would be dominant, the possibility of deployment of shield and dish by means of solar sailing was early recognized. A literature review of papers dealing with Earth-orbiting and interplanetary solar-sailing missions was then conducted; however, relevant information concerning suitable attitude-control laws for solar-sailing flight to SEL2, and corresponding durations of transfer, could not be obtained. For this reason, independent studies of the component deployment problem have been conducted in recent years in the Department of Mechanical Engineering of the University of Hawaii at Manoa [7,8,9]. These studies were based on a relatively simple mathematical model describing the attitude-controlled motion of a solar sail (representing shield or dish) in the ecliptic plane as governed by both solar and terrestrial gravitational attractions together with the solar radiation pressure. The resulting equations of motion were integrated numerically for a relevant range of values of component area-to-mass ratio (as obtained from preliminary mass estimates for shield and dish established in earlier work) and for an appropriate attitude-control law known to lead to Earth escape (as obtained from the solar-sailing literature). Numerical experimentation with varying initial conditions in the geosynchronous departure orbit, and with modifications of the attitude-control law (including free flight) after having achieved Earth escape, has established the feasibility of component deployment to SEL2 by means of solar sailing. This being the case, the principal focus of current work is on a broader definition and more exact solution of the solar-sailing transfer problem, including the case of asymptotic arrival at SEL2 as well as the case of smooth insertion into an appropriate halo orbit in its vicinity. Problems concerning the conceptual design and nominal operation of a fully deployed SETI space-based antenna system at or near SEL2 are also being considered.

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N91-22166

# SPACEPORT OPERATIONS FOR DEEP SPACE MISSIONS

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Space Station Freedom is designed with the capability to cost-effectively evolve into a transportation node which can support manned lunar and Mars missions. To extend a permanent human presence to the outer planets (moon outposts) and to nearby star systems, additional orbiting space infrastructure and great advances in propulsion systems and other technology will be required. To identify primary operations and management requirements for these deep space missions, an interstellar design concept was developed and analyzed. The assembly, test, servicing, logistics resupply and increment management techniques anticipated for lunar and Mars missions appear to provide a pattern which can be extended in an analogous manner to deep space missions. A long range, space infrastructure development plan (encompassing deep space missions) coupled with energetic, breakthrough level propulsion research should be initiated now to assist us in making the best budget and schedule decisions.

## INTRODUCTION

Sometime during the next 50 years, interplanetary flights between Space Station Freedom (Spaceport Earth) and Mars Outpost 1 will be established on a continuous and permanent basis. Manned exploration missions to the outer planets and moons will have been planned and initiated. Systems required for advanced space transportation and associated infrastructure will be researched, tested, checked out and serviced at or near Freedom. The implementation of the Space Exploration Initiative and associated U.S. space policy will require that a long range, propulsion R. & D. plan be initiated to provide assured interplanetary space transportation. Once we establish a permanent outpost or colony on Mars, our commitment to energetic, long-range technology R. & D. and the maintenance and improvement of orbiting space infrastructure will no longer be optional, it will be mandatory.

To make human transportation to the outer planets "practical" on a continual basis will require propulsion systems with the capability to reduce one way trip times to a couple years or less. Candidate propulsion systems include nuclear thermal and matter/anti-matter propulsion. To conduct manned interstellar missions which have meaning and value to an emerging space civilization (and which therefore can be economically justified) will require propulsion breakthroughs which "effectively" allow a spacecraft to exceed the speed of light. New and innovative research and development approaches are needed to develop interstellar transport capability or capabilities.

A conceptual design for a manned interstellar transport has been developed to assist in identifying spaceport infrastructure and operations requirements for the research and development, assembly and checkout, performance tests and trial runs of the manned interstellar transport. Advanced robotics, high temperature superconductor shielding and microengineered materials may play important roles in minimizing risks associated with the assembly and servicing of the propulsion systems. To enable routine, outer planet and interstellar transportation will likely require additional remote, orbiting space infrastructure elements and outer planet moon outposts with the capability to support orbital servicing. Increment management considerations for Space Station Freedom and the interstellar spacecraft assembly and test suggest the types of generic and specialized outfitting of space infrastructure which will be required (see Figure 1).

## DEEP SPACE MISSION PROPULSION REQUIREMENTS

The propulsion requirements to explore and establish an outpost on Mars are well within the range of our current technology capabilities. To reduce trip times to Mars and to enhance the exploration of the Martian surface

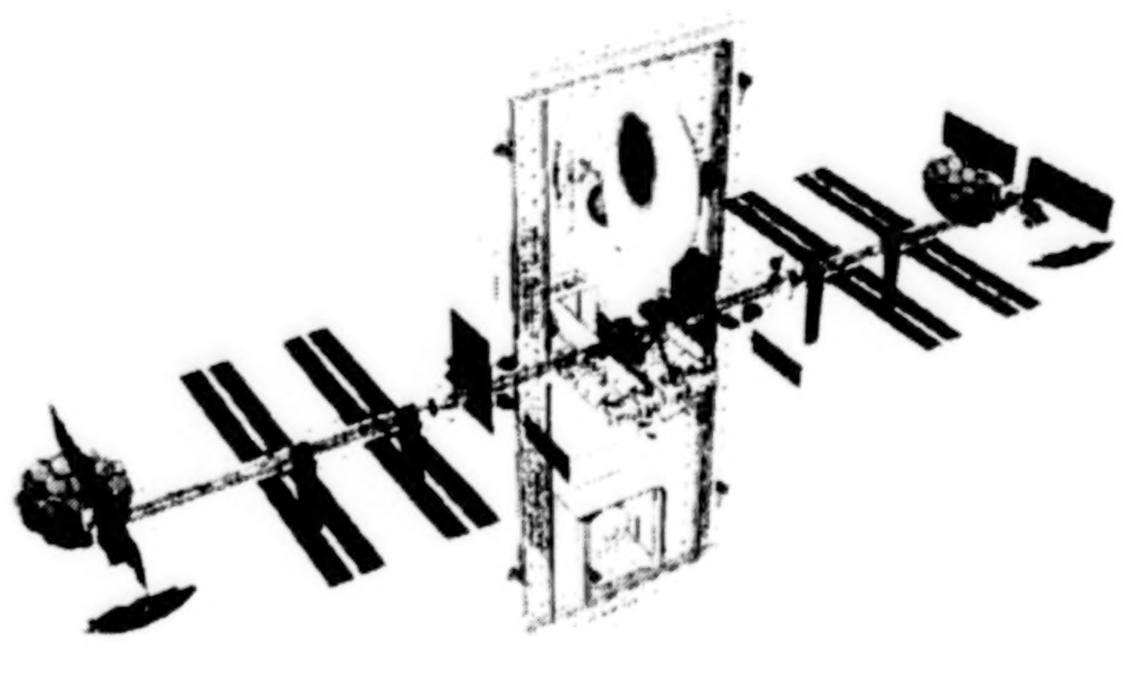

Figure 1. Space Station Freedom: Transportation Node Mars Vehicle Checkout

could require the development (or improvement) of nuclear thermal, nuclear electric and electromagnetic propulsion. These enhancements would also set the stage for the development of the advanced propulsion systems needed for manned exploration of the outer planets and moons.

It can be argued that with the development of advanced radiation protection systems and a greater understanding of human space physiology, one way trip times of 5 to 10 years to the outer planets is feasible. However, to establish outposts on the moons of outer planets which can be justified economically and sociologically, substantial advances in propulsion technology are required. Propulsion technology with high specific impulses such as nuclear fusion thermal/electric and matter/anti-matter systems may be sufficient to meet this challenge. These outposts and the associated transportation logistics support activity will provide the experience and space infrastructure needed to initiate manned interstellar missions.

Some imaginative concepts (limited by the constraints of space-time, the speed of light) have been proposed for interstellar spacecraft which can be used to demonstrate that such missions are feasible with our current technical understanding. Unmanned interstellar probes should be studied with these concepts in mind. The trip times for manned exploration, however, represent a lifetime commitment even for the nearest stars. Even taking advantage of the time dilation factor of General Relativity, round trip times of 40 years or more would be required. It is difficult to see how support for such a mission could be generated.

If a means could be found to work around the constraints of space-time (to "effectively" exceed the speed of light), then the picture could change dramatically. The detection of planets around other star systems with the prospect for the existence of other lifeforms, including advanced intelligent lifeforms, would be additional compelling motivation for manned interstellar exploration. Even this motivation might not be sufficient, however, if the economic base is not exceptionally strong.

The economics of such endeavors can be dealt with if one of the following scenarios exists:

(1) Advanced, faster than light propulsion systems are developed which enable round trip times of less than 5 years.

  - Economics associated with this trip are then no more formidable than those associated with outer planet exploration.

(2) Human civilization in the Solar System is threatened and options for new homes must be found.

  - Economics are overridden by the motivation for survival.

(3) Economic justification is based on economic and technology development gains resulting from Solar System exploration and colonization and known potential for similar or greater gains in interstellar expeditions.

  - Preliminary contact with other civilizations could also play into this scenario (SETI and other programs).

  - Faster than light systems are likely still to be enabling in this scenario.

Thus in 2 out of the 3 scenarios, faster than light propulsion is enabling for a manned interstellar mission. Only the colony ship which is not concerned with maintaining contact with the civilization left behind would likely justify slower than light propulsion options.

Science fiction writers have for many years discussed faster than light propulsion systems. Realistic research and development approaches to develop such systems have been suggested from time to time. Based on the recent developments in technologies and materials such as high temperature superconductors, microengineered materials, compact superconducting magnets, free-electron lasers, high power, tunable microwave systems (masers), quasi-crystals, plasmoid generators, etc. and theoretical developments associated with the superstring multi-dimensional theory and other similar theories, we are poised to begin serious pursuit of faster than light propulsion systems.

Taking all these factors in account the design reference concept developed for the interstellar mission assumes the availability of faster than light propulsion systems. The hazards of such systems do not appear to be any more challenging than nuclear thermal propulsion systems. Faster than light propulsion systems could ultimately reduce space infrastructure needs, but the availability of such systems would not dramatically affect current and projected Solar System orbiting infrastructure requirements (which includes support for unmanned sub-light interstellar probes).

## INTERSTELLAR TRANSPORT DESIGN CONCEPT

The interstellar transport concept developed for this analysis was based primarily on the following considerations:

- Modular design which can readily accommodate the removal and addition of elements during and after the design process.

- Adequate shielding of crews from nuclear power system and nuclear thermal, anti-matter/matter and space-time disengagement propulsion systems.

- Shielding and boundary constraints associated with radiation protection, micrometeoroid protection, and space-time disengagement system.

- Unique envelope configurations requirements associated with pulsed magnetic field vernier system and disengagement system.

These factors resulted in a compromise configuration which resulted in a reduction in the ease of module (and Orbital Replacement Unit) replacement. Multiple propulsion systems were selected for different mission phases and redundancy purposes.

Nuclear power sources are the highest density/unit mass power sources currently available and thus are at this point mandatory for mission success. Nuclear fusion has much cleaner products of reaction which would enable some hands-on maintenance activity.

A nuclear fusion thermal propulsion system was selected for interplanetary travel and as an option for intermittent use with the space-time disengagement system(ref 16). The system heats up hydrogen which is then expelled as a propellant through 8 thruster nozzles. The nuclear fusion propulsion systems requires 1.5 gigawatts of sustained reactor power and 2.5 gigawatts of peak power achieved utilizing MHD superconducting peak power modules. The 8 thrusters provide a variable thrust of 50,000 to 300,000 lbs (see Table 1).

| Advanced Propulsion System Type | Translation Interaction | Effective Specific Impulse (Isp) | Thruster/ Propulsion Elements | Effective Thrust | Power Requirements (Avg/Peak Gigawatts) |
|---|---|---|---|---|---|
| VERNIER/Orbit Transfer | | | | | |
| (1a) Electro-plasma | High velocity ions<br>– Action/reaction effect | 5,000 - 10,000 | 12 | 0.5 - 40 lbs./thruster (up to 20 thrusters) | 0.2/0.5 |
| (1b) Pulsed Magnetic Field with pulsed plasmoid generators | Reaction against background magnetic field and plasma | $1 \times 10^{6}$ | 4-Plasmoid<br>16-Pulsed * Field | Dependent on Background Field (Dual Redundant) | 0.2/0.5 |
| INTERPLANETARY | | | | | |
| (2a) Nuclear Fusion | Heated/ionized gas<br>– Action/reaction effect | 1000 - 3500 | 8 | 50,000 - 300,000 lbs. (Quad Redundant) | 1.5/2.5 (Reactor Power Level) |
| (2b) Matter/Anti-Matter | Heated/ionized gas<br>– Action/reaction effect | 1000 - 5000 | 8 | 50,000 - 400,000 lbs. | 0.2/0.5 |
| INTERSTELLAR | | | | | |
| (3) Space-Time Field Disengagement (STFD) – with Field bias | Space-time bubble created<br>– Relocated to space-time position which balances out bias field | $1 \times 10^{12}$ | 16* | N/A | 0.5/0.7 |

* Share use of these elements

Table 1. Summary of Advanced Space Propulsion Systems

Coupled with the nuclear thermal system is a nuclear electric-plasma distributed thruster propulsion system which uses high energy electrical currents to heat the hydrogen gas with much higher Isp but lower thrust (ref. 2,3). The electric-plasma distributed thruster system requires an average of 0.5 gigawatts of electrical power. It generates thrusts from 2.0 to 40 lbs. with an Isp in the range 5000 to 10000.

As a backup system or higher performance special use system a matter/anti-matter propulsion capability has been included (ref 20). It would use many of the same components and the same thrusters used for the nuclear fusion propulsion system. The antimater pellets would be used as an alternative or backup to the nuclear power heat source for propulsion using auxillary magnetically confined storage and reaction chambers. The pellets could also be used as part of the fusion generation system itself (ref 16).

For vernier and orbit transfer propulsion the electro-plasma distributed thruster system or the pulsed magnetic field interaction system is used. Both systems have very high Isp's which is critical for an interstellar spacecraft. The pulsed magnetic field system requires 200 to 500 megawatt bursts of power to obtain a near-infinite Isp from its 4 primary (pulsed plasmoid mode, ref 5) and 16 secondary pulsed field sources (ref. 3). The maximum thrust for the pulsed magnetic field system varies depending on the background field and plasma densities. The system uses high temperature superconducting components.

For deep space propulsion, primarily for interstellar travel, a system which is capable of disconnecting or disengaging the spacecraft from the velocity constraints of space-time is used. This system is assumed to consist of two field generation systems. Each system has 16 field amplifiers or disengagers distributed along a critical boundary surface. These amplifiers have an operational mode which also allows them to be used in conjunction with the pulsed magnetic field system. The system can be viewed as forming a space-time bubble around the spacecraft. The bubble provides a natural and near perfect protection against radiation, micrometeoroids and any stray directed energy.

After the bubble has been formed, the spacecraft can be accelerated by biasing the bubble field or through a resonance interaction between the pulsed magnetic field and space-time fields. The disengagement system requires a sustained power of 0.5 gigawatts with 0.7 gigawatts of peak electrical power.

It is also possible to use the disengagement system in a sub-light mode alternating between nuclear or matter/antimatter propulsion and disengagement phases. This mode allows the protection capabilities to be used even when hyperlight velocities are not required or desired. This system also takes advantage of high temperature superconductors and materials sensitive to nuclear spin alignments and transitions.

Figure 2 depicts a conceptual layout of the interstellar spacecraft along with the multiple propulsion systems. The total mass of the vehicle is estimated to be 500,000 lbs. with a total internal usable volume of 5 million cu. ft. for habitability and mission elements. The length of the vehicle is 250 feet with a width of 200 feet. The habitable modules and storage provisions can accommodate a crew size of 24. The nuclear fusion power system is a quad redundant system with each system capable of generating 2.5 gigawatts of power. Peak power is normally limited to 3.5 gigawatts using superconducting MHD storage modules. Thermal rejection is accomplished through the use of coherent IR radiators and structurally integrated micro-radiators.

The primary structures of the spacecraft are aluminum-silicon and aluminum-lithium with integrated, micro-engineered sensors and data flow channels. A closed ECLSS system is complemented by a close-cycle greenhouse. Communications consist of multiple wavelength laser systems, a high power microwave system, and advanced experimental units which are designed to work during the disengagement phases. Extensive use is made of internal robotics systems and artificial intelligence (neural network) architectures. All systems as well as structures have pre-integrated or microengineered sensors. External robotics can be deployed with freeflying or "crawling" capabilities.

A wide variety of elements and functions will have to be supported by the interstellar spacecraft (ref 13). These elements are modular to the extent that the demands of propulsion and protection systems allow. They include: (1) four habitable modules (crew quarters), (2) two systems management modules, (3) proximity operations/training module (with cupolas/wall imagery), (4) three storage modules (oversized), (5) field induced simulated gravity system (?)-a by-product of disengagement system, (6) four excursion/landing vehicles (with integrated simulation capability), (7) two unmanned transport vehicles for surface logistics and outpost establishment, (8) vehicle servicing, maintenance and training facility, (9) vehicle refueling facility (hazardous processing facility), (10) external ORU maintenance and repair station (deployable), (11) deployable or built-in technology test and science experiment facilities, (12) multiple mini-labs/facilities – medical/life science lab, ORU maintenance/diagnostic facility, food production lab, data/comm center/library facility, materials repair, development and production facility, two bio-isolation labs, and data distribution center, (13) exercise and recreation module, and (14) two general purpose science and technology experiment labs.

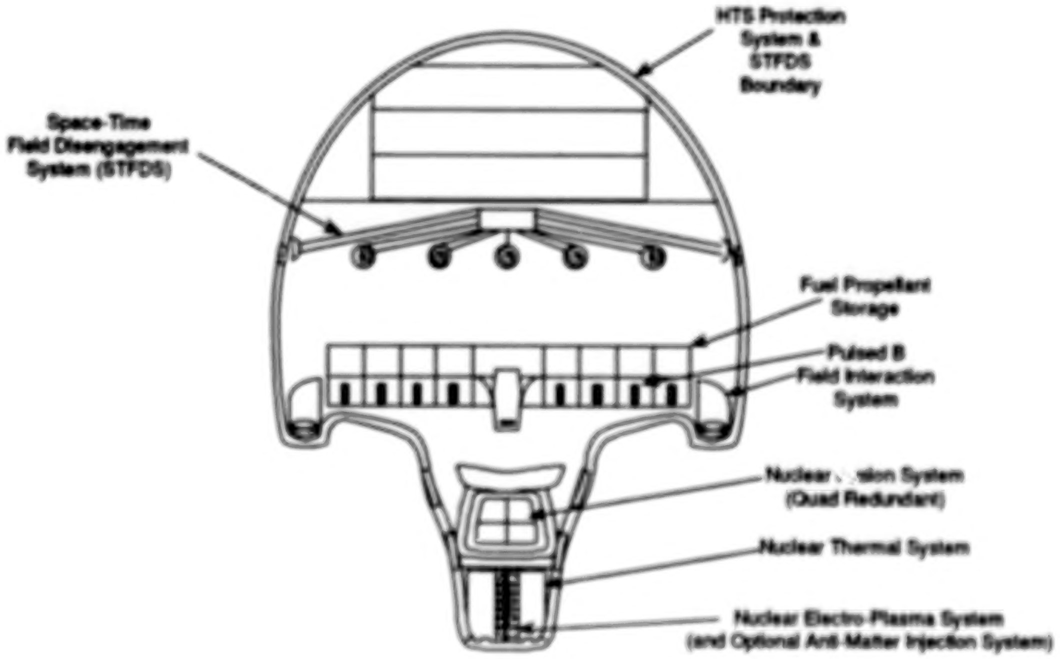
HTS Protection System & STFDS Boundary
Space-Time Field Disengagement System (STFDS)
Fuel Propellant Storage
Pulsed B Field Interaction System
Nuclear Fusion System (Quad Redundant)
Nuclear Thermal System
Nuclear Electro-Plasma System (and Optional Anti-Matter Injection System)

Figure 2. Conceptual Layout of Interstellar Transport Systems

Proximity operations capabilities include (ref. 4): (1) active and passive vehicle rendezvous and docking capabilities, (2) multiple vehicle, robotic systems and manned maneuvering unit, simultaneous tracking and guidance systems, (4) magnetic field/plasma "tractor" beam system, (5) remote power transfer capability – bi-directional, maser and laser.

Protection systems include (ref. 5): (1) infrared and active meteoroid scanning, detection and tracking system, (2) magnetic field/superconducting surfaces for radiation, ion and micro-meteoroid protection (including microengineered composites), (3) high energy particle and cosmic ray scanning, tracking and identification system, (4) Stellar, planetary and interstellar cloud (dust and gas) imaging systems – multiple wavelengths, (5) magnetic and gravitational field anomaly detection systems, (6) laser/microwave dispersal systems. As indicated earlier while the disengagement system is in operation very effective protection against radiation and meteoroids is provided.

The 24 crewpersons conduct three shift, round-the-clock operations with the exception of two shift operations for two days every 5 days. This break in operations insures that each crewperson gets one day completely off each week and one day with limited duty. During the mission a substantial crew training capability will be utilized to get the crew trained to meet contingencies and upcoming mission phases. The onboard training will include (ref. 6): (1) in-orbit systems normal and malfunction operations, (2) excursion/landing vehicle systems and subsystems training and simulations, (3) outpost setup training, (4) experimental research preparation, (5) individual study.

## INTERSTELLAR SPACECRAFT ASSEMBLY AND CHECKOUT REQUIREMENTS

The development, assembly and checkout of an interstellar transport can be divided into the following phases:

(1) Technology Research and Demonstrations (A)

(2) Transport Subsystems Tests (A)

(3) Transport Assembly and Interface Verification (A)/(B)

(4) Transport Final Subsystems and Low Power Performance Tests (B)/(C)

(5) Local and Distant Full Performance Test Runs (C)/(D)/(E)

(6) Crew and Cargo Transfer and Servicing (A)/(D)/(E)

The letters listed behind each phase refer to Figure 3 and the selected sites for the conduct or implementation of these phases. Space Station Freedom (Spaceport Earth) will still function as a major orbital technology/research and demonstration platform (ref. 7,8,17). Technology which requires the extended plasma environment and vacuum of space for testing can be mounted on the external truss network of this transportation node (See Figure 1). The availability of extensive robotics and crewpersons provides the capability to schedule technology tests when operational impacts are a minimum. Priorities can be shifted and environmental disturbances can be accommodated which might be prohibited or much more expensive and time-consuming to carry out on other unmanned or distant spaceports. Nevertheless, safety considerations will require that and some advanced propulsion tests, for example, would have to be conducted on unmanned co-orbiting platforms (see Table 2).

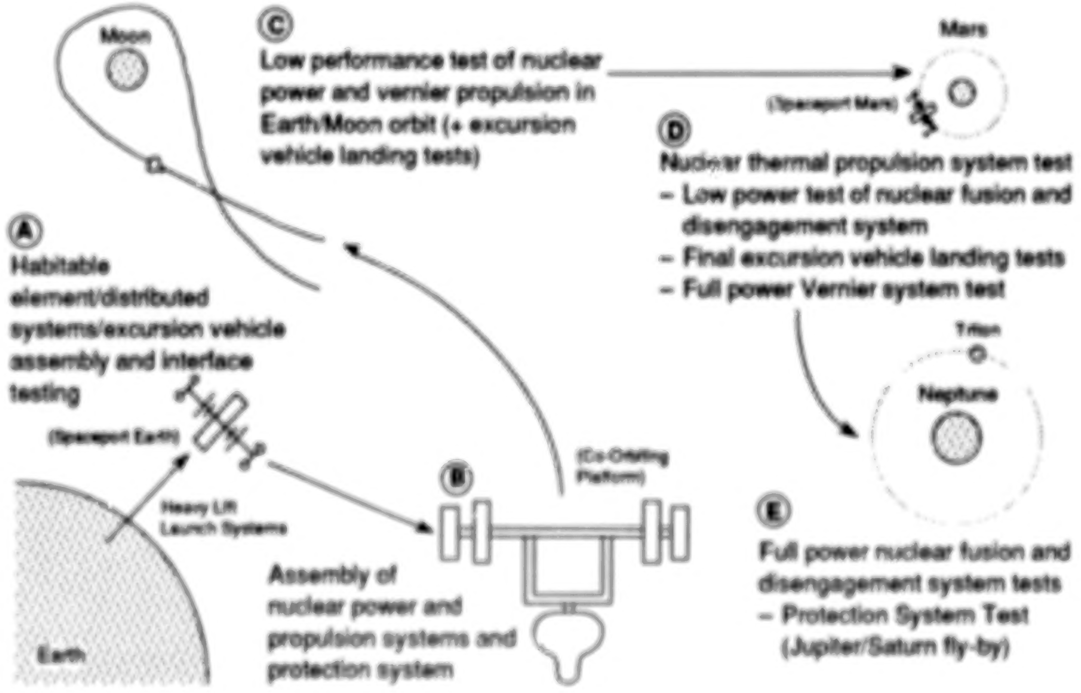
Moon
Ⓒ
Low performance test of nuclear power and vernier propulsion in Earth/Moon orbit (+ excursion vehicle landing tests)
Mars
(Spaceport Mars)
Ⓓ
Nuclear thermal propulsion system test
– Low power test of nuclear fusion and disengagement system
– Final excursion vehicle landing tests
– Full power Vernier system test
Ⓐ
Habitable element/distributed systems/excursion vehicle assembly and interface testing
(Spaceport Earth)
Triton
Neptune
(Co-Orbiting Platform)
Ⓑ
Heavy Lift Launch Systems
Ⓔ
Full power nuclear fusion and disengagement system tests
– Protection System Test (Jupiter/Saturn fly-by)
Assembly of nuclear power and propulsion systems and protection system
Earth

Figure 3. Assembly and Test Concept for Interstellar Spacecraft

| Item | Assembly Location | Checkout/Interface Location | Performance Test Location | Robotics | | Crew | |
|---|---|---|---|---|---|---|---|
| | | | | Telerobotic Mobile | Freeflying | EVA | IVA |
| Habitable Elements | SSF | SSF | SSF | X | | X | X |
| ECLS, DMS & Utility Distribution Systems | SSF | SSF | SSF | X | | X | X |
| Landing/Excursion Craft (Dry) | SSF | SSF | Moon/Mars | X | X | X | X |
| Storage - Crew, ECLSS | SSF | SSF | N/A | X | | X | X |
| Propellant Storage | SSF | SSF | SSF/COP | X | | X | X |
| Special Modules | SSF | SSF/COP | SSF/COP | X | | X | X |
| Nuclear Fusion Power System | SSF/COP | COP/Moon | Moon/Mars | X | X | | X |
| Propulsion Systems (Vernier) | | | | | | | |
| – Pulsed Ion Thruster (1a) | SSF | SSF | Moon/Mars | X | | X | X |
| – Pulsed B Field Interaction System (1b) | SSF | SSF | COP/Moon | X | | X | X |
| Propulsion Systems (Planetary) | | | | | | | |
| – Nuclear Fusion Thermal (2a) | SSF/COP | COP | Moon/Mars | X | X | | X |
| – Matter/Anti-Matter (2b) | SSF/COP | COP/Moon | Moon/Mars | X | X | | X |
| Propulsion Systems (Interstellar) | | | | | | | |
| – Space-Time Field Disengagement with 2a (3a) | SSF/COP | SSF/COP | Mars/Neptune | X | | X | X |
| – STFD with 2b (3b) | SSF/COP | SSF/COP | Mars/Neptune | X | X | X | X |
| – STFD with STF Bias (3c) | SSF/COP | SSF/COP | Mars/Neptune | X | | X | X |

Table 2. Spacecraft Assembly and Checkout Requirements and Implementation Approaches

Some of the on-orbit interface testing can be accomplished at the subsystem level on Space Station Freedom (SSF)(ref. 18). Transport subsystems such as the data handling system, the environmental control life support system, communication subsystem and various module subsystems would be checked out and tested while attached to Space Station Freedom. The "active components" of the nuclear power system and matter/anti-matter propulsion system are hazardous and should be installed and checked out on a co-orbiting platform (ref. 1,15,19). Assembly of the major elements of the spacecraft would be easier to accomplish at the co-orbiting platform assuming that element subsystem in-orbit tests are accomplished on the station.

The interface testing for the overall spacecraft would be conducted at the co-orbiting platform (COP) which stays close enough to SSF to allow daily and extensive crew and robotics visits to the COP (ref 12). System power and data end-to-end checks would be conducted using the COP's power capabilities. Low power performance tests of portions of the nuclear and propulsion subsystems would also be conducted. The reactor would not be activated but the vernier and electro-plasma propulsion capabilities could be checked out using COP power input.

Once the reliability of the interstellar transport's propulsion capability is verified and a lunar transfer tug is available to escort the transport into a lunar/high Earth orbit, the transport can be decoupled from the COP. A final systems test will be conducted with a minimum checkout crew onboard. The transport's electro-plasma propulsion system will then be initiated to put the interstellar transport into a lunar transfer orbit or the transfer tag could be used to accomplish the orbit insertion. After the successful completion of this phase, the nuclear reactor will be started and checked in a low performance mode. Depending on checkout requirements and the number of problems which develop, one or more orbits of the moon will be required before the transport accepts the remainder of the crew.

At this point the transport will begin a lengthy duration test of all major systems to expose any system faults and gain confidence in the overall integrity of the vehicle. A lunar transfer tug will remain on standby during this period to handle any emergencies. In addition, two of the landing vehicles carried by the transport will be in a standby and fully checked out condition. All manned and unmanned vehicles will undergo test runs and landings on the Moon during this period. Any significant problems in these vehicles should be uncovered as a result of these tests. All of the performance tests for the landing craft will not be attempted on the Moon, however.

During this period the changeout of crews will be accomplished by regular space plane flights to SSF and subsequent transfer to the interstellar transport via lunar transfer tug. At the completion of this phase, low to moderate power tests of the nuclear power system and low power tests of the pulsed magnetic field, and nuclear electro-plasma propulsion systems will be conducted. Subsystems of the nuclear fusion thermal and matter/anti-matter systems will also be tested out using low quantities of tracer particles instead of fusion and anti-matter propulsion "pellets."

The next phase of performance testing will involve a flights to Mars, electromagnetic braking into Mars orbit and return to a high Earth orbit. During this phase low power tests of the nuclear fusion thermal propulsion system and matter/anti-matter system will be conducted. The transport will initially be injected into a free return trajectory around Mars to provide crew safety options in case of any major contingencies. Spaceport Mars will be used for station-keeping servicing and repair operations, when needed. It will be outfitted with extensive robotics for hazardous systems servicing and checkout. If the transport's propulsion systems are functioning normally, including several starts and stops, then an electromagnetic braking and transfer into a Mars orbit co-planer with Spaceport Mars will be initiated (ref. 10).

While in Martian orbit, the manned and unmanned landing vehicles will be put through all remaining performance and endurance tests. The pulsed magnetic field interaction system will be tested at full power after shifting to a higher orbit around Mars. Once these tests have been completed, the transport will return to Earth. Lunar or GEO transfer tugs will changeout the crew and resupply the transport. Any significant repairs will be done with a small servicing platform brought up to the higher orbit. Should major repairs be required, for any reason, the COP's orbit can be raised and the transport brought down to it. However, this would be considered a contingency mode since major hazardous repairs are planned for Spaceport Mars.

The final phase of performance testing of the interstellar transport is conducted on a free return trajectory around the Neptune/Triton outpost. During this trip full power, nuclear thermal and matter/anti-matter propulsion system tests are conducted. In addition, space-time disengagement system tests are conducted for short periods. If everything proceeds smoothly on the outgoing leg, the transport will enter a Neptune orbit co-planar with the orbit of Triton. While the crew has received routine examinations during the other test phases, a special exam is scheduled on Triton to monitor any irregularities associated with the disengagement tests. Backup crewmembers are available on Triton if any of the crewmembers need to be replaced.

The unmanned and manned vehicles are deployed for test runs to verify that the disengagement process has not affected any of their systems. If the disengagement tests have been successful on the way out, a flyby of Jupiter or Saturn would be conducted upon the return to further test the protection systems of the transport. In particular, to test the capabilities of the disengagement induced protection system. The full performance trip to Neptune can be repeated as often as necessary to gain confidence in the durability and reliability of the transport and its systems.

The first manned interstellar flight would also be setup on a free return trajectory around a nearby star such as Alpha Centauri A. The mission would start from a high Earth orbit with a rendezvous at Triton for a final mission readiness review prior to committing to an interstellar mission. The crew would have the option, with some guidance from mission control center on Triton, to proceed with a trip to the primary destination of Epsilon Bootes should transport performance meet pre-determined criteria. Epsilon Bootes may have a planetary system with a star similar to that of the Sun. The crew has the authority to explore the planet or planets most likely to harbor life and if advisable, establish an outpost prior to returning. Of course, many alternate scenarios have been developed to deal with any indigenous intelligent life forms.

Table 2 summarizes the various subsystems and elements of the interstellar transport and associated checkout requirements and implementation approaches. Locations of tests and the need for robotics and crew are also indicated.

## SPACEPORT OPERATIONS AND INCREMENT MANAGEMENT

### Increment Management Options and Considerations

In the context of the various test, checkout and performance test phases which have been presented, an increment refers to any segment of the activities which has a clear start and stop associated with the interaction with other space infrastructure and spacecraft (ref 11). What will be described in this section is not comprehensive, but represents some initial considerations.

In general, it is expected even with the availability of Nova class launch systems, that the interstellar spacecraft will have to be assembled in low Earth orbit at Space Station Freedom or Spaceport Earth. If many launch packages are involved, SSF would probably be the best site for the initial, non-hazardous integration activities. The large number of crewpersons, robotic systems and servicing capabilities would favor SSF over the Co-orbiting Platform (COP). If, on the other hand, a few very large packages were put into orbit with Nova class launchers, then one of two options could be pursued. The packages could be assembled at the COP or at a separate site much like SSF was originally (but with many less flights).

A combination of these options could also be considered. The interstellar transport could be assembled initially to serve as a mobile spaceport and evolve into use as a deep space transport. The practicality of this approach for interstellar spacecraft is doubtful because of the need for faster than light propulsion technology and the associated configuration considerations. It could, however, prove more than adequate for outer planet exploration and moon outpost support.

Nuclear system components and other potentially hazardous components or components with other kinds of public sensitivity, should be considered for launch from a relatively isolated Pacific spaceport. Alternatively, the components could be developed at a lunar outpost and transferred to a high Earth orbit for installation and assembly. Unless nuclear elements are mined and processed on the lunar surface, however, the launch of nuclear components from Earth can not be avoided. In addition, economics may not allow the lunar outpost to specialize in certain kinds of activity. For example, the collection and storage of anti-matter may require a lot of expensive and highly specialized equipment.

The servicing and maintenance of nuclear powered systems and other hazardous operations are best accomplished at co-orbiting platform sites in high Earth, lunar or Mars orbits. The platforms have to have the capability to fly to the deep space vehicle or guide a vehicle into a soft or hard docking.

The number and location of Orbital Replacement Units (ORU) is a challenge which will always be with us. With the use of built-in and microengineered sensors, our ability to predict and detect failures should greatly improve. The use of active microengineered elements such as heating elements, electrical and magnetic field effect variation devices, and built-in optical data paths can help compensate for and in some cases prevent ORU failures. As we continue to test new systems, especially propulsion systems, critical spares will still be required. The pattern of testing described earlier in this paper will allow many critical spares to be located on orbiting space infrastructure and at outposts rather than all on the transport itself. As we proceed in our space exploration and colonization activities, the improvement of component reliability and failure prediction should be one of the major design engineering efforts. At the same time logistics systems must plan and implement greater capabilities than will ever be needed to cover unforeseen contingencies.

While much has already been said about phases of systems and performance testing, it is worthwhile pointing out that contingency modes of environmental and crew systems should also be tested while attached to Space Station Freedom or while in low Earth orbit. Outer planet or interstellar spacecraft crews should not proceed on any mission without demonstrated viability of all planned contingency modes.

### Operations and Life Cycle Costs

Operations and life cycle costs for outer planet and interstellar missions will have the benefit of earlier lunar and Mars activities. As technology and materials improve, we expect the life cycle costs to continue to decline. Operations costs should also decline with increased reliance on automation and artificial intelligence systems. Most of the operations costs will be shared with other space operations activities. If commercial space infrastructure and self-reliant colonies and outposts have made sufficient progress, the operations costs could be quite reasonable in comparison to the costs likely to be associated with government ownership and operation.

In general, space infrastructure elements (spaceports, servicing platforms, outposts, colonies, etc.) should be multifunctional with a lead capability in one or more assembly, servicing, test and/or refueling functions. Each space infrastructure element will likely have some natural advantage which allows it to more cost-effectively perform certain functions. For example, a dedicated orbiting facility to conduct hazardous servicing operations or safe and recover from contingency situations might be an area of emphasis. While Earth orbiting platforms in this area are essential, the Mars Spaceport might specialize in major cleanup jobs involving nuclear radiation and matter/anti-matter systems.

To develop multifunctional facilities with special areas of emphasis, commercial involvement to add space infrastructure capabilities should be greatly encouraged and supported. In addition, the primary logistics routes between Earth, Spaceport Earth and the Lunar Outpost should eventually be a commercially bidded and operated activity. Multiple companies and vehicles should be simultaneously involved. Logistics between Earth, Spaceport Mars and the Mars colony/outposts could be supported by a combination of commercial and government funded efforts (ref 14). Logistics to the outer planets including Neptune and the Triton outpost will probably remain government funded until very advanced transport vehicles are more commonly available.

While it may or may not reduce operations costs, command and control, crew training and degrees of payload integration should be distributed among space infrastructure elements. Operations efficiency and safety are great benefactors of a planned distributed approach.

## RESEARCH AND DEVELOPMENT OPPORTUNITIES AND SUGGESTIONS

The author has argued that support for manned interstellar missions will be dependent on the development of technology which will "effectively" enable a spacecraft to travel faster than the speed of light. Without faster than light capability even the nearest stars (Alpha Centauri A is 4.35 light years away) requires a 40 year round trip time. The more interesting planetary systems are likely to be found around more distant stars such as Epsilon Bootes which is 114.1 light years away. Epsilon Bootes is a single star with a size much like that of our star, the Sun.

Recent theoretical studies, such as those associated with Superstring Theory, twistors, and other theories, point to the potential existence of higher dimensional physics. If the Superconducting SuperCollider does allow us to detect a Higgs boson, which may have linked the primary forces in the early universe, then the motivation for finding a means for interacting with the remnant hyperfield physics will be greatly increased. Even with the absence of such evidence there are R. & D. approaches which the author believes are worthwhile pursuing today to attempt to uncover "shortcuts" through space-time or to eliminate the speed of light restrictions which space-time imposes.

While specific attempts at resonant interactions with hyperfield physics are needed, a review of past and future experimental tests in other areas might also be useful. Experimental activity (ref. 5) associated with (1) plasmoid generation and anomalous magnetic flux replenishment (Los Alamos National Laboratory), (2) macro spin physics (Japan), (3) NASA gravity wave interferometer detector, (5) microfield anomalies in high temperature superconductors, quasi-crystals and other microengineered materials and (6) electromagnetic pulse tests should be examined.

New technology and research tools are available to assist in specific research. The tools include: (1) high temperature superconductors, (2) compact superconducting magnets with very high magnetic fields, (3) free-electron lasers, (4) high power, tunable microwave systems, (5) quasicrystals, (6) Superstring Theory, (7) microengineered materials with optical and magnetic "tailoring," (8) plasmoid generators and (9) supercomputers. These tools coupled with fundamental insight into the physics of the universe can lead to some startling technology systems.

The following suggestions for research into space-time disengagement or faster than light systems have not, to the author's knowledge, been pursued to any significant degree. Perhaps when these suggestions and others are pursued seriously the spark of insight which these suggestions represent will result in the igniting of a brilliant flame. Think of the enthusiasm which could be generated in students who are asked to examine these and similar ideas — who are encouraged "to go where no students have gone before."

Since there is no effective way of explaining these suggestions in a comprehensive way in this paper, these suggestions are best viewed as stimulation for the generation of your own ideas.

- Microwave/pulsed B field interactions with or without topologically, microengineered structures.
  - Could create microscopic discontinuities in space-time leading to macro-boundary discontinuities.
- Magneto-optical interactions for topological bending of light fields.
  - Could stimulate gravitational field interactions and space-time disturbances.
- Laser/plasma boundary generation – "light fluid."
  - Could create photon lattice-like structure enhancing and enlarging quantum field fluctuations.
- Topological patterned ferrofluid boundary – EM wave excitation.
  - Could create resonance with remnant hyperfields, generating artificial "space-time bubble"

## SUMMARY

Mankind has a great future in space awaiting it if we allow our innate frontier spirit to carry us forward. We can not afford to shy away from the challenges which we face in the development of cost-effective orbiting space infrastructure and associated operations. By looking ahead we can acquire a perspective which will help us make the right decisions in our near-term space development activities, such as Space Station Freedom evolution, lunar utilization and outpost and Mars exploration and outpost development .

The same approach which is used to assembly and test a manned Mars transfer vehicle will likely be applicable to manned outer planet and interstellar missions. Earth, Mars and Neptune/Triton spaceports and outposts will provide a very effective performance testing "safety net." Standardized, verified and consistently improved procedures and approaches for assembling and testing hazardous elements and systems should be primary goals of all exploration/colonization missions.

Manned interstellar propulsion will require faster than light propulsion to be economically justifiable. Even if mankind's survival were at stake so that the economics of sub-light propulsion were acceptable, we should still be challenging the speed of light constraint. Focused research and development activities by our national laboratories addressing these and other challenges of deep space missions should be started now (not a 100 years from now). These activities will help maintain our leadership in new technology and encourage new levels of student motivation and interest in science and math, both of which are critical to our nation's and mankind's future.

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N91-22167

# MINI-ROVERS FOR MARS EXPLORATION

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## ABSTRACT

Rovers are desirable for surface exploration because they allow sampling, and sample returns from several diverse locations on a planet's surface. Unfortunately, the rovers currently being examined for Mars exploration have several undesirable features. These rovers are quite massive (500kg to one ton), have very complicated operations, and are very expensive. This paper describes a possible alternative to using large rovers for exploring the surface of Mars. In this paper, the idea of mini-rovers is proposed. Mini-rovers weigh less than five kg, are trivial to control from the ground, and can do a more thorough survey of the terrain (per kilogram of mass) than can be obtained by large rovers. By redesigning the Mars sample return mission to accommodate the idea of mini-rovers and small spacecraft, considerable mass and cost savings can be achieved.

## 1. INTRODUCTION

The two major cost drivers in any spacecraft that has been flown have been mass and reliability. Mass is inherently expensive due to the energy costs for getting a spacecraft where it is going. Reliability has been approached either through subsystem redundancy (which increases mass) or major new technology developments combined with extensive testing (which drives cost directly). An alternative method for increasing reliability would be to have mission component redundancy; this again drives up mass and cost - but these can be brought back down if the components can be reduced in size, cost, and mass. For some types of mission, particularly robotic missions where the science payload is small or distributable, this may now be possible.

When the size of a mobile robot is reduced, most aspects of the robot improve or simplify. The power for the mobility system is greatly reduced. For most operations this makes solar power more practical. The mobility of the system may actually improve; in natural terrains, the fractal nature of the terrain combined with the reduced surface pressures allow greater freedom of movement.

Up until now, a major stumbling block in reducing the size of robots for planetary missions has been the combination of communications and intelligence. Communications does not scale well. Maintaining a sufficiently high data rate to control a robot from the Earth precludes reducing the size and power of the robot. To reduce communications, one can make the robot more autonomous. JPL projects in this vein (e.g., Pathfinder rover) have usually taken approaches that require several tens of MIPS of processing power onboard, which meant that the computation system was the power and mass driver. However, these are not the only possibilities.

Recent work in behavior control languages [Brooks86, 89], [Gat 90] has shown that robots can be controlled to perform useful unstructured tasks with relatively little computation and very small programs (a fraction of a MIP and a few K bytes of program). The resulting control programs allows a robot to operate almost completely autonomously. This again reduces the need for communications and computation. With communications and computation greatly reduced, the entire size of the robot can be reduced as well. A smaller lighter robot, requires less fuel to get it to its destination, can more easily be landed on a planetary surface (parachutes and impact limiters become feasible), and can be sent in greater numbers for comparable costs than can larger robots [Miller89a, 89b].

By carefully thinking out the robot's tasks, it is possible that small robots can be made to accomplish most of what larger robot rover can. The behavior control languages used in mini-robots are more than adequate for having a rover avoid obstacles, rendezvous with the ascent vehicle, and collect samples. Careful selection of sensors and placement of radio beacons will allow such a robot to carry out other science tasks. Things that are difficult for this type of robot to accomplish are map-making and carrying massive payloads. However, map-making is mostly used to guide a robot, and there are alternatives. Massive payload can often be distributed among several small robots.

A behavior controlled robot has many of the characteristics of an insect. It has a robust set of simple behaviors that will allow it to handle most situations.Such a robot does not maintain a detailed world model - it therefore cannot readily determine when something unexpected has occurred. Like an insect, in situations that fall outside of its design parameters it will often fail. But also like an insect, such a robot can be small and inexpensive enough so as to have replacement robots waiting to take over.

## 2. MINI-ROVER MISSION SCENARIO

Below is a possible mission scenario for a multi-sample site, sample return mission to Mars that I believe could accomplish the major scientific and manned precursor goals of a full-up MRSR (Mars Rover Sample Return) mission at a substantially lower cost. The central feature in this mission scenario is the use of 100 mini-rovers (each < 5kg) rather than a single one ton rover. The use of the mini-rovers also effects the landers, orbiter, and Mars ascent vehicles. The Mars-Earth return vehicle would probably remain more or less unchanged, as would the launch system - though it may now be possible to go to a smaller launch vehicle.

### Mission Elements

The major mission elements are a relay communications orbiter (MCO) placed in Mars-synchronous orbit, 100 mini-rovers and their landing pods, a Mars ascent vehicle (MAV) and its landing pod, and a Mars-Earth sample return vehicle (ERV). Every five rovers form a group that has an associated landing pod which deploys a small parachute and then hard lands on the Martian surface. An airbag or crushable front-end is used to decelerate the rovers upon impact. Two Mars ascent vehicles, each with an associated landing pod, use similar methods to land on the Martian surface. The MAVs also has an inflatable belt that can be used to reorient them into a launch orientation. The MAVs contain a non-directional radio beacon (NDB). The MAVs use a two-stage launch sequence from Mars. The first stage boosts the MAV to near orbital velocity. The second burn occurs at apoapse, and puts the return capsule into low Mars orbit. The return capsule has a solar powered docking beacon and is spherical in shape. The capsule is not stabilized, but its spherical shape (and central CG) will allow simple docking with the Earth return vehicle.

### Mission Scenario

1) The MCO is placed in Mars synchronous orbit at the longitude of the landing site.
2) The ERV is placed in a Mars rendezvous orbit.
3) The MAVs and rover groups are landed on the surface with the MAVs in the approximate center of the dispersion of rovers. The rovers form a buckshot pattern covering approximately 100km diameter spread (see Figure 1). At the East Mangala site this would cover several different types of terrain.
4) The primary MAVs' NDB is deployed and the rovers home to that.
5) Each rover gathers one or two samples near its landing site, recording the relevant information. The group of rovers employ their different science instruments on each of the sample areas.
6) At the start of each day the rovers progress towards the primary MAV.

7) Each day the rovers find a piece of clear local high ground and take a stereo pair, and other science data.
8) Before sundown, the rovers relay their data to the MCO.
9) The MCO relays the new data to Earth.
10) As each rover reaches the MAV, it transfers the sample to the MAV, does an about face and starts its extended mission.
11) When the MAV is ready to ascend, it leaves the NDB so that the rovers will continue to have orientation information.
12) The MAV ascends. The ERV docks, retrieves the sample canister, and returns it to Earth.

## 3. MASS ANALYSIS

The rovers small size allows them to be relatively hard landed on the surface. It also makes them much more power efficient. The rovers use local radio communications, to minimize the power expenditure. The rovers are solar powered during the day. It may also be possible to use the temperature differential between the top soil and the night air to power the rovers at night. The only commands that need be given to the rover from the ground would be to drop its current sample and get another one from its current location. The rovers could have a low-power slow speed vise to crack rock and get fresh samples. Each rover would return a few grams of sample (see Figure 2).

The MAVs could be very small. Assuming 200g of samples (4g each from 50% of the rovers), it should be possible to design a total return capsule to mass approximately 1.5kg. Remember, the capsule is a totally passive system with a small solar powered radio beacon. The capsule has no active thermal control, but is heavily insulated. The capsule is either in Mars normal conditions, when it is on the surface, or part of the ERV. The ERV has less

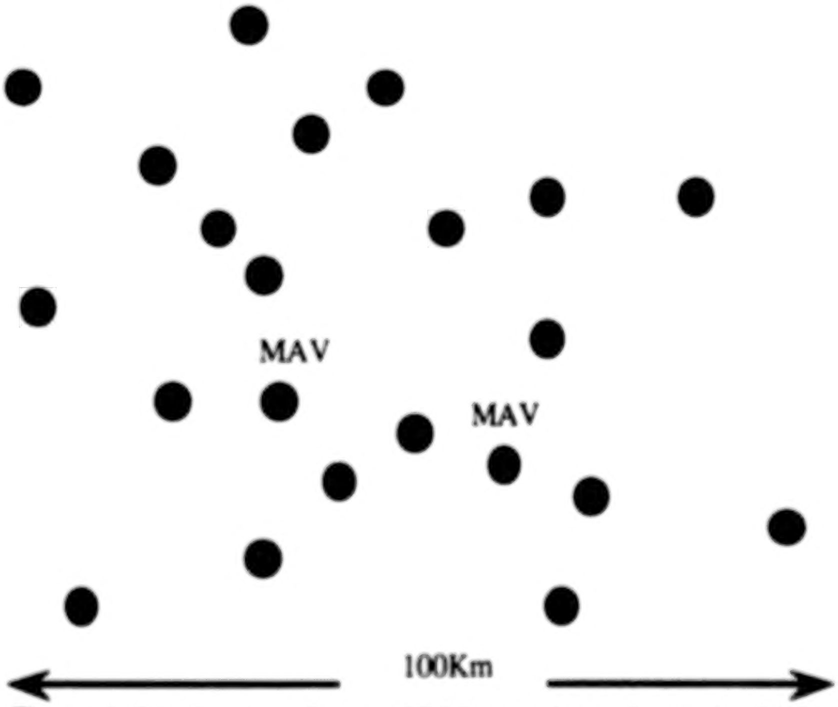

**Figure 1.** Distribution of rover CSAD capsules and ascent vehicles

stringent mass constraints and could therefore exhibit active thermal control on the capsule. The capsule would be on its own, in orbit, only for a short time.

Low Mars orbit requires a velocity of 3.6km/sec. 280second impulse is typical for the type of low-maintenance fuel that might be used on a MAV. Such a fuel has an exhaust velocity of approximately of 2.7km/sec. This leads to a mass ratio of 3.79 between fuel and the remainder of the MAV to get it into orbit. Assuming the rocket casing requires a mass of approximately 10% of the propellant weight, and that a short term (missile style) gyro and gas reaction jet system would mass approximately 2.5kg. The masses on the MAV have the following breakdown:

| | |
|---|---|
| Sample return capsule | 1.5kg |
| Guidance & Control | 2.5kg |
| Rocket casing | 3.0kg |
| Rocket fuel | 30.0kg |
| TOTAL MASS | 37kg |

The MCO acts as a relay between the MAV and the Earth. The pointing accuracy needed for the orbiter is the minimum needed for communications with Earth. No imaging is needed for this mission. Viking and MO data will suffice for landing site selection. The Areo-stationary Direct-relay Communications Orbiter is one possible model for the MCO. With its reaction control system fully loaded it masses approximately 400kg.

The Earth return vehicle must return the 1.5kg sample capsule and will (from D.Bernard) also require a guidance system massing approximately 60kg. Approximately 2.3km/sec deltaV is required for moving from Mars orbit into Earth insertion. This leads to a mass ratio of 2.35 when using the same fuel as used in the MAV. For an Earth orbit rendezvous, it should be possible to build an Earth return vehicle to bring back the 1.5kg sample canister where the entire return vehicle masses under 600kg. This includes a large margin for performing the Mars rendezvous with the return capsule.

The *Capsule System Advanced Development System*, developed in 1967 to survivably land a science package on the surface of Mars, used a mass ratio of approximately 1.6 between the support hardware (e.g., aeroshell, parachute, and shock absorbers, etc) and the payload that was landed in working condition. The CSAD capsule was derived from the Ranger landing capsule [Ranger63] which had a slightly higher mass ratio (approximately 2:1). CSAD was designed for

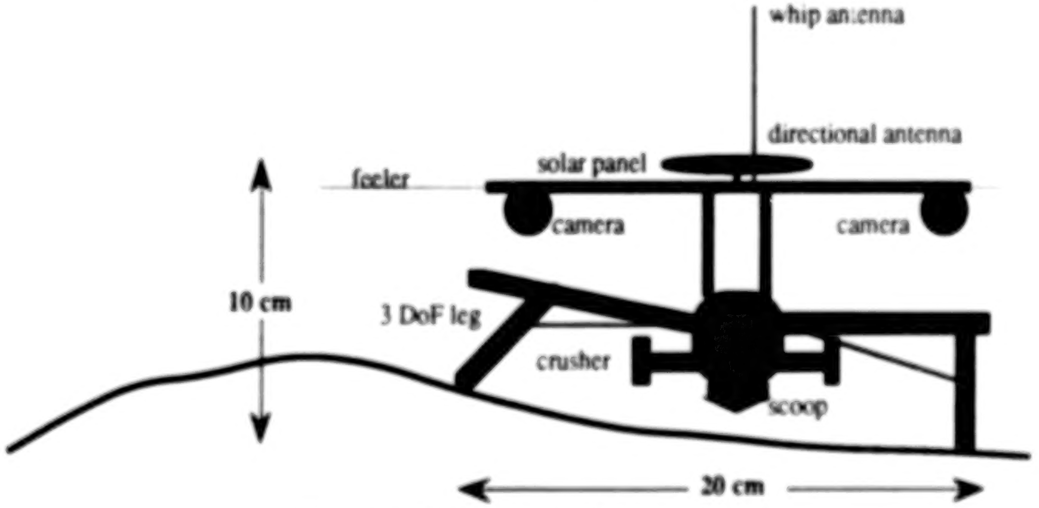

**Figure 2.** Front view of a Mini-Rover

landing payloads in the 30-50kg range. The MAVs and the mini-rovers mass approximately 580kg. Using the 1.6:1 ratio for the CSAD capsule, the capsules would mass approximately 928kg giving a combined mass to be sent into Mars terminal descent of 1508kg. 400kg is placed in Mars-synchronous orbit (the MCO), and 600kg are in a low-Mars rendezvous orbit (the ERV). The mission described above would require a total of about 2.5 metric tons to be flown into Mars space.

## 4. ADVANTAGES OF USING MINI-ROVERS

Some of the advantages of this scenario over the traditional MRSR scenario are:

1) Very high redundancy: While it is almost certain that some rovers will not survive to deposit their samples, many will.
2) Larger scientific coverage: Approximately 2500km will be traversed in the scenario outlined above (assuming a 50% rover survival rate). More varieties of terrain will be covered. More varied samples can be gathered.
3) This mission should be much cheaper. The technology is simpler. There is no pin-point landing required, there is no imaging orbiter required, there is no nuclear power technology needed, the computer technology needed already exists and is already space qualified, as is the pointing technology, landing technology, and the communications technology; the amount of mass landed on the surface is also greatly reduced.
4) This mission could be brought together in a relatively short time, perhaps for a '94 launch. This means that samples could be back in time to provide landing direction and instrumentation analysis clues for a full-up rover mission during a later opportunity
5) The rover technology for this mission is relatively cheap and easy to test and demonstrate.

## 5. CONCLUSIONS

The pieces needed to make use of mini-rovers for planetary missions are all under intense development with the exception of the robots themselves. Lightweight, low power cameras [Ravine89], and science instruments [Murphy81, Manning77] have already been developed. Power systems (both solar and RTG) that mass a few hundred grams and deliver a few watts, have been developed [JPL88]. The same is true of communications systems that can broadcast to the DSN at 55bits per second at three AU, and masses under a kilogram, drawing less than four watts [JPL88].

Mini-rovers offer a potentially cost-effective way of exploring Mars and other planetary surfaces. The use of small rovers has many advantages and few disadvantages over using large rovers. By taking advantage of scaling laws and currently available technology, a micro-rover sample return mission is doable in the very near future.

**Acknowledgements:** The research described in this paper was carried out by the Jet Propulsion Laboratory — California Institute of Technology under a contract with the National Aeronautics and Space Administration.

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N91-22168

# ADVANCED SPACECRAFT: WHAT WILL THEY LOOK LIKE AND WHY?

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## ABSTRACT

The next century of spaceflight will witness an expansion in the physical scale of spacecraft, from the extreme of the microspacecraft to the very large megaspacecraft. This will respectively spawn advances in highly integrated and miniaturized components, and also advances in lightweight structures, space fabrication, and exotic control systems. Challenges are also presented by the advent of advanced propulsion systems, many of which require controlling and directing hot plasma, dissipating large amounts of waste heat, and handling very high radiation sources. Vehicle configuration studies for a number of these types of advanced spacecraft have been performed at the Jet Propulsion Laboratory over the past decade, and some of them are presented in this paper along with the rationale for their physical layouts.

## SPACECRAFT CONFIGURATION

Over the years, the Jet Propulsion Laboratory (JPL) has studied many concepts for advanced and exotic spacecraft which might come to fruition in the 21st century. For a number of these studies, the author was involved in developing spacecraft mechanical configurations to integrate the various elements of a vehicle into an optimized structural and mechanical layout.

Advanced spacecraft concepts generally push the state of the art in propulsion, temperature control, materials, precision pointing control, size, mass, or packaging density. These requirements often conflict with one another, and complex trade studies must be undertaken to achieve an optimal design.

Integration of function, though often costly, is one method to reduce the size or mass of a vehicle. For example, using a pressure vessel or a thermal radiator as primary support structure, or integrating an antenna reflector and a solar power concentrator into a single structural component reduces the number of elements to be supported. The price to be paid is loss of modularity and more complex analyses and interfaces.

Requirements on fields of views for solar panels or radiators, or geometric constraints for radiation protection often force the layout of a vehicle to a particular configuration. Additionally, large vehicles in planetary orbits must trade off the above constraints against such external forces as gravity gradient and atmospheric drag. To avoid control problems, vehicles which are spin stabilized (or which rotate to be nadir pointed in a low planetary orbit) should be designed to rotate about one of the three principle inertial axes of the spacecraft, preferably about the axis of greatest inertia. This is especially important for large flexible structures.

In the end, trading off these many complex constraints requires an iterative approach which is often unique for each vehicle. Some attempts have been made to integrate the optimization of different disciplines, such as a combined structures and controls optimization, and in the future the spacecraft design process may become more direct and less iterative.

## ADVANCED PROPULSION

Most space vehicles today utilize chemical propulsion with specific impulses of under 460 sec (or exhaust velocity less than 4.5 km/sec). Many advanced spacecraft of the next century will require more exotic forms of propulsion to achieve higher velocities or to carry greater payloads.

## SOLAR SAILS

Solar sails are attractive because they utilize solar photon pressure for propulsion and therefore require no propellant. Large flat sheets of shiny material reflect sunlight, and some momentum is transferred to the reflective film. The resultant force depends upon the angle of incidence of the light, therefore the vehicle can be steered to direct the force vector in a desired direction.

JPL performed extensive studies of a Halley's Comet rendezvous mission in 1977, including a design for a three-axis stabilized square sail vehicle and a spin-stabilized "heliogyro" solar sail (reference 1). More recently, JPL has provided some support to the World Space Foundation in developing a smaller engineering test vehicle to demonstrate deployment and control of a solar sail and to obtain flight data (reference 2).

Spin-stabilized sails may provide higher performance because they require less support structure, but they are more difficult to steer rapidly because of the gyroscopic forces which must be overcome, and the attendant structural control problems inherent in a rapid precession maneuver for a large flexible vehicle. This is not much of an issue for vehicles in a solar orbit since the required turn rates are so slow. However, in a planetary orbit, a solar sail must typically turn at least 180° each orbit, which can lead to relatively fast turn rates for such a large flexible structure.

The World Space Foundation design (see Figure 1) calls for a 3,000 $m^2$ square sail which is supported by four simple cantilevered beams (spars) emanating from a central body. Three-axis attitude control is provided by steerable triangular vanes at the tips of the spars, and by moving a mass on a steerable boom to shift the center of mass relative to the center of solar pressure. The deployment sequence for the vehicle is rather simple as solar sails go (see Figures 2 and 3).

Square sails larger than about 5,000 $m^2$ probably cannot be supported by simple cantilevered spars and will require extensive stays and guy wires to stabilize the structure, as was the case with the Halley square sail. Autonomous deployment for that type of complex structure may be risky, and on-orbit construction may be preferred for such a vehicle.

Some disadvantages of solar sails are their low acceleration, typically about 1 $mm/sec^2$ at 1 AU, and their very low performance beyond the orbit of Mars. Their application of greatest utility may be as reusable interplanetary cargo shuttles for the inner solar system. High performance solar sails may find utility in Earth orbit for positioning communication satellites in non-equatorial locations using levitated geostationary orbits, or as non-orbiting hovering statites at high latitudes (reference 3). These two latter groups of vehicles do not require fast turn rates.

Related vehicles which could become prevalent in the coming millennium include the solar photon thruster (reference 4), laser sailing (reference 5), and microwave sailing (reference 5).

## ELECTRIC PROPULSION

Electric propulsion (ion drive, arc jet, or plasma jet) will almost certainly be utilized in the next century due to its high performance (2,000 to 30,000 sec $I_{sp}$). One of the disadvantages of electric propulsion is the requirement for a high energy source (many kilowatts electric). This will most likely

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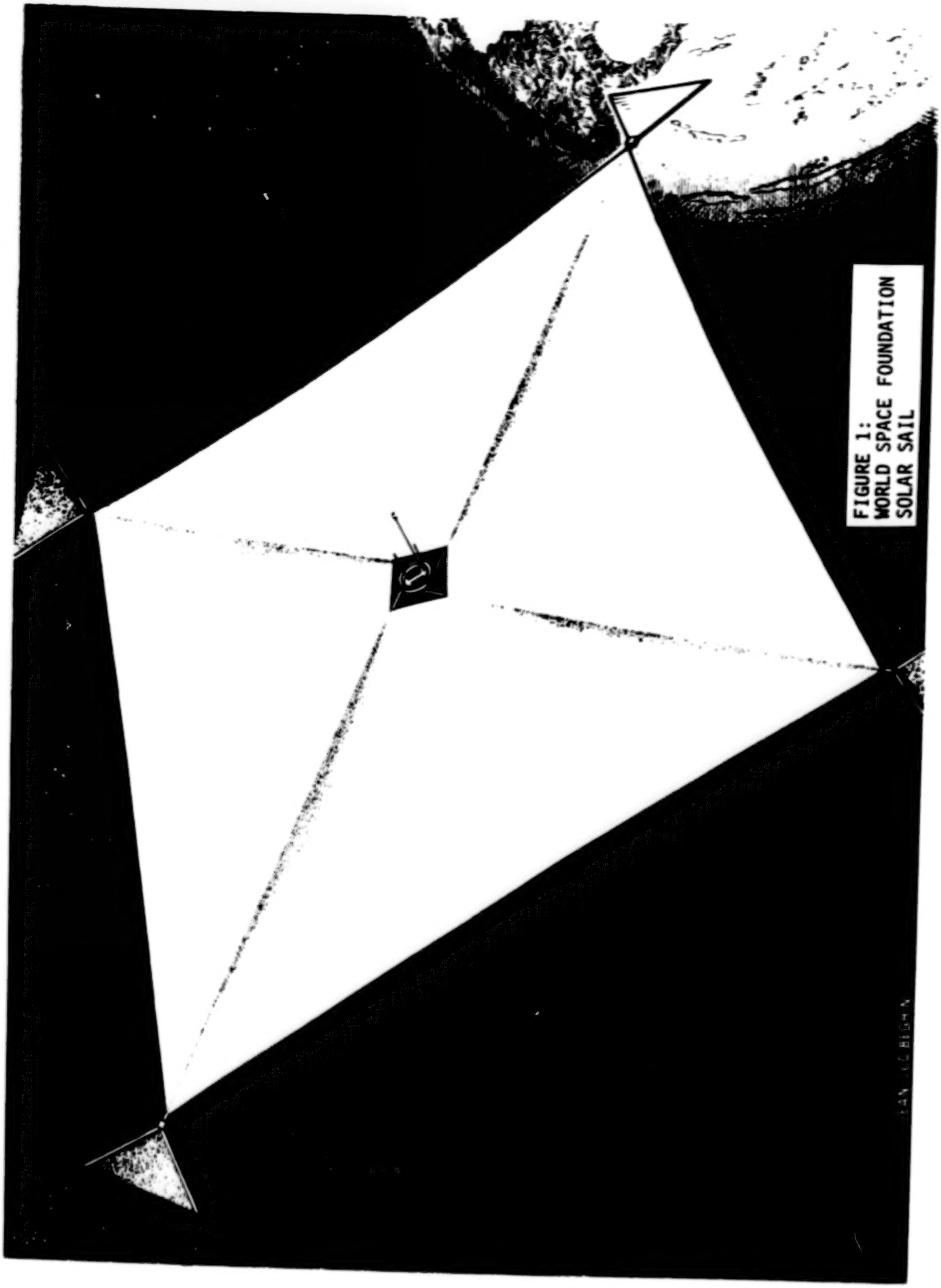
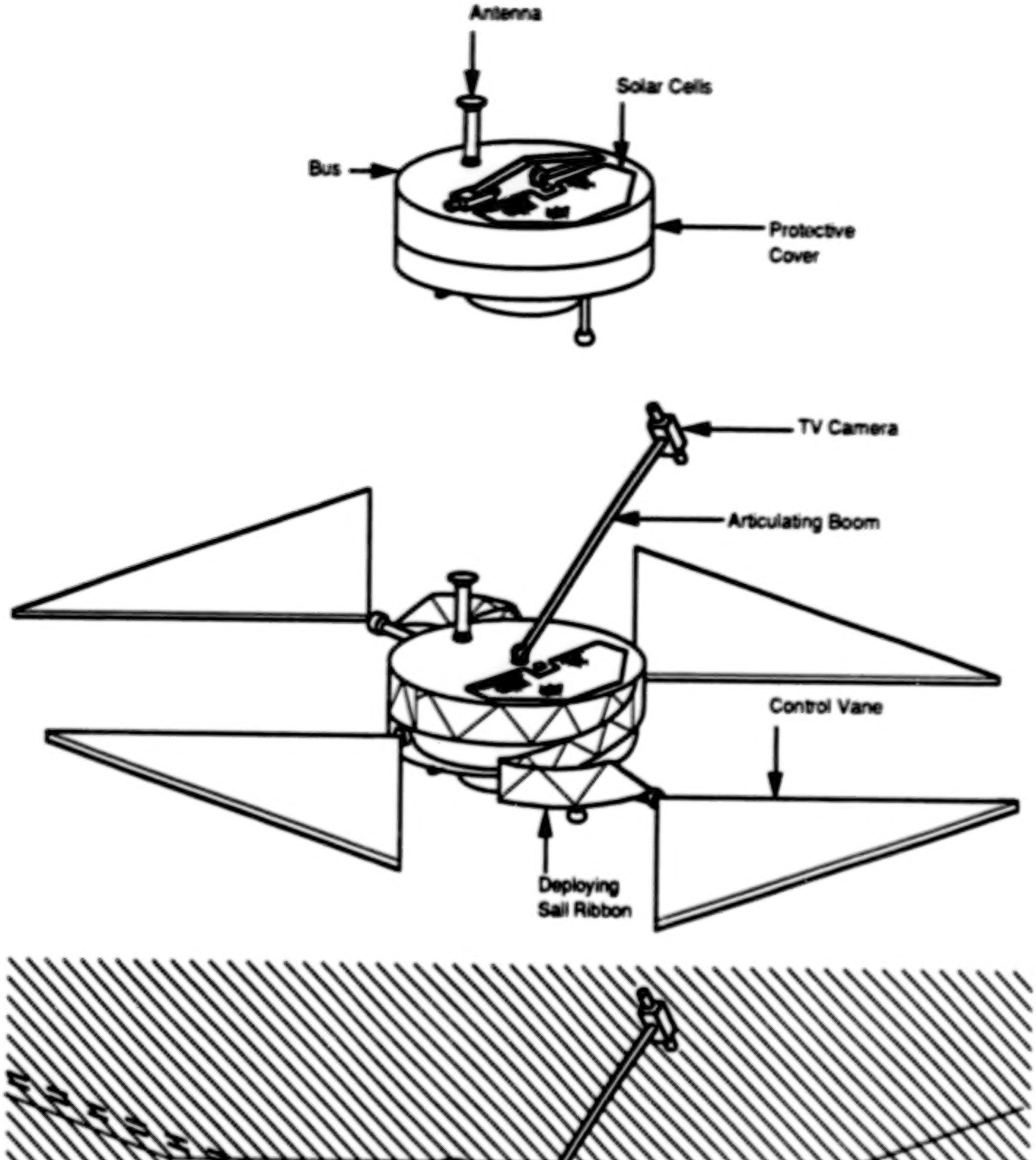


FIGURE 1:
WORLD SPACE FOUNDATION
SOLAR SAIL


Antenna
Solar Cells
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Deploying Sail Ribbon
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Bus
Sail

Figure 2: Deployment of Vanes and Spars on the World Space Foundation Solar Sail Vehicle

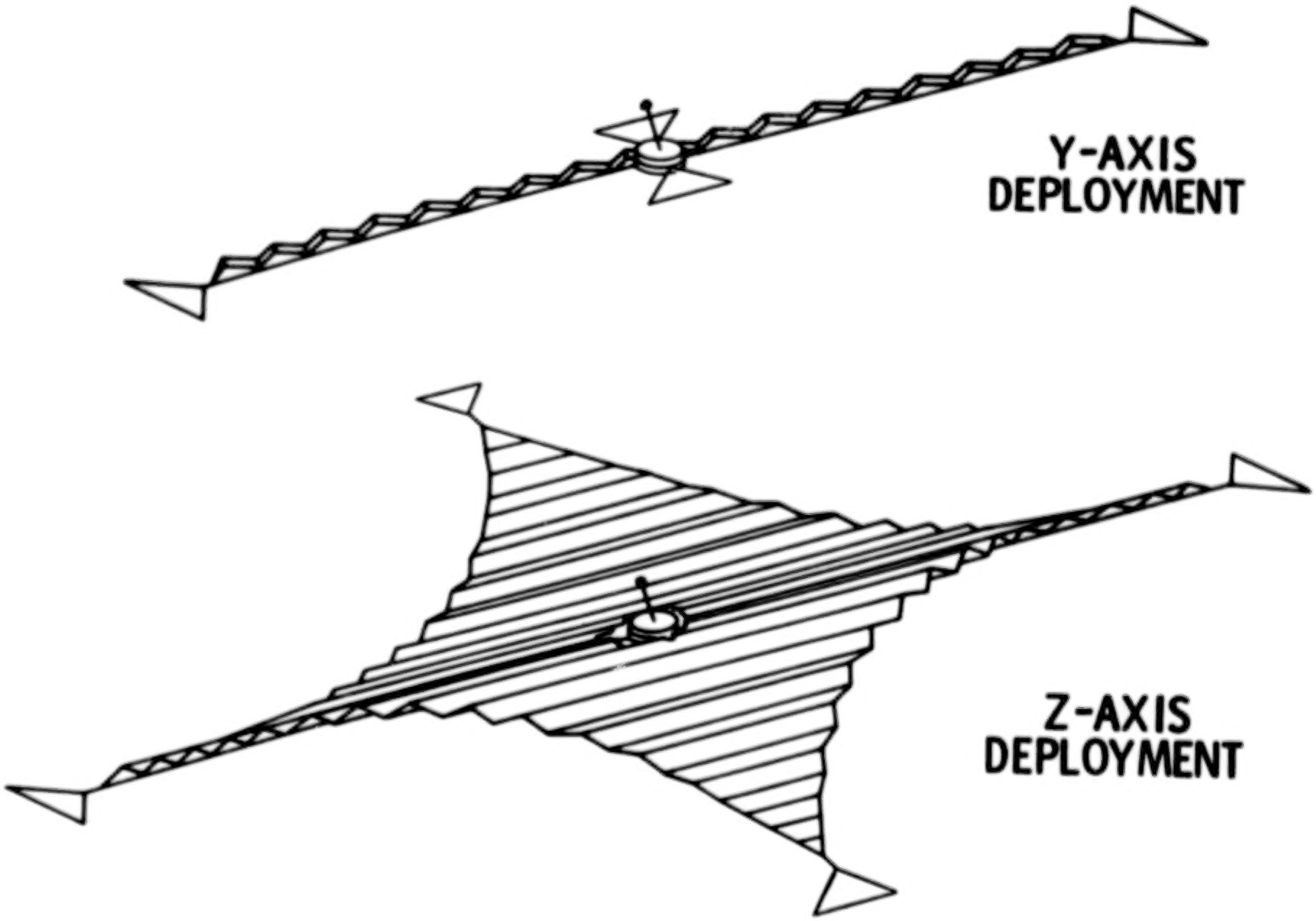

Y-AXIS
DEPLOYMENT
Z-AXIS
DEPLOYMENT

Figure 3: Deployment of Sail on the World Space Foundation Solar Sail Vehicle

be achieved by solar or nuclear means.

Power processing components for the ion thrusters require the dissipation of large amounts of heat. Additional heat rejection requirements arise from the solar or nuclear electrical generation system; therefore, these spacecraft will have significant radiator surfaces whose orientation must be maintained relative to the sun and also configured to minimize their field of view to the spacecraft itself. Clever vehicle design may tap some of this waste heat for useful purposes such as temperature control or secondary power generation. Ion drive has been extensively ground tested, space tested, and is in a state of immediate technological readiness.

**Solar Electric Propulsion (SEP)**

The development of multi-kilowatt solar arrays (SAFE, space station, APSA) brings SEP within easy technological reach, with all major components of the system having been developed. JPL has developed many designs for SEP vehicles including the detailed Halley flyby/Tempel 2 rendezvous mission studies in 1979-1980 (reference 6). A more recent SEP study was performed for the Mariner Mark II Project in 1986.

The Mariner Mark II (MMII) spacecraft is JPL's next generation of interplanetary spacecraft, now under development. The first two units, Comet Rendezvous/Asteroid Flyby (CRAF) and Cassini, will go to a comet and Saturn respectively. Follow on missions for the MMII vehicle class are planned. The addition of SEP would greatly enhance the utility of this spacecraft by expanding its propulsive capability.

The design depicted in Figure 4 integrates SEP as an add-on stage to what would be an already existing chemical propulsion system, except for the large solar arrays which are added to the main vehicle structure. The ion drive power processing electronics are integrated into a moderate sized radiator on the SEP stage. This design utilizes five independently gimbaled ion thrusters, and xenon propellant.

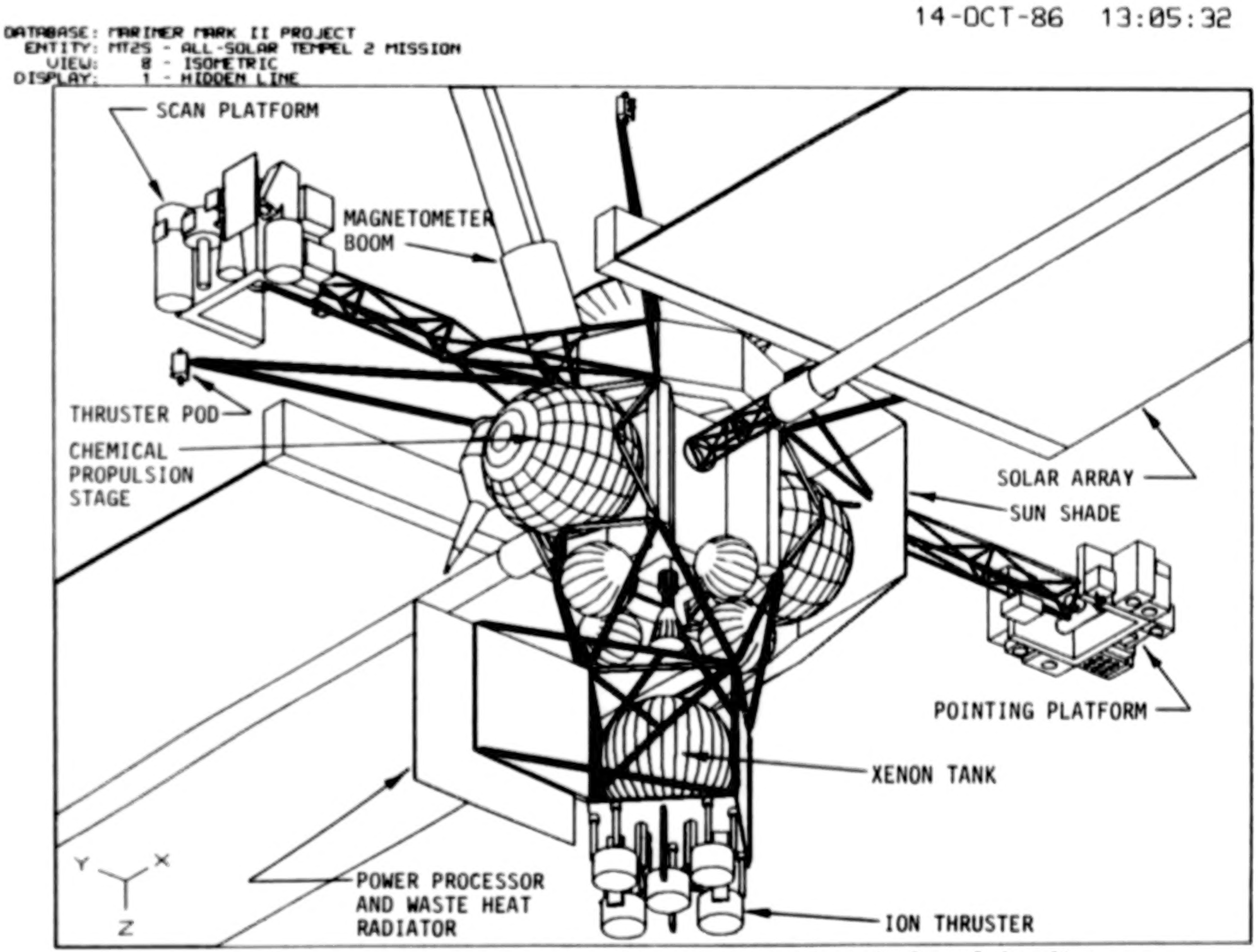
SEP places some additional configuration constraints over a standard propulsion system. Since the "burns" take place over several months rather than several minutes, the vehicle must be continually oriented to the thrust direction rather than to the sun during interplanetary cruise. This means the spacecraft must be able to tolerate sun illumination from a variety of directions. In addition, large steerable solar arrays must be continuously pointed at the sun, and sun must be kept off of the power processor radiator(s).

The MMII SEP design requires that the spacecraft maintain roll control about the thrust (Z) axis to keep the sun in the Y-Z plane (see Figure 4). The solar arrays are then articulated about the X axis to sun point, and the radiator is fixed in the Y-Z plane to avoid sun incidence. Roll control about the Z axis can restrict the sun to be in the -Y hemisphere, and a large sun shade is required to be added to the -Y side of the vehicle to protect it from broadside sun onto the chemical propulsion system and the electronics bays.

These additional complications to the spacecraft are somewhat costly, including provision for the launch stowage and later deployment of the large solar arrays, and the obscuration of science instrument fields of view by the large arrays. The large arrays also present some attitude control complications for the vehicle, but the enormous increase in propulsion performance makes SEP an enabling technology for many possible MMII missions.

**Nuclear Electric Propulsion (NEP)**

For missions which require propulsion beyond the orbit of Mars, NEP is generally favored over SEP. Using a nuclear fission reactor as its electrical source, NEP offers the benefit of much higher power and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ENTITY: MT2S - ALL-SOLAR TEMPEL 2 MISSION
VIEW: 8 - ISOMETRIC
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14-OCT-86 13:05:32
SCAN PLATFORM
MAGNETOMETER BOOM
THRUSTER POD
CHEMICAL PROPULSION STAGE
SOLAR ARRAY
SUN SHADE
POINTING PLATFORM
XENON TANK
POWER PROCESSOR AND WASTE HEAT RADIATOR
ION THRUSTER
Y
X
Z

Figure 4: Mariner Mark II Spacecraft with Solar Electric Propulsion Stage

performance than SEP, independent of distance from the sun, with the disadvantage that extensive shielding and physical separation measures are required to protect most components of the vehicle from radiation emitted by the nuclear power source. Some of the most recent NEP studies at JPL have been in support of the Thousand Astronomical Units (TAU) mission, led by Aden and Marjorie Meinel.

The TAU concept uses NEP to accelerate a large spacecraft complex over a period of 10 years to a velocity of over 100 km/sec, heading out of the solar system from its original assembly point in Earth orbit. After fifty years, the vehicle will reach a distance of 150 billion kilometers (1000 AU, or 0.016 light years). Among its compliment of science instruments would be a large telescope to function as a wide baseline astrometric platform relative to the Earth from which to provide greatly improved estimates of interstellar distances and the Hubble constant.

The TAU design depicted in Figure 5 uses a 100 $kW_e$ nuclear power supply based on the SP-100 reactor system. It is located at one end of the 43 m long complex. Most spacecraft subsystems and the scientific payload are located at the opposite end of the complex to achieve maximum separation for radiation protection. In the middle of the complex, desirably near the center of mass, is the ion propulsion system. These three major elements are connected by a long structural trusswork which could be either deployable or assembled in Earth orbit. The separation of the elements is dictated primarily by radiation constraints, requiring a trade-off of shielding mass versus truss structure mass versus cost of radiation hardening of components.

The configuration looks like a long stick, and the thrust direction is perpendicular to the long axis of the vehicle. The peak acceleration is about .5 $mm/sec^2$, so the structural loading is very slight, and the mass of the truss is driven by control stiffness requirements rather than by loads.

Both the nuclear power module and the ion drive module have substantial radiative cooling requirements. The configuration shown here utilizes cylindrical radiators in which the sun is allowed to illuminate them from any direction. If required, it would be possible to substitute flat radiator elements which could be kept edge-on to the sun by controlling roll about the vehicle's thrust axis.

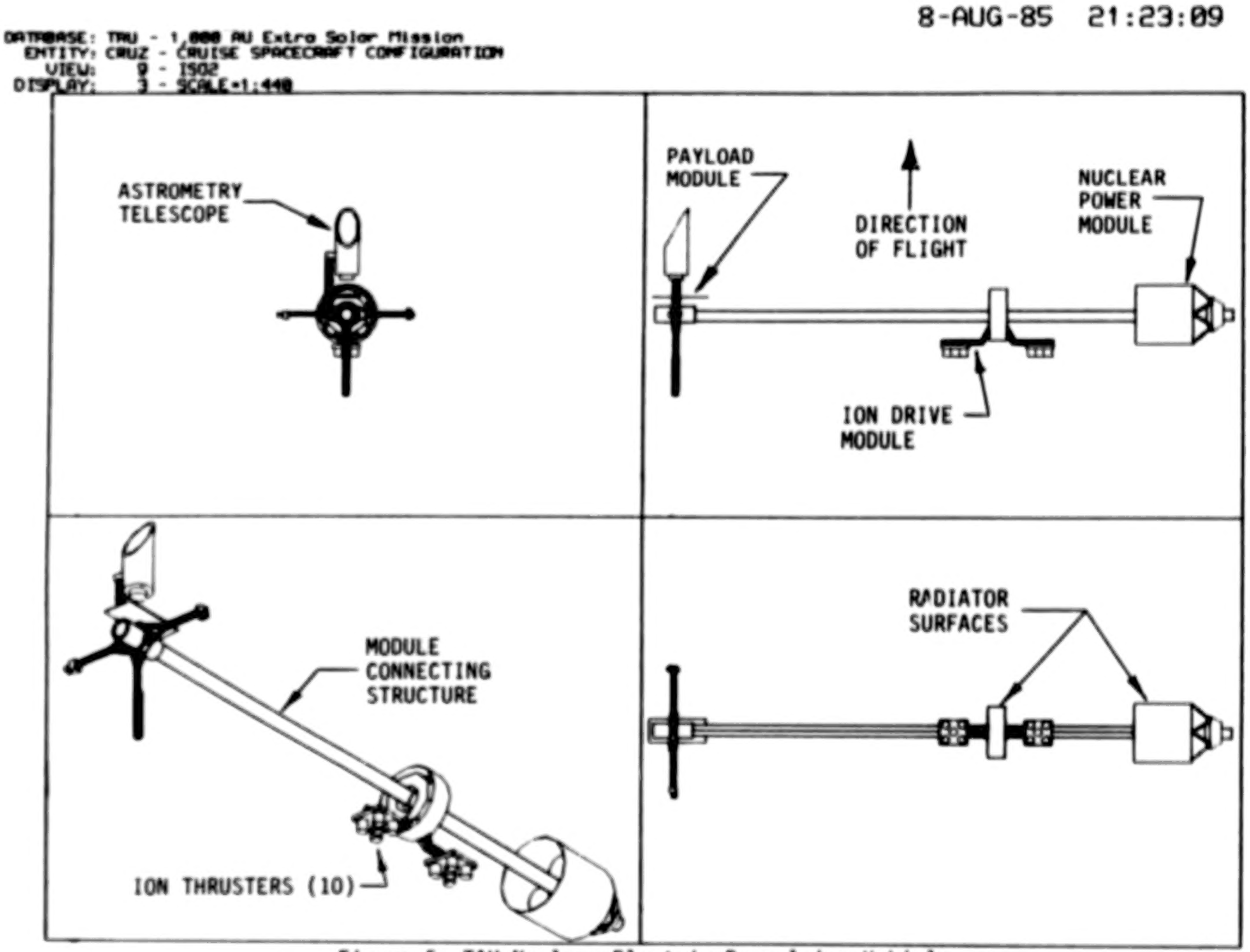
A more recent study defines an option for a larger vehicle which is 140 m long, uses a 500 $kW_e$ reactor system, and carries a larger payload (reference 7).

## PLASMA PROPULSION

Missions requiring velocity changes greater than about 100 km/sec must look for more exotic forms of propulsion than ion drive can offer. Much higher specific impulses might be achieved by using nuclear energy sources to create a high energy plasma which can be expanded and directed at very high velocities. Since the plasma temperatures are too high for any solid material to contain, the plasma must be directed by a powerful electromagnetic field. An exception is the Orion concept (reference 8) which uses a heavy ablative blast shield.

Plasma propelled vehicles require an enormous investment in radiation protection. Perhaps the most challenging problem is in the shielding of the magnetic drive coils which must be relatively close to the plasma in order to contain it or direct it. A particularly clever concept was developed by Rod Hyde at Lawrence Livermore National Laboratory (reference 9) which utilizes only a single torroidal drive coil (Figure 6). The idea is to minimize the interception fraction of the plasma radiation with the drive coils and thereby minimize the mass of shielding required.

Magnetic plasma nozzles with multiple drive coils may more efficiently direct the plasma exhaust, but each coil requires its own heavy radiation shield with its attendant cooling requirements. Additionally, the radiation shield for one coil produces secondary radiation for which additional shielding must be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ASTROMETRY TELESCOPE
PAYLOAD MODULE
DIRECTION OF FLIGHT
NUCLEAR POWER MODULE
ION DRIVE MODULE
MODULE CONNECTING STRUCTURE
ION THRUSTERS (10)
RADIATOR SURFACES

Figure 5: TAU Nuclear Electric Propulsion Vehicle

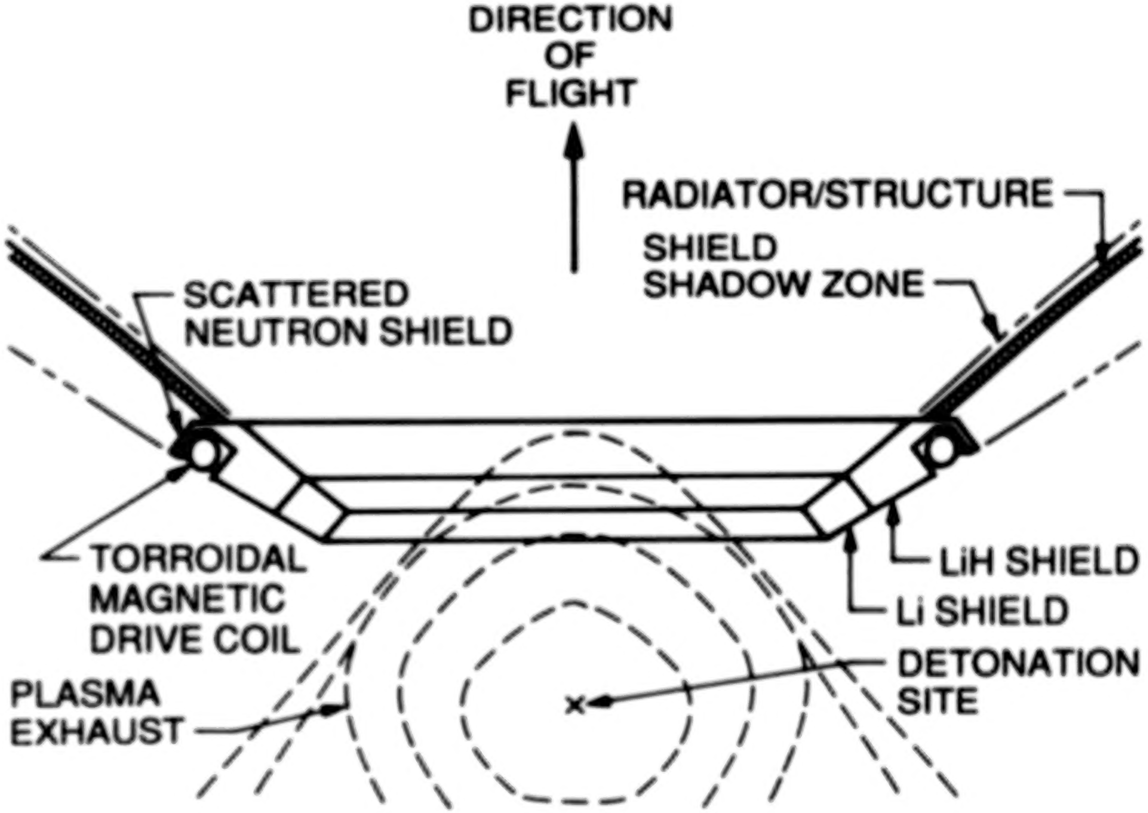

DIRECTION OF FLIGHT
RADIATOR/STRUCTURE
SHIELD SHADOW ZONE
SCATTERED NEUTRON SHIELD
TORROIDAL MAGNETIC DRIVE COIL
LiH SHIELD
Li SHIELD
DETONATION SITE
PLASMA EXHAUST

Figure 6: Geometry for Plasma Rocket Using a Single Magnetic Drive Coil

provided in adjacent coils.

The most prominent feature of any plasma propelled vehicle will be the waste heat rejection radiator surfaces, and a major portion of the waste heat will come from cooling radiation shields.

Adopting the Rod Hyde drive coil geometry, Joel Sercel of JPL first suggested trying to fit the entire vehicle into the shadow cone of the drive coil radiation shield, thus reducing shielding mass and cooling requirements to an absolute minimum. To integrate function, a conical shell radiator surface can also serve as the spacecraft's primary structure and provide the attachment base for all of the vehicle's various components. A torroidal propellant tank can be efficiently integrated atop the conical shell radiator (at the wide end of the cone), thus minimizing the field of view of the radiator to both the cryogenic propellant, cryogenic drive coil, and the hot plasma (see Figure 7). The structural load paths are direct and efficient.

The payload is supported atop the torroidal propellant tank, minimizing its exposure to radiation. Thus, the optimized vehicle is a large conical shell traveling with the wide open end forward, expelling plasma out the drive coil at the narrow aft end of the cone. For even more advanced propulsion options, such as the Bussard ramjet (reference 10), the cone might be adapted into a scoop to obtain additional reaction mass from the interstellar medium.

**Fusion Propulsion**

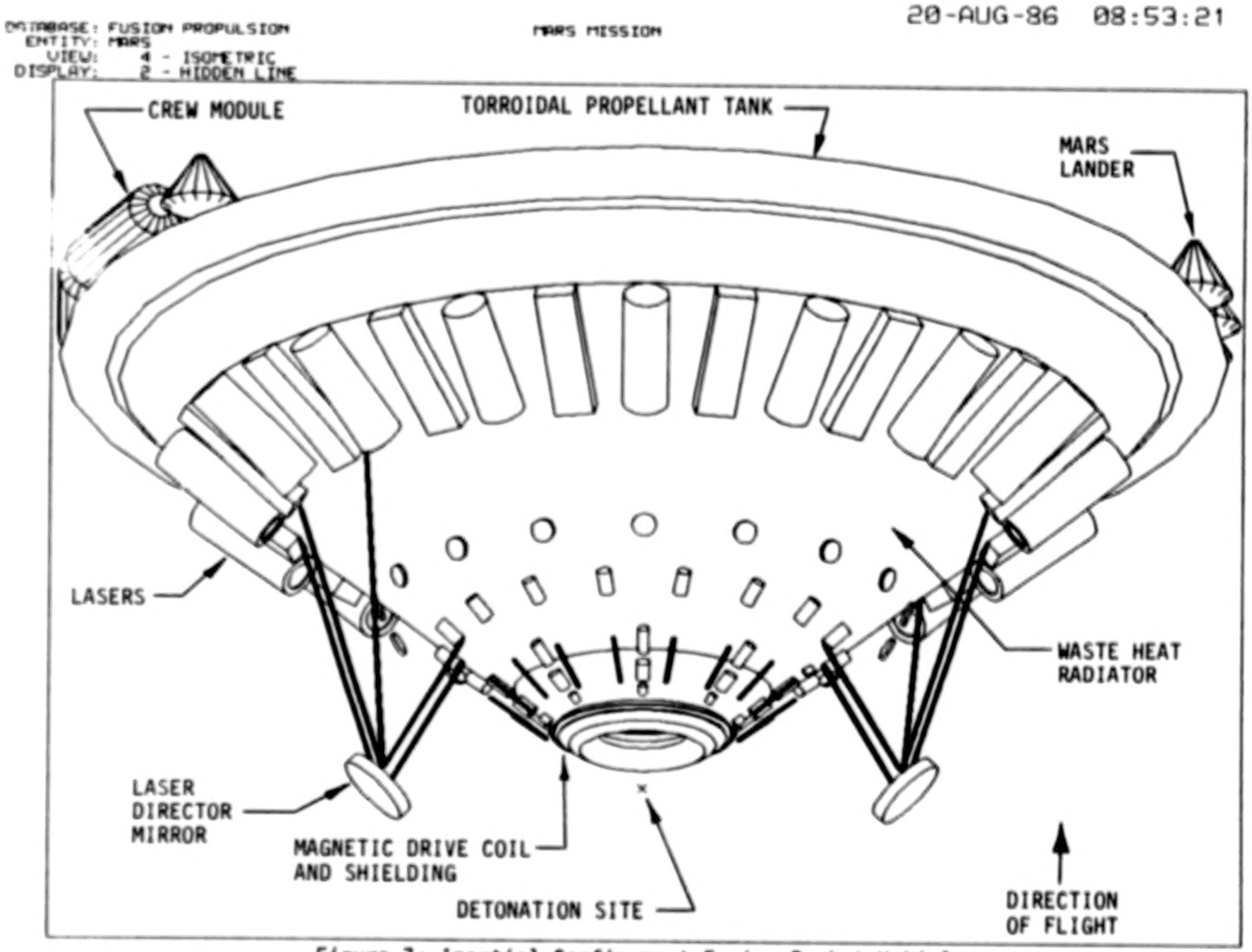
In 1986-1987, JPL participated in a study led by Charles Orth at Lawrence Livermore to develop a conceptual design for an Inertial Confinement Fusion (ICF) rocket named VISTA (Vehicle for Interplanetary Space Transport Applications, reference 11). The vehicle would use many high energy lasers mounted around the surface of the cone described above, and mirrors would direct the beams to a detonation site at the apex of the cone. Pellets of deuterium, tritium, and hydrogen expellant mass would be ejected on a trajectory to the detonation site, and once reaching the site, the bank of lasers would pulse fire at the pellet, imploding it to initiate a fusion reaction, creating a high energy plasma which would be directed by the drive coil's magnetic field.

Pellets would be ejected and imploded at a rate between 5 and 30 per second, resulting in a pulse-mode rather than a continuous propulsion system. It is estimated that such a system could achieve an acceleration of 0.02 g or greater, with a specific impulse of about 25,000 sec. Typical mission velocity changes are 100 km/sec or more.

The point design for the VISTA study was a manned Mars vehicle with a 45 day trip time to Mars for rendezvous (see Figure 7). The dry mass was 1,600,000 kg, excluding the payload of 250,000 kg which included crew accommodations, crew shielding, and lander craft. The propellant mass was 4,150,000 kg for a total loaded vehicle mass of 6,000,000 kg. Of this, the primary structure/radiator was estimated to be 200,000 kg.

A NASTRAN finite element structural analysis was performed by Rob Calvet at JPL for the primary structure. The shell structure was assumed to consist of titanium alloy heat pipes integrated together with local stiffeners and structural attachment points provided for mounting and thermally isolating various spacecraft components. It was determined that special shock mounting of the pulsed propulsion drive coil is not required. The mass of the drive coil itself and the compliance of the surrounding structure form a sufficient mass-spring momentum absorber to sharply attenuate the drive pulse peak loads. Several meters out along the shell, away from the drive coil, only an averaged acceleration is seen by the structure.

The mass area-density arrived at for the primary radiator/ structure was 14 kg/m$^2$. Stretching the current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MARS MISSION
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CREW MODULE
TORROIDAL PROPELLANT TANK
MARS LANDER
LASERS
WASTE HEAT RADIATOR
LASER DIRECTOR MIRROR
MAGNETIC DRIVE COIL AND SHIELDING
DETONATION SITE
DIRECTION OF FLIGHT

Figure 7: Inertial Confinement Fusion Rocket Vehicle

state of the art, radiators are envisioned with mass area-densities as low as 1.5 kg/m$^2$. Although idealized stresses in this thin of a structure are probably still acceptable, one must be concerned about stiffening for local buckling, structural stiffness (first mode frequency) for stability and control, meteoroid protection, and manufacturability, among other things. For mass estimation purposes for this type of vehicle configuration, one should probably use a value of 10 kg/m$^2$ for the primary radiator/ structure.

**Antimatter Propulsion**

Matter/antimatter annihilation may someday promise even higher performance vehicles than fusion propulsion. Rather than imploding D-T pellets to create a high energy plasma, protons and antiprotons would be placed in contact at the detonation site. Besides resulting in a more efficient conversion of matter into energy, the lasers and mirrors required for the VISTA concept would be eliminated (reference 12). Also the waste heat radiator requirements of the lasers are eliminated, although large radiator surfaces are still required for the magnetic drive coil shield. Cooling must also be provided for the energy system which powers the drive coil. This system could either extract energy from the plasma through induction, possibly using the existing drive coil, or use waste heat for a thermodynamic cycle engine.

John Callas at JPL has performed monte carlo particle interaction analyses for several configurations of antimatter rockets, one of which (the beam-core) is similar to the configuration described here (reference 13). The analysis considered numerous loss mechanisms, including gamma ray losses and magnetic "mirror" losses. His analysis is for a system with no additional expellant mass added to the annihilation reaction. This system is expected to result in a conversion efficiency of annihilation energy into useful propulsion of under 20%; however, the annihilation energy is of a considerable magnitude.

Figure 8 depicts an interstellar vehicle which might use matter/ antimatter annihilation propulsion. The vehicle is quite large (about 1 km diam.), as is indicated by the Space Shuttle shown for scale. This size is representative of a single stage vehicle which might travel to the star Epsilon Eridani (10.7 light years) and stop there within a time span of 100 years. By placing a second smaller torroidal propellant tank partway up the conical shell structure, it might be possible to stage (jettison) the upper radiator area and larger propellant tank at the top halfway through the mission, retaining the lower section (with its smaller tank), and thereby improve performance.

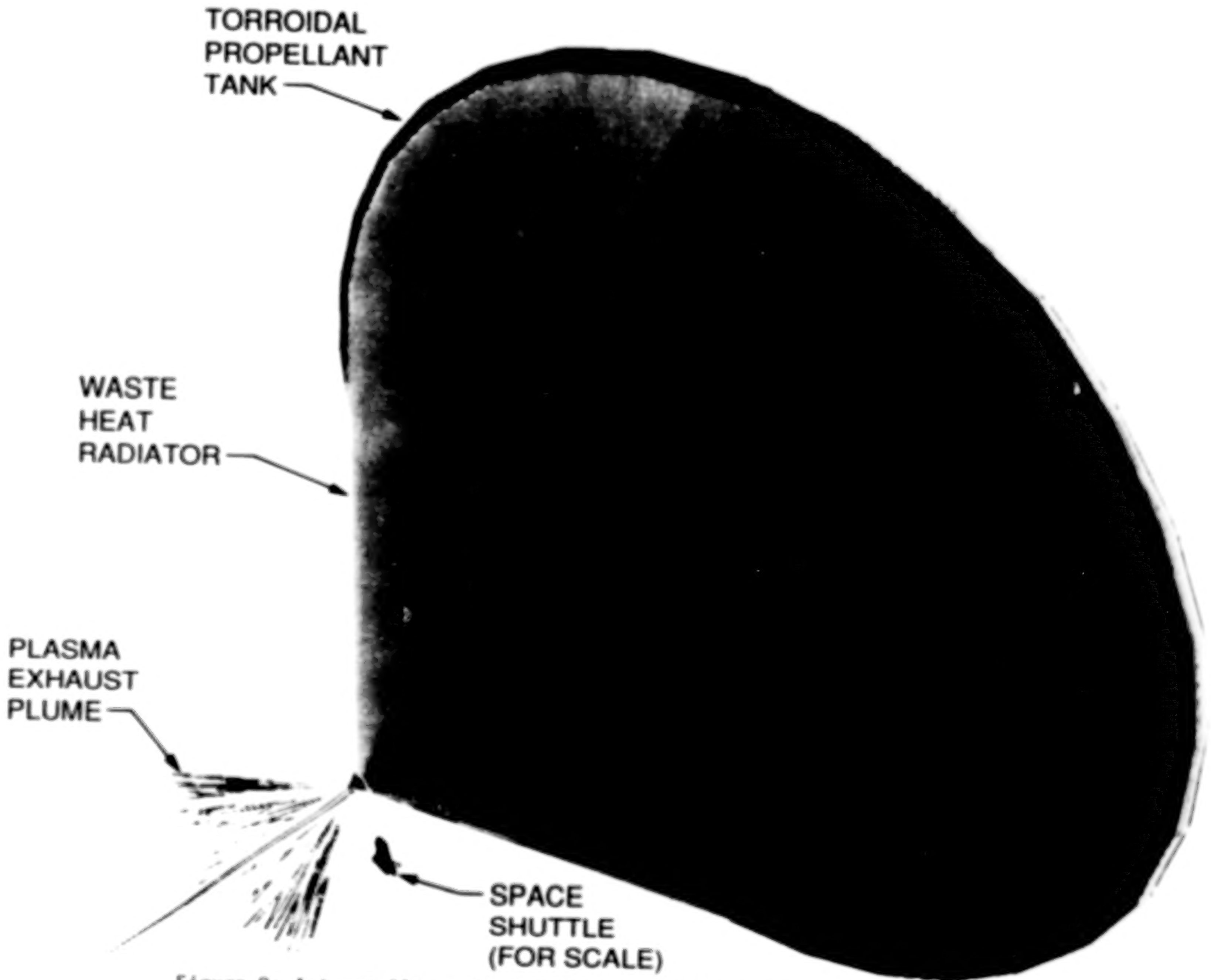
## EXTENDED SCALE SPACECRAFT

In the 33 years of the Space Age, spacecraft have ranged in size from 1.5 kg and 0.16 m (Vanguard 1) to 100,000 kg and 37 m (Shuttle orbiter and payload). The next century will see the range in scale of spacecraft expanded in both extremes.

**Micro Spacecraft**

Advances in microelectronics, Very Large Scale Integration (VLSI), and microrobotics will allow for the reduction in size and mass of spacecraft to an unprecedented degree. This will be useful for achieving higher velocities and faster trip times, putting more spacecraft on existing launch vehicles, allowing for smaller launch vehicles, or utilizing electromagnetic launchers.

In the early days of the U.S. space program, there were a few spacecraft that would qualify as micro spacecraft, such as Vanguard 1 and Pioneer 3/4. The latter vehicle, using 1958 technology, weighed only 5.9 kg and was only 0.2 m in diameter and 0.4 m tall, yet had a respectable imaging camera (among other scientific instruments) and could transmit data to Earth from beyond Lunar distances. The current generation of "light-sats" include AMSAT's Microsat which is 0.23 m on a side and weighs 10 kg.



TORROIDAL PROPELLANT TANK
WASTE HEAT RADIATOR
PLASMA EXHAUST PLUME
SPACE SHUTTLE (FOR SCALE)

Figure 8: Interstellar Vehicle Utilizing Matter/Antimatter Propulsion

Microspacecraft development at JPL (since the days of Pioneer 4) includes studies performed by Jim Burke in 1980 (reference 14) and more recent work by Ross Jones (reference 15). One recent small spacecraft concept studied by Kerry Nock and Ron Salazar is the Lunar GAS (Get Away Special, reference 16) shown in Figure 9. This vehicle would be deployed from a Shuttle GAS canister and utilize SEP to reach lunar orbit, starting from low Earth orbit.

In its ultimate limit, one might envision the microspacecraft as a monolithic silicon block with integrated circuits, solar cells, phased array antenna, and other spacecraft subsystems integrated together into one "chip". Recent advances in the fields of microscopic motors and microrobotics may lead the way to subminiature planetary rovers.

**Mega Spacecraft**

In the coming millennium, requirements will continue to grow for larger spacecraft such as space station complexes, tethered satellite platforms, space-based radar, large precision interferometers and segmented reflectors, solar power stations, solar sails, etc. Challenges will be presented in making these structures as light weight as possible, and in controlling their geometry, orientation, and pointing stability. Some of these platforms and vehicles will utilize fixed structures and passive damping, while others will require active structural elements and special control actuators dispersed throughout the structure to control its stability and/or allow it to achieve precision pointing control.

JPL is one of several NASA centers involved in a Control Structure Interaction (CSI) program to study and develop control techniques for such large structures. A major focus of JPL's work is a conceptual design for a large space-based interferometer to be used for high resolution imaging and precise astrometry (reference 17). This structure uses piezo-electric devices to control strut length and dynamic characteristics, proof mass actuators to control damping (in addition to passive damping elements), and voice coils and piezoelectric stacks to position optical elements.

As shown in Figure 10, the Focus Mission Interferometer (FMI) consists of a long box truss which supports three separate interferometer telescope systems for which positioning must be controlled to tolerances as small as a few nanometers. A separate metrology tower contains ranging and position measuring optical systems which provide feedback to the active control system. Although the FMI is only 26 m in length, the technology embodied in it can enable much larger actively controlled space structures to be built.

Large space complexes can also be enabled with tethered systems stabilized by gravity gradient, centrifugal, or other forces. One recent JPL study led by Dave Collins involves a Martian aeronomy subsatellite which would be deployed from and tethered to a Mars orbiter spacecraft, similar to the Shuttle tethered satellite system. Figure 11 depicts a small U.S. vehicle deployed from a Soviet orbiter in a cooperative effort.

## OTHER ADVANCED CONCEPTS

There are of course many other concepts for vehicles of the next millennium than those few presented here. Of immediate utility are aerocapture and aerobraking which use atmospheric drag to reduce propulsion requirements for orbit insertion and orbit lowering maneuvers. The savings in propellant mass are traded off against the mass of the aeroshield and the configuration and mass penalties resulting from having to fit within the aeroshield and meet its center of mass requirements.

The NERVA (Nuclear Engine for Rocket Vehicle Application) rocket engine stands at a high level of readiness for usage in interplanetary travel requiring large payloads or high velocities. Beamed propulsion concepts using lasers, microwaves, etc. may well find utility in the next century, as may space

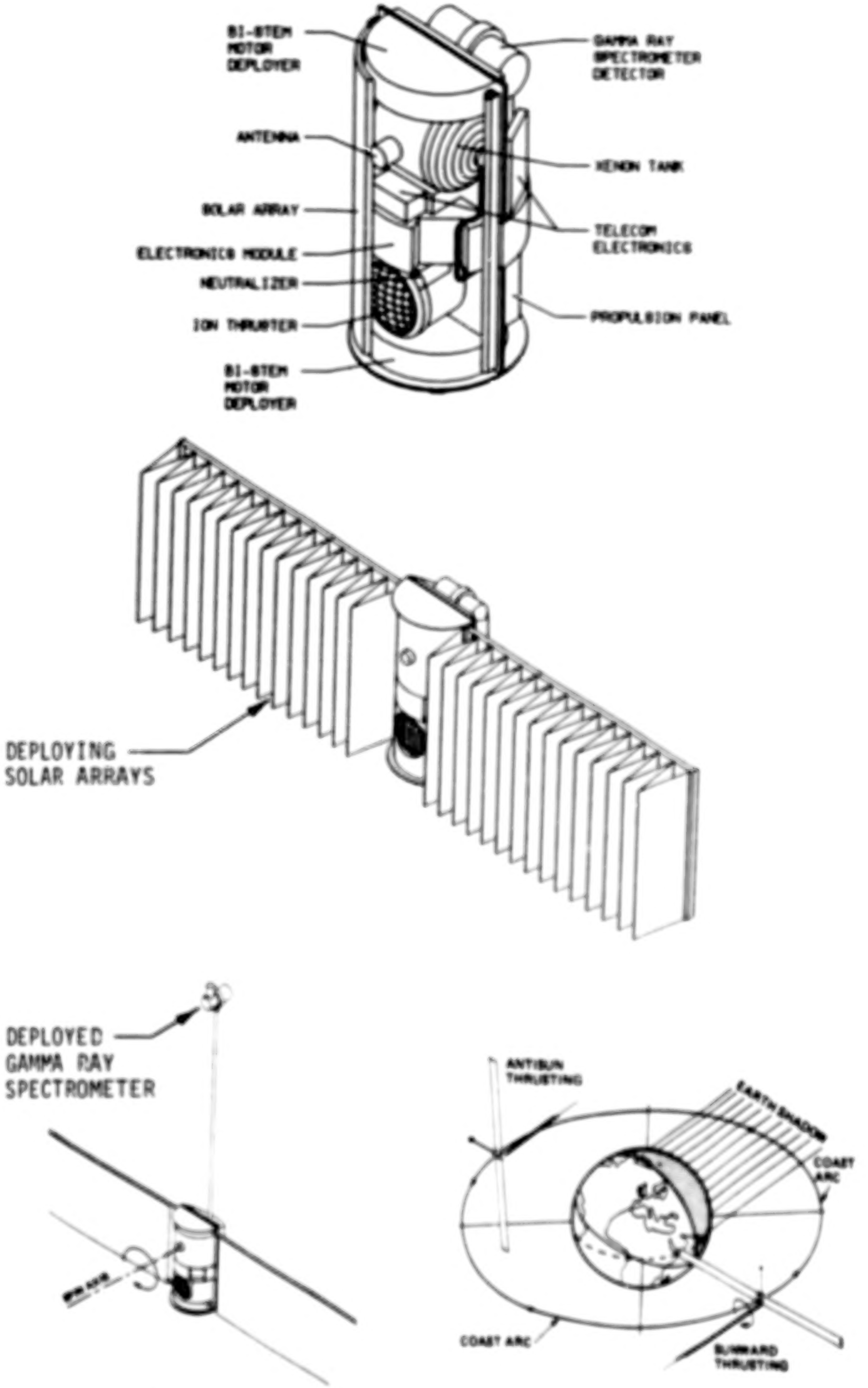

BI-STEM MOTOR DEPLOYER
GAMMA RAY SPECTROMETER DETECTOR
ANTENNA
XENON TANK
SOLAR ARRAY
TELECOM ELECTRONICS
ELECTRONICS MODULE
NEUTRALIZER
ION THRUSTER
PROPULSION PANEL
BI-STEM MOTOR DEPLOYER
DEPLOYING SOLAR ARRAYS
DEPLOYED GAMMA RAY SPECTROMETER
SPIN AXIS
ANTISUN THRUSTING
EARTH SHADOW
COAST ARC
COAST ARC
SUNWARD THRUSTING

Figure 9: Lunar "Get Away Special" Spacecraft

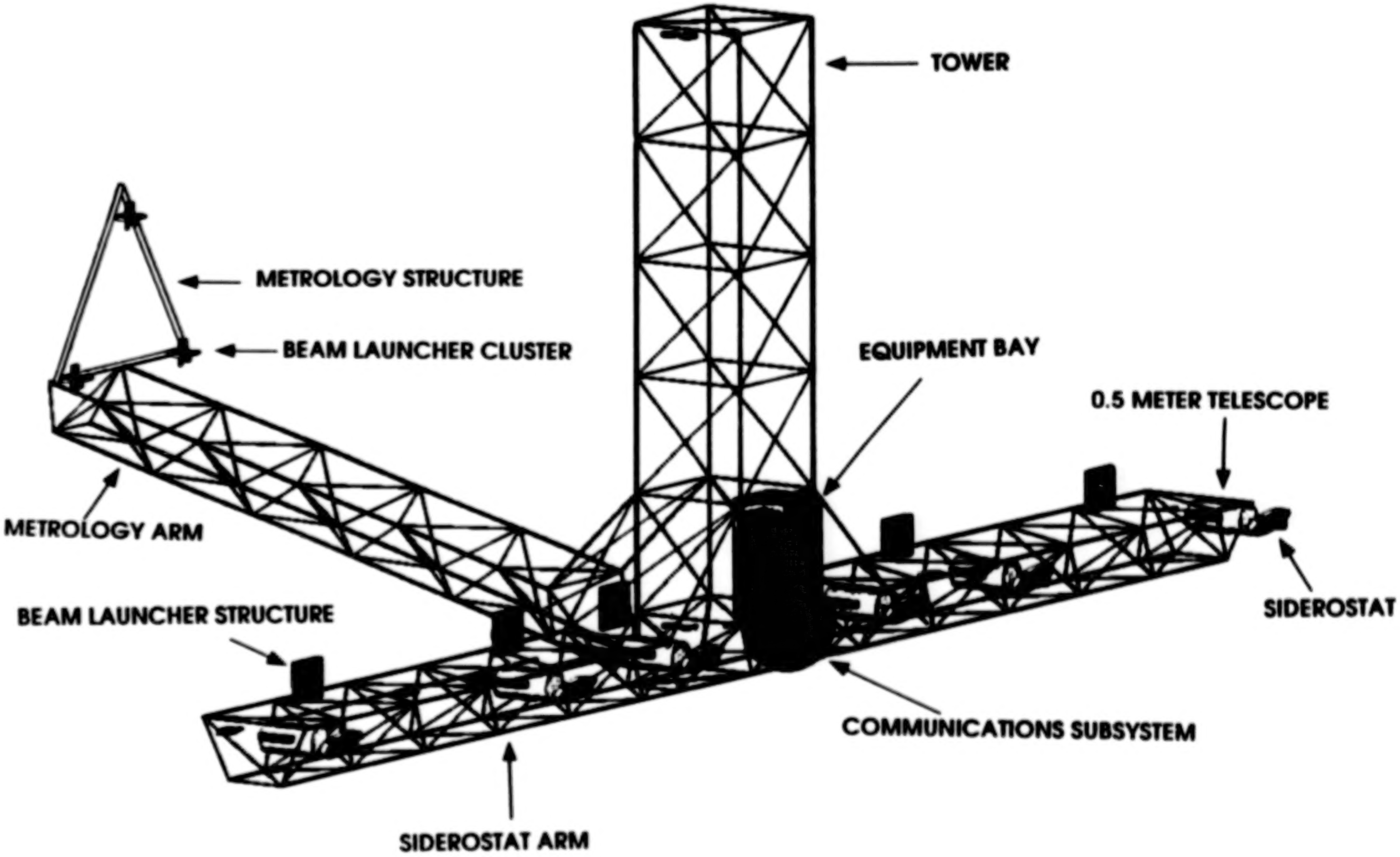

TOWER
METROLOGY STRUCTURE
BEAM LAUNCHER CLUSTER
EQUIPMENT BAY
0.5 METER TELESCOPE
METROLOGY ARM
SIDEROSTAT
BEAM LAUNCHER STRUCTURE
COMMUNICATIONS SUBSYSTEM
SIDEROSTAT ARM

FIGURE 10: THE FOCUS MISSION INTERFEROMETER

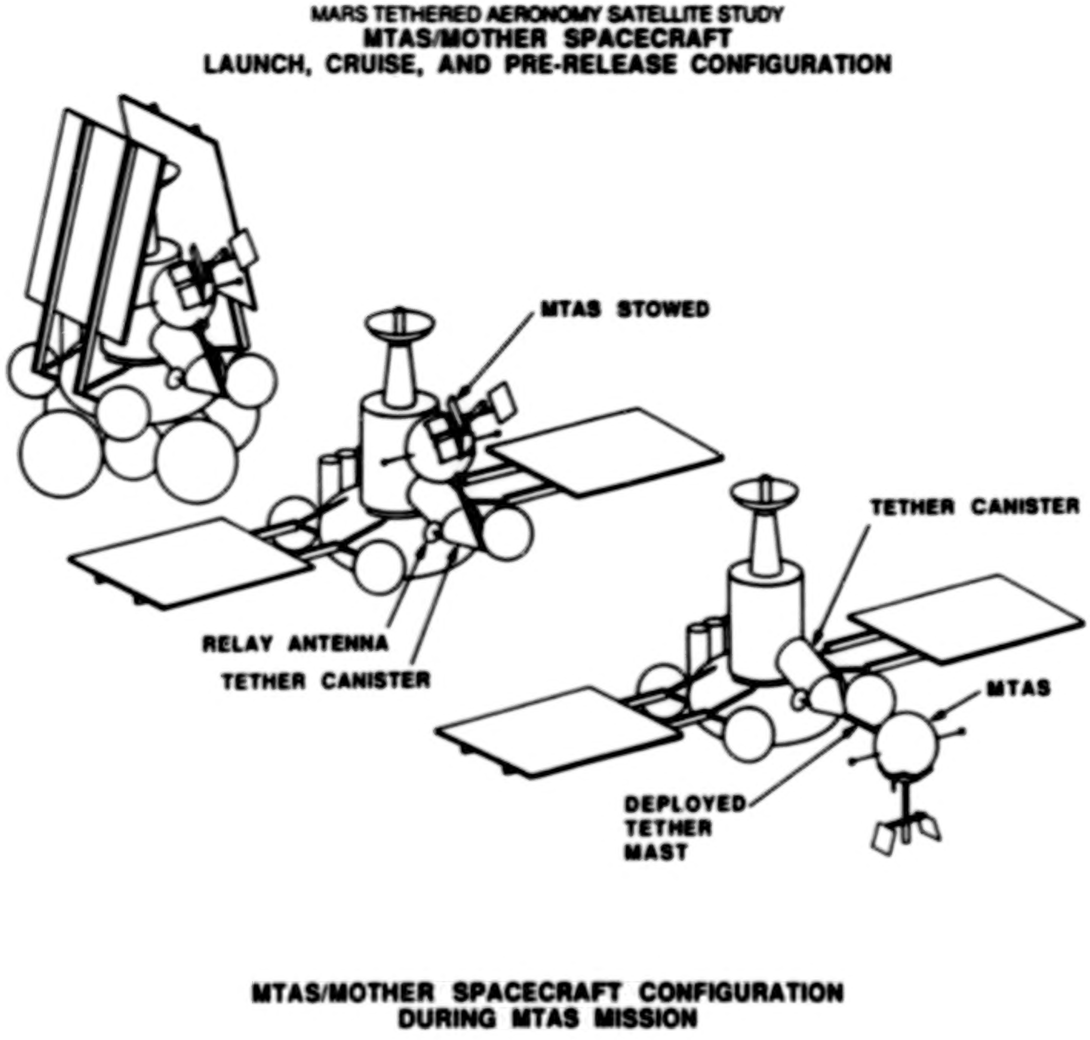

MARS TETHERED AERONOMY SATELLITE STUDY
MTAS/MOTHER SPACECRAFT
LAUNCH, CRUISE, AND PRE-RELEASE CONFIGURATION
MTAS STOWED
RELAY ANTENNA
TETHER CANISTER
TETHER CANISTER
MTAS
DEPLOYED TETHER MAST
MTAS/MOTHER SPACECRAFT CONFIGURATION
DURING MTAS MISSION
SUN
ORBITAL VELOCITY VECTOR
NADIR

Figure 11: Tethered Mars Satellite

or Earth based electromagnetic launchers. For achieving interstellar travel velocities, the Orion concept is probably a nearer term solution than either fusion or antimatter propulsion.

What will the advanced spacecraft of the next millennium look like? We can only make engineering judgments based on our imagination, our current understanding of physics, and extrapolations of technologies we are aware of today. On one hand we always fail to realistically estimate the performance losses and difficulties inherent in bringing abstract physical principles to practical engineering realization. On the other hand we always fail to imagine the unexpected physical discoveries and new technologies that the future will bring.

## ACKNOWLEDGMENT

The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

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N91-22169

# UNUSUAL SPACECRAFT MATERIALS

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## 0.0 ABSTRACT

For particularly innovative space exploration missions, unusual requirements are levied on the structural components of the spacecraft. In many cases, the preferred solution is the utilization of unusual materials. This trend is forecast to continue. Several hypothetical examples are discussed in this paper.

## 1.0 SAILS, SUNGRAZERS, AND ICESHIPS

For particularly innovative space exploration missions, unusual requirements are levied on the structural components of the spacecraft. Titanium and carbon-carbon have become materials of choice, the Lunar Excursion Module's thermal protection blanket was literally gold-plated, and so on. But this trend is likely to continue, straining the ingenuity of space engineering, construction, and operations capabilities.

To list three examples: (1) solar sail missions require extremely lightweight, reflective, flexible deployment of very large surface areas, as discussed by Clarke (ref. 3) and Anderson (ref.1); (2) Starprobe is a mission being considered at NASA-JPL which would send a spacecraft close to the sun, and future follow-on Sungrazer missions may go even further: into the corona of the sun, requiring structural stability at extreme temperatures; (3) unmanned interstellar probes, as outlined by Jaffe (ref. 5) and Mallove (ref. 9), powered by nuclear fusion require a minimum deadweight ratio (fraction of non-payload mass remaining after all fuel is expended) and minimum molecular weight of exhaust material.

In many cases, the preferred solution is the utilization of unusual materials. In the three examples above, solutions include: (1) monatomic layer ion sputtered deposition of metal on a substrate subsequently etched or sublimed away as described by K. Eric Drexler (ref. 4); (2) tungsten carbide, rhenium, and other extreme refractory materials for the structural components of Starprobe successors; (3) The fuel and the structural components of a fusion probe being one and the same. Among contenders, beryllium (molecular weight 8) is rather difficult to vaporize, leaving the prime choices as either water ice (molecular weight 18) or lithium (molecular weight 6) stiffened by boron (molecular weight 11) or carbon (molecular weight 12) fibers, being vaporized, ionized, and expelled as reaction mass until no structural components remain besides the payload and now-useless engine.

In the last case, recent analysis by J. B. Stephens et. al. at NASA-JPL and this author, as popularized by Moser (ref.12), suggests that even hydrogen ice (molecular weight 2) can be stiffened by admixture of fibrous or particulate material far beyond its normal pliability -- about the same as butter. Hydrogen ice can also be adequately protected against sublimation by very modest insulation. A one meter radius sphere of Hydrogen ice, insulated by a centimeter of low-density Hydrogen ice fluff and one centimeter of layered reflector, can last ten years in Earth orbit.

Each of these cases, all feasible within the next 30 years, raises further material-related concerns.

(1) What is the best metal for solar sail deposition in strength/weight tradeoff of ultra-thin films, and to what extent can weight be further reduced by perforation with holes smaller than the wavelength of light? What is the ideal disposable substrate? How can enormous numbers of microscopic holes best be engineered? Aluminum has been recommended, but additional analysis is required.

Dr. Robert L. Forward assumed aluminum could be used for a laser-propelled round-trip interstellar lightsail (ref. 7). But, as Geoffrey A. Landis pointed out (ref. 10), achievable thrust is limited by the maximum acceptable temperature that the sail attains by inadvertant absorption of light.

Landis ignores reflectivity and high-temperature tensile strength for sail metals, using a figure of merit of melt temperature divided by density, normalized for aluminum (T = 940°K, density = 2.7 grams/$cm^3$) as Al=1. Other metals have superior figures of merit, including boron (3.6), beryllium (2.8), scandium (2.1), and titanium (1.5). Richard Feynman (ref. 6) pointed out that scandium was hitherto unique among elements in having NO known useful applications. Beryllium and scandium are also of perversely low relative abundance in the Solar System -- see the zig-zag graph in New Scientist, (ref. 13). Since beryllium (T = 1550°K, density = 1.8 grams/$cm^3$) is also very reflective, Landis considers it the best metallic sail material.

Provocatively, Landis compares transparent nonmetallic dielectric materials, which can be deposited as quarter-wavelength thin-film reflectors. Forward suggested an 8-layer stack alternating diamond [T > 3500°K] with vacuum, as per Forward (ref. 8). Materials with good figures of merit include diamond (2.41), silicon carbide (2.65), zinc sulfide (2.35), tantalum pentoxide (2.5), zirconium dioxide (2.15), and silicon dioxide. Furthermore, if the incident laser light is infrared or ultraviolet instead of visible, then silicon (3.5 at 1200 nm) and lithium fluoride (1.45 at 130 nm) become viable candidates. silicon carbide is judged to be best, but any of these materials offers accelerations "orders of magnitude higher than those achievable with metal films":

Diamond, SiC, ZnS, $Ta_2O_5$, $ZrO_2$, $SiO_2$, Si, LiF

Diamond thin-film can be made by glow-discharge decomposition of methane; tantalum pentoxide, zirconium dioxide, and zinc sulfide are commonly deposited as thin-film for optical coatings by electron beam evaporation; silicon carbide is currently grown by epitaxial deposition on silicon, with the silicon substrate capable of being "etched away to leave a free-standing film ... eminently suitable for a small-scale demonstration" according to Landis (ref. 10). It isn't clear if LiF would make a suitably strong free-standing film.

The solar sail concept represents a spacecraft which needs no fuel, as fictionalized by Clarke (ref. 3), and has historical/ political significance as shown by

Anderson (ref. 1) and Uphoff and Post (ref. 18). Refinements to the concept give it an almost poetic elegance, as in the unique presentation co-authored by Ray Bradbury and this author (ref 2).

(2) What are the optimal refractory materials for Starprobe successors, and how can they best be combined and engineered? Some particular materials can be easily manufactured from in situ regolith on the lunar surface:

$Al_2O_3$, $CaO$, $MgO$, $TiO_2$, $SiO_2$, $Ti_5Si_3$, $Cr_2O_3$, $K_2TiO_3$

Problems remain to be solved. What are the optimal mixtures of these materials (and others, such as the spinels)? Is the moon an appropriate site for materials extraction and processing for near-sun spacecraft components? How may passive refractory insulation and dynamic cooling best be combined? What is the optimum refractory geometry? Robert Waldron (ref. 19) suggests a sharply tapered wedge pointing toward the sun for optimal combination of reflection angle and radiator surface area.

(3) What are the structural limits of fiber-stiffened water ice, hydrogen ice, and lithium? Is it preferable to filter out the fibers, adding them to the deadweight, or to add engine weight to allow them to also be ionized and added to the exhaust? And, in the case of a water-ice fusion spacecraft, can we legitimately refer to this as the ultimate steamship?

## 2.0 HYDROGEN ICESHIP: DETAILED EXAMINATION

This section of the paper concentrates on the hydrogen ice spacecraft concept, and consists of an introduction, thermal analysis, experimental results, conclusions, and suggestions for future research.

Much of this analysis was contributed by James Salvail, of SETS, Inc. Honolulu.

Rather than having astronauts perform extravehiclular activity to chip off chunks of hydrogen ice for fuel, we visualize a spacecraft made of a structural cluster of hydrogen ice spheres which can be robotically detached, one at a time as needed, and melted or slushified in a conventional fuel tank. I have expounded on this at greater length, with an extended bibliography on hydrogen ice and slush in another paper (ref. 15).

### 2.1 Hydrogen Iceship: Introduction

Visualize a bunch of grapes. Replace each grape with a cocktail onion. Our configuration for a hydrogen ice spaceship (the bunch) can be modelled as a cluster (connected to a common stem) of concentric metal spheres (onions) each of which has hydrogen ice (onion pulp) filling the spaces between its spherical surfaces (onion skins).

Most of the mass of the spacecraft is composed of these spheres, with the mass of the engine, avionics, payload, and so on comprising a much smaller fraction. The spheres are designed to provide some structural capability and to maintain fuel in solid form until needed; then they are liquefied or slushified for propellant usage. This is conceptually similar to the scene in the Marx Brothers' film "Go West", where Groucho, Harpo, and Chico feed the furnace of a steam locomotive with boxcar slats, and then demolish and burn the caboose.

Concentric spheres (within a single "onion") are connected to each other by at least two rods made of a material that has very low thermal conductivity, such as hard rubber. This is necessary so that the spheres above the instantaneous level of the subliming ice surface do not move relative to each other. The outer shells are made of a highly reflective material, such as aluminized (more easily ionized in propulsion: lithium-ized) mylar, thick enough to provide reasonable structural integrity. The inner spheres are made of the same materials, but much thinner (<<0.1 cm), as they are merely radiation shields.

The radiation shields and outer hulls must contain enough sufficiently sized holes or pores so that sublimed hydrogen molecules are quickly lost into space. The evacuated spaces between the slowly receding ice surface and the outer hulls thus have negligable gaseous heat conduction because the gas is very rarified. Gas flux is small enough (barring close flyby of the sun, nuclear explosions, or laser heating) that heat convection is also negligible. Under the listed abnormal operating conditions, gaseous conduction/convection would still be much smaller than radiative heat transfer.

The effects of varying sphere radii, radiation shield number/spacing, outer hull albedo, and external environment are investigated to optimize the design for cost, size, and lifetime.

## 2.2 Hydrogen Iceship: Thermal Analysis

Thermal analysis consists of temperature calculations for outer hull, radiation shields, fixed radii from centers, and (crucially) at ice surfaces. Hull and shields are sufficiently thin and heat conductive as to be effectively isothermal through their thicknesses.

The energy balance at the outer hull consists of incoming solar radiation, emitted radiation from both sides, incoming radiation from the adjacent lower surface, and downward gaseous heat conduction. The two sides of the outer hull have, in general, different albedos and emissivities (i.e. painted black externally for visual stealth, as the low temperature already offer IR-stealth). These effects are described by:

$$E_m Sb\, T_{i,j}^4 + E_b Sb\, T_{i,j}^4 - f_b (1-A_m) E\, Sb\, T_{i-1,j}^4 - K \frac{dT}{dr} = \frac{(1-Ab)Sc}{4R^2}$$

Where $E_m$ is the emissivity of the natural metallic surface, $E_b$ is the emissivity of the outer surface (possibly black), unsubscripted E is the emissivity of the adjacent lower surface. If the lower adjacent surface is a radiation shield, then $E=E_m$. If the lower adjacent surface is the ice surface, then $E=E_h$, the emissivity of the ice. $A_m$ is the albedo of the natural metallic surface. T is temperature, with subscripts i and j denoting depth and time. $f_b$ is a geometric factor accounting for the smaller area of the adjacent inner surface. Sb is the Stefan-Boltzmann constant. Sc is the solar constant at 1 AU (Astronomical Unit). $K_g$ is the gaseous thermal conductivity of hydrogen. r is the radial coordinate (from the center of the ice sphere). R is the heliocentric distance in AUs. The factor 4 in the right hand solar insolation term reflects the assumption of rapid rotation. Constant values are given in section 2.3. This equation, and a related one for the moving ice surface (with an energy balance including upward radiation from the ice surface, downward radiation from the adjacent metal surface, heat conduction into the ice, and the latent heat energy due to sublimation) are the basis for the results of section 2.3.

## 2.3 Hydrogen Iceship: Computer Simulation Results

Computer simulation based on the preceding analysis calculated temperature at outer surface, radiation shields, surfaces and interior of the hydrogen ice. Ice surface temperature allowed derivation of hydrogen gas flux and radial position of the receding ice surface as a function of time, and thereby deriving hydrogen ice lifetime. Various runs determined the effects of radiation shields, outer albedo, and outer hull radius, in normal conditions and in a simulated nuclear blast. Kirchoff's law was assumed for metallic and ice surfaces. Parameter values are as listed below:

| Property/Parameter | | Value |
|---|---|---|
| $A$ | Vapor pressure constant | $6.17 \times 10^{9}$ dynes/cm$^2$ |
| $B$ | Vapor pressure constant | 149.44 °K |
| $A_m$ | Albedo of metal | 0.95 |
| $A_h$ | Albedo of hydrogen ice | 0.65 |
| $E_m$ | Emissivity of metal | 0.05 |
| $E_h$ | Emissivity of hydrogen ice | 0.35 |
| $C_h$ | Specific heat of hydrogen ice | $2.7 \times 10^{7}$ erg/g-°K |
| $D_h$ | Density of hydrogen ice | $7.06 \times 10^{-2}$ g/cm$^3$ |
| $K_h$ | Thermal conductivity of ice | $1 \times 10^{7}$ ergs/cm-s-°K |
| $H_s$ | Latent heat of hydrogen ice | $6.2 \times 10^{9}$ ergs/g |
| $S_c$ | Solar constant at 1 AU | $1.3928 \times 10^{6}$ ergs/cm$^2$-s |
| $T_i$ | Initial temperature of ice | 5°K |
| $M$ | Molecular weight of hydrogen | 2.0 g/g-mole |
| Sb | Stephan-Boltzmann constant | $5.6 \times 10^{-5}$ ergs/cm$^2$-s-°K |
| $R_u$ | Universal gas constant | $8.315 \times 10^{7}$ ergs/mole-°K |

First, a 1 meter radius sphere with 50 radiation shields spaced 2 centimeters apart and natural metallic surface was simulated. The ice remained nearly isothermal at the initial temperature of 5°K, with a negligible temperature gradient and a near-constant mass flux of 17.8 nanograms/cm$^2$-sec. The outer hull remained at a temperature of 236°K at 1 AU from the sun. After a simulated 10 years, the sphere had shrunk to 21 cm in radius, and the total lifetime was roughly 12 years.

Reducing the radiation shields from 50 to 10 had no effect. Painting the outer surface black (albedo = 0 for stealth) gave a tripled mass flux of 53.8 nanograms/cm$^2$-sec, a surface temperature of 5.2°K, and a reduced lifetime of 4.2 years.

At 0.1 AU from the sun a 50-shield 1 meter shiny sphere stays at 5.81°K, with a mass flux of 1.06 micrograms/cm$^2$-sec, and a lifetime of 75 days. With 10 shields, a 1 meter shiny sphere stays at 6.39°K, with a mass flux of 10.5 micrograms/$cm^2$-sec, and a lifetime of 35 days. Hence radiation shields are important for larger thermal loads, such as would occur if the hydrogen iceship mission began with a gravity assist swingby close to the sun.

The thermal effects of nearby nuclear detonation were simulated as a temporary change in heliocentric distance from 1.0 AU to 0.01 AU (where the radiative equilibrium temperature for a black body is 2808°K) for 20 seconds. If the outer metallic coat doesn't melt at the maximum temperature attained (2361°K), then the hydrogen ice adjacent to the outer surface peaks at 8.73°K, with a gas flux of 4.7 milligrams/$cm^2$-sec, decreasing after 10 minutes to 5.85°K (ten shields) or 5.79°K (fifty shields), at which time the ice has receded 2.5 cm.

All other things being equal, the lifetime of a hydrogen ice sphere was found to be directly proportional to the first power of the the initial radius. Thus, a 2 meter radius sphere has a 24 year lifetime at 1 AU, and 2 years at 0.1 AU. For deep space missions, loss becomes rather small for spheres several meters in radius.

Similar thermal analyses have been performed by Hustvedt for slab and cylindrical geometries (ref. 9).

## 2.4 Hydrogen Iceship: Summary of Concepts

The greatest advantage for a hydrogen ice spacecraft is obtained if the craft is an unmanned monolithic composite solid cryogen with embedded insulation and superconducting avionics. As disclosed by J. Stephens at JPL (who generated the original concepts in 1984 and 1985, while in communication with this author), the primary intellectual properties for patent purposes are:

(1) Ice embedded insulation,
(2) Vapor cooled insulation,
(3) Isomer conversion catalyst integral with insulation (activated carbon),
(4) IR photon reflective and vapor conductive insulation (variable mesh cloth multi-layer insulation),
(5) Vapor cast crystalline hydrogen ice using nuclear magnetic resonance heating of non-crystalline ice,
(6) Self-forming filamentary insulation from dispersed particles in the ice that cohere due to ice cleaning.

Attributes of the primary intellectual properties are:

(1) Unitized design; ice is the cryogen, structure, propellant, shielding, absorber, power source, window, and insulation support during launch,
(2) Superconducting temperature cryostat (<5 °K for Hydrogen),
(3) Self-insulating solid cryogen,
(4) Long lifetime in earth orbit,
(5) Low cost materials (<$10/lb),
(6) Low cost fabrication (casting process),
(7) Low launch cost (withstand high acceleration forces),
(8) Low cost operation (efficient superconducting solid-state system),
(9) Acoustically quiet (no moving parts) so good for very accurate optics or interferometry,
(10) Thermally stable (large thermal capacity well insulated),
(11) High density ice vapor cast and used at same temperature avoiding shrink stresses in insulation and components embedded in ice.

Ancillary intellectual properties enabled by the primary concepts are:

(1) Vapor cooled refractory insulation,
(2) Neutron absorbing cryogen (Hydrogen),
(3) Microwave absorbing ice/insulation,
(4) Microwave reflecting ice/insulation.

The advantages of the ancillary concepts are:

(1) Laser tough shielding,
(2) Neutron tough shielding,
(3) Neutral and charged particle beam tough shielding,
(4) Radar stealth,
(5) Superconducting phased array radar.

Concepts enabled by Cryostat primary and secondary concepts are:

(1) Propulsion and electric power systems:
    (a) Solar powered ion rocket and superconducting magnet power generator and storage system,
    (b) Magnetohydrodynamic detonation wave ion rocket and superconducting magnet power generator and

        storage system (detonate layers of solid
        oxygen alternated with solid hydrogen);

- (2) Guidance and control:
    - (a) Superconducting computer,
    - (b) Superconducting gyroscope,
    - (c) Superconducting magnet attitude control,
    - (d) Superconducting radio and antenna;
- (3) Launch forces resistant structure:
    - (a) fiber reinforced composite ice;
- (4) Remote sensing synersensory systems in phased arrays in Cryostats orbiting in formation:
    - (a) Synthetic aperture superconducting phased array radar,
    - (b) Synthetic aperture superconducting phased SQUID array Magnetic Anomaly Detection,
    - (c) Synthetic aperture superconducting phased SQUID array Gravimeter,
    - (d) Synthetic aperture superconducting phased array Altimeter,
    - (e) Blue-Green synthetic aperture superconducting phased array lidar,
    - (f) Thermal IR telescope spectrometer.

This astonishingly rich set of concepts of Jim Stephens is only moderately challenged by demands of the interplanetary or interstellar regime, as opposed to the near-Earth applications originally envisioned.

Individual hydrogen ice spheres can be cheaply orbited by small boosters (or electromagnetic propulsion, most recently discussed by Wallich, ref. 20), and later assembled into a large spacecraft. Solid hydrogen is inherently safer than liquid hydrogen.

The spheres can have embedded avionics, providing distributed redundant capability for the spacecraft at superconducting temperatures.

Once assembled, the low accelerations typical of an ion, fission, or fusion propulsion would not endanger the inherently low tensile strength of hydrogen ice as a structural material. The hydrogen ice spheres would be between the crew (or payload) and the nuclear propulsion, providing neutron-absorbant shielding at no extra cost.

Space exploration applications include:

(1) Sungrazer,

(2) Outer planet explorer,
(3) 1000 AU mission (TAU),
(4) Subterranean radar mapping of planets,
(5) Manned Mars Mission,
(6) Propellant transfer and storage for Space Station refueling depot (ref. 15),
(7) Interstellar mission.

## 2.5 Hydrogen Iceship: Future Research

Future areas of hydrogen iceship analysis include:

(1) Comparisons of fiber-stiffened water ice, carbon dioxide, lithium, or other alternatives to hydrogen ice;

(2) Experimental determination of strength, stiffness, etc. for hydrogen ice with various fiber compositions (boron, carbon);

(3) Structural design optimization for various types of propulsion system;

(4) Exploration of the concept of detonation wave propulsion/attitude control with alternating layers of hydrogen ice and oxygen ice, or solid ozone with an ISP of 494 according to Stwalley (ref. 16);

(5) Sensor capabilities of phased arrays of embedded cryogenic detectors in a fleet of coorbiting iceships;

(6) Relativistic kinematics of multi-staged interstellar iceships, following the mathematical treatments of Jaffe (ref. 5), and other papers as compiled by Mallove (ref. 9);

(7) Extension of hydrogen ice concepts to hydrogen metal, which might remain solid at low pressures and have an ISP in the thousands, taking into account results announced by Peterson (ref. 14), Stwalley (ref. 16), and Thierschmann (ref. 17).

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N91-22170

# MALEO: MODULAR ASSEMBLY IN LOW EARTH ORBIT. A STRATEGY FOR AN IOC LUNAR BASE.

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Abstract
Modular Assembly in Low Earth Orbit(MALEO) is a new strategy for building an initial operational capability lunar habitation base. In this strategy, the modular lunar base components are brought up to Low Earth Orbit by the Space Transportation System/Heavy Lift Launch Vehicle fleet, and assembled there to form the complete lunar base. Modular propulsion systems are then used to transport the MALEO lunar base, complete and intact, all the way to the moon. Upon touchdown on the lunar surface, the MALEO lunar habitation base is operational. An exo-skeletal truss superstructure is employed in order to uniformly absorb and distribute the rocket engine thrusting forces incurred by the MALEO lunar base during translunar injection, lunar orbit insertion, and lunar surface touchdown. This conceptual paper discusses the components, configuration, and structural aspects of the MALEO lunar base. Advantages of the MALEO strategy over conventional strategies are pointed out. It is concluded that MALEO holds promise for lunar base deployment.
Keywords: Modular Assembly, Low Earth Orbit, Lunar Bases, Space Architecture, Truss Superstructure, Lunar Landing System(LLS), Space Station Freedom(SSF-1), Space Transportation System(STS).

## 1 Introduction

A phase 1 lunar habitiation base(LHB-1) is conceived as the first permanently manned facility on the lunar surface. The facility will provide a test bed for extended habitation, exploration, and scientific investigation.

It is considered somewhat analogous to a forward base camp in the Antarctic or on Mt. Everest. LHB-1 will also form the nucleus, if needed, for further expansion and experimentation, which might be necessary during the evolution of what might become the first fully self-sustained, permanently manned lunar colony. This lunar colony will support spacecraft operations in cislunar as well as interplanetary space by providing propellants and other material manufactured on the lunar surface. (NASA,1988)

The first priority in a phase 1 extended duration mission of this nature, is the provision of an assured safe haven for the astronaut crew which will alleviate astronaut anxiety associated with build-up operations, followed by a comfortable environment within the facility, which will enhance crew productivity.

## 2 Development of the MALEO Strategy

Space station-like modules are considered the major components which need to be assembled in order to form a phase 1 lunar habitation base(LHB-1) (Duke et al,1988). Conventional strategies suggest launching these modules separately from the Earth, landing them one at a time on the lunar surface, and assembling them there, using robots and astronauts, in extra-vehicular activity(EVA)(Alred et al,1989). Precursor missions are employed to land crew and assembly equipment. If Earth based robotic teleoperation is suggested for assembly operations, a rather complex cislunar satellite tele-communication network is required before automated assembly operations could commence [Figure 1].

At the 1988 inaugural summer session of the International Space University(ISU), held at MIT in Cambridge, Massachusetts, M.Thangavelu, a graduate student in Building Science from the University of Southern California(USC), proposed the idea of assembling the components of the lunar base in Low Earth Orbit. Modular Propulsion systems would then be employed to transport the entire base, complete and intact, directly to the lunar surface, for immediate occupation by the astronaut crew [Figure 2]. The strategy thereby avoids the cost and risk associated with manned extravehicular activity on the lunar surface, which had been proposed in several earlier concepts(Johnson, 1985). Thangavelu along with another ISU student, G.E. Dorrington from Cambridge University, presented the idea in a paper entitled " MALEO: Module Assembly in Low Earth Orbit. Strategy for Lunar Base Build-up" at the 39th Congress of the International Astronautical Federation, in October 1988, held in Bangalore, India(Thangavelu, 1988). M. Thangavelu continued to work upon this idea at USC. It became the focus of his masters thesis in Building Science(Thangavelu,1989).

| | |
|---|---|
| Several ETO Launches | Several ETO Launches |
| Several componentwise TLIs & LOIs | Assembly in LEO |
| Several lunar landings | One or two TLIs and LOIs |
| Assembly on lunar surface | One lunar landing |
| Substantial precursor missions | Minimum precursors |

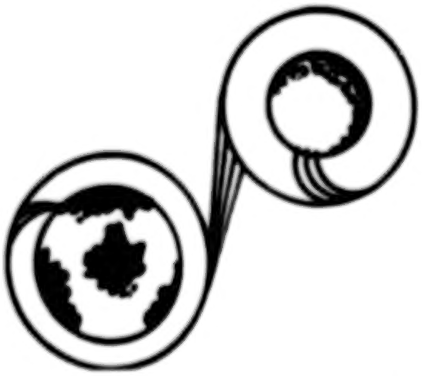

Figure 1.Lunar Surface Assembly

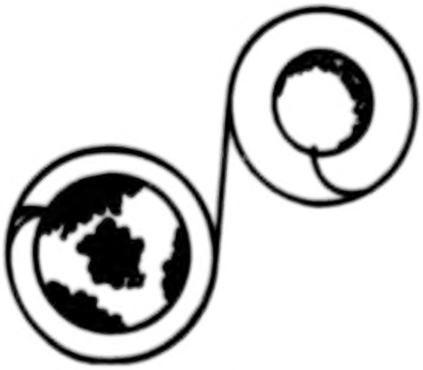

Figure 2. The MALEO Strategy

ETO - Earth to Orbit, TLI - Translunar Injection, LOI - Lunar Orbit Insertion

## 3 Configuration of the Lunar Habitation Base-1(LHB-1)

Several configurations are possible for the design of LHB-1, and two or more modules might be employed, as required by the mission objective[Figure 3]. In the past, horizontal and vertical configurations have been proposed for lunar habitation bases(Johnson,1985). In the MALEO strategy, the horizontal configuration was preferred to the vertical one for the following reasons.

1. Commonality with space station Freedom enhances design and engineering economy.
2. Commonality enhances crew adaptation and productivity.
3. Better and larger work spaces for extended duration missions, which minimizes circulation spaces.
4. Wider footprint for better landing stability.
5. Ease of expansion during evolution by attaching additional horizontal modules.

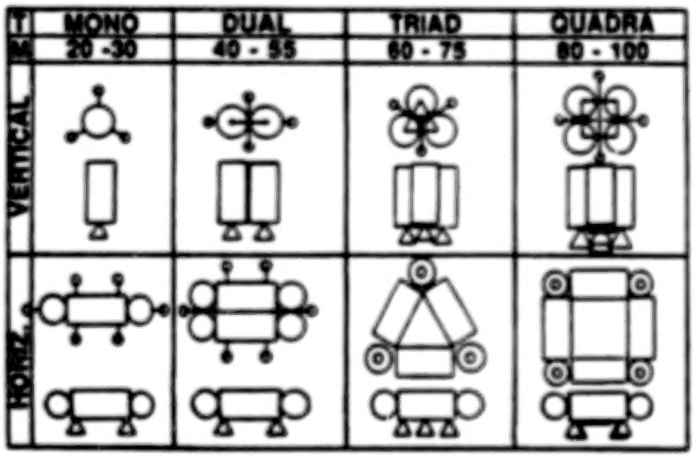

Figure 3. Schematic Lunar Base Module Configuration Study

Using space station-like modules as a base line, it may be possible to contain all the necessary systems required for the phase 1 LHB-1, for a crew of four astronauts / mission specialists, in three modules and three nodes. They would be interconnected in a triangular closed loop configuration which will facilitate dual egress for the astronaut crew. This triangular configuration also enhances structural stiffness by virtue of its geometry. This configuration is less susceptible to twisting moments than other optimal configurations and hence would transmit the least bending moments to the connecting joints of LHB-1 in the event of a lunar landing on uneven terrain.

## 4 The components of the Lunar Habitation Base-1(LHB-1)

The major components of LHB-1 are the three modules and three nodes. They constitute the manned core of LHB-1 and are described below.[Figure 4]

1. The habitation module houses the four astronauts/mission specialists, and contains crew sleeping quarters, a gymnasium/recreation facility, and a galley.

2. The sanitation/hygiene node is conveniently located adjacent to the habitation module and besides those functions will also house water recycling and solid waste management systems.
3. The laboratory module is also adjacent to the sanitation / hygiene node. This module is equipped with interchangeable racks which are partially outfitted, so that modifications and new setups are possible, as the base evolves.
4. The primary EVA node is the airlock which is used for all of the extra-base activity. It contains the EVA suits, and equipment to prevent regolith back-tracking and contamination.
5. The power and logistics module contains the power generation, storage, and regulation equipment. The module contains solar power panels which are deployed externally, can accomodate a nuclear power source that could be deployed externally.
6. The ECLSS node is also the command center of LHB-1 and is designed so that the base may be extended by attaching additional modules, if necessary.

**MALEO LHB-1 MASS SUMMARY**

| | | |
|---|---|---|
| HABITATION MODULE | 15-17.5 | MT |
| LABORATORY MODULE | 15-17.5 | MT |
| POWER/LOGISTICS MODULE | 15-17.5 | MT |
| PRIMARY EVA NODE | 5- 7 | MT |
| AIR REVITALIZATION NODE | 5- 7 | MT |
| SANITATION/HYGIENE NODE | 5- 7 | MT |
| TRUSS SUPERSTRUCTURE | 6 | MT |
| LANDING SHOCKS/AIRBAGS | 4 | MT |
| SOLAR ARRAYS/COMM.ANT. | 3 | MT |
| LUNAR ROVER X 2 | 2 | MT |
| MISCELLANEOUS | 10 | MT |
| TOTAL - | 100 | MT |

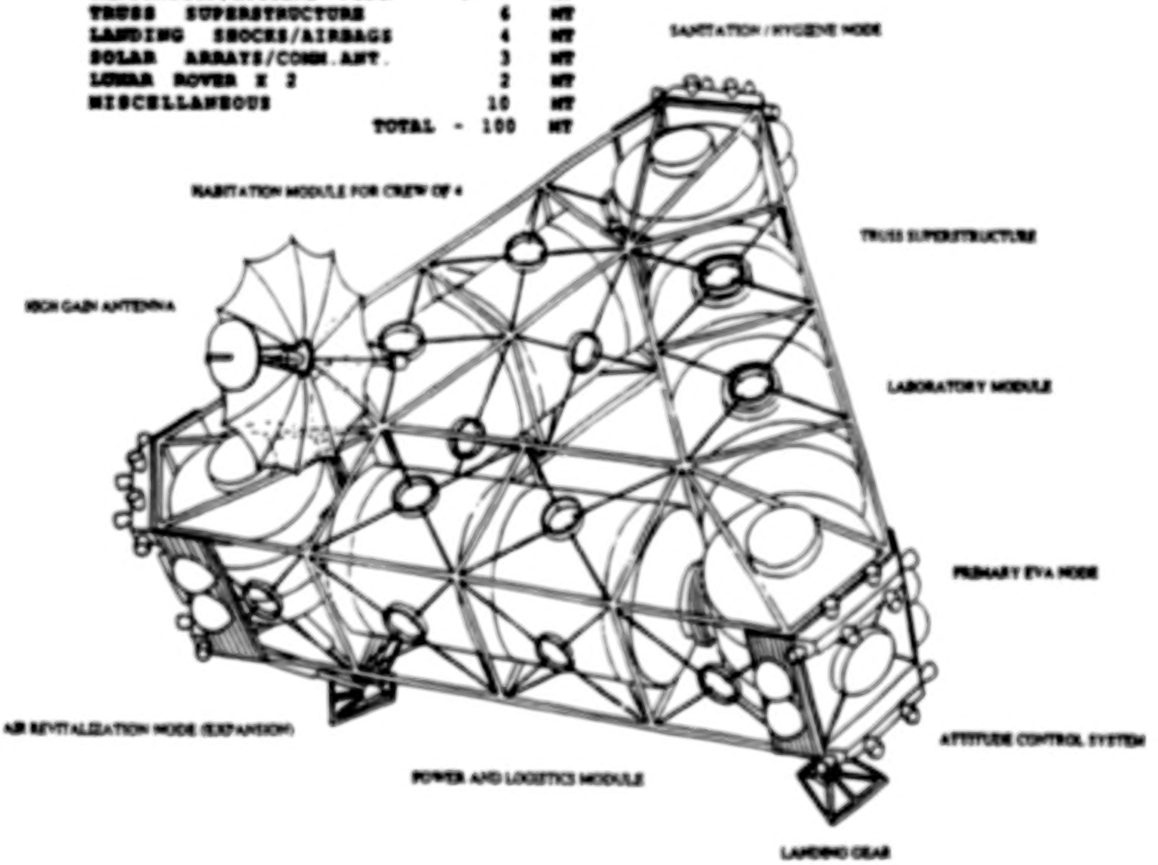
SANITATION / HYGIENE NODE
HABITATION MODULE FOR CREW OF 4
TRUSS SUPERSTRUCTURE
HIGH GAIN ANTENNA
LABORATORY MODULE
PRIMARY EVA NODE
AIR REVITALIZATION NODE (EXPANSION)
ATTITUDE CONTROL SYSTEM
POWER AND LOGISTICS MODULE
LANDING GEAR

Figure 4. Components of a 3 Module Lunar Habitation Base and LHB-1 Mass Summary

The components which integrate the modules together as a complete spacecraft are the following:

1. The truss superstructure has three functions. It is the structure employed to support the thrust structure of the modular orbital transfer vehicle(mOTV) and to distribute the forces transmitted from the mOTV uniformly through the entire LHB-1 during TLI, LOI and lunar landing. It also offers the primary attachment points for the attitude control system pallet(ACSP), the landing gear/airbag system and the storage pockets.
2. The landing gear/airbag deployment system is designed to absorb the shock on impact and may be conventional lunar excursion module type absorbers or controlled gas escape airbags or a hybrid system.
3. The three attitude control propulsion pallets are assembled and fuelled on Earth, brought up to LEO by the STS, attached to the three corners of the LHB-1 after the truss superstructure has been built around the modules, and they stabilize the attitude of the spacecraft during transit and landing operations.
4. The storage racks may be configured as required and are placed symmetrically about the MALEO truss superstructure in order to maintain the thrust structure symmetry and balance. They contain the rovers, the solar panels, and other EVA equipment, which are essential for the effective initial operational capability(IOC) of LHB-1.

## 5 The Modular Orbital Transfer Vehicle(mOTV)

The mOTV is used to transport the MALEO lunar base, complete and intact, to the prescribed lunar parking orbit. The mOTV provides the required delta V for TLI, LOI, and midcourse impulse and correction manuevers.Two current or near term options are available for the design of the mOTV propulsion system. They are Chemical Cryogenic Propulsion and Nuclear Electric Propulsion.

## 6 The Lunar Landing System(LLS)

The LLS is used to de-orbit and softland the MALEO LHB-1 on the lunar surface. Chemical cryogenic propulsion is favored for the descent and landing operation. Advanced RL-10 rocket engine technology would be applicable for the development of the LLS.

## 7 The MALEO Assembly and Deployment of LHB-1

At least four options exist for the MALEO Assembly of LHB-1.
They are:

1. Free Space Assembly using the STS as the work platform.
2. Free Space Assembly using the STS as the primary assembly platform with assistance from Space Station Freedom(SSF-1).
3. MALEO LHB-1 Assembly attached to Space Station Freedom(SSF-1).
4. MALEO Assembly connected to the manned core of SSF-1.

The fourth option utilizes the SSF-1 infrastructure most effectively by providing the assembly facilities and crew required for the MALEO assembly and operation of LHB-1.

Two options are possible for the deployment of MALEO LHB-1. They are :

1. Single phase direct lunar landing option
2. Two phase Lunar Orbit Rendezvous(LOR) option

In the simplest MALEO single phase direct lunar landing option, the Lunar Habitation Base(LHB-1) and the Lunar Landing System(LLS) are integrated in LEO and the entire assembly is directly delivered to the lunar surface.The operation is expected to take a few days from TLI to touchdown on the lunar surface. This option favours a fully cryogenic mOTV propulsion system.

In the two phase option, the LHB-1 is first transported to a lunar parking orbit(LPO) by the mOTV. In the second phase, the lunar landing system is transported by a similar mOTV to dock with the LHB-1 in LPO. After the LHB-1 and the LLS are securely interconnected in lunar orbit, the LHB-1+LLS descent and touchdown on the lunar surface as in the first option. The two phase LOR option effectively reduces the TLI mass of the MALEO operation in half and might be advantageous for cryogenic propellant management during the MALEO operation. This option is particularly suited for the slower electric mOTV orbital transfer, where cryogenic propellant boil-off might be a prime concern. Manned EVA is required in lunar orbit.

The following sequence of illustrations help to visualize the MALEO strategy 2 phase option for the deployment of the lunar habitation base.

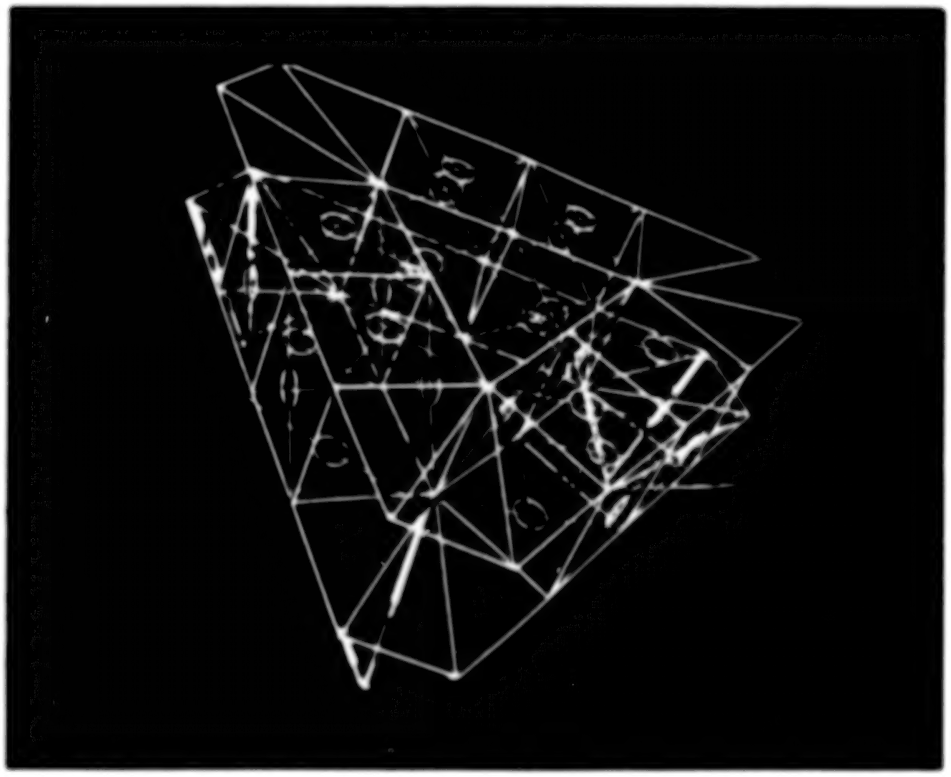

Figure 5. MALEO LHB-1 Truss Superstructure

Figure 6. MALEO LHB-1 Module Assembly

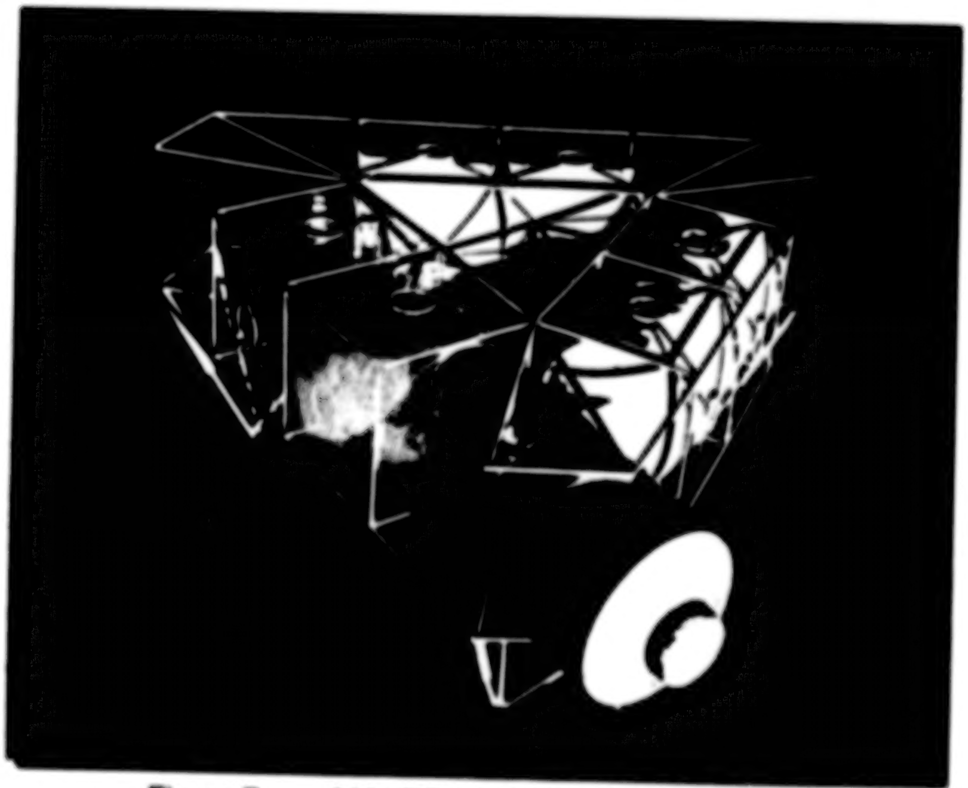

Figure 7. MALEO LHB-1 Module Insertion

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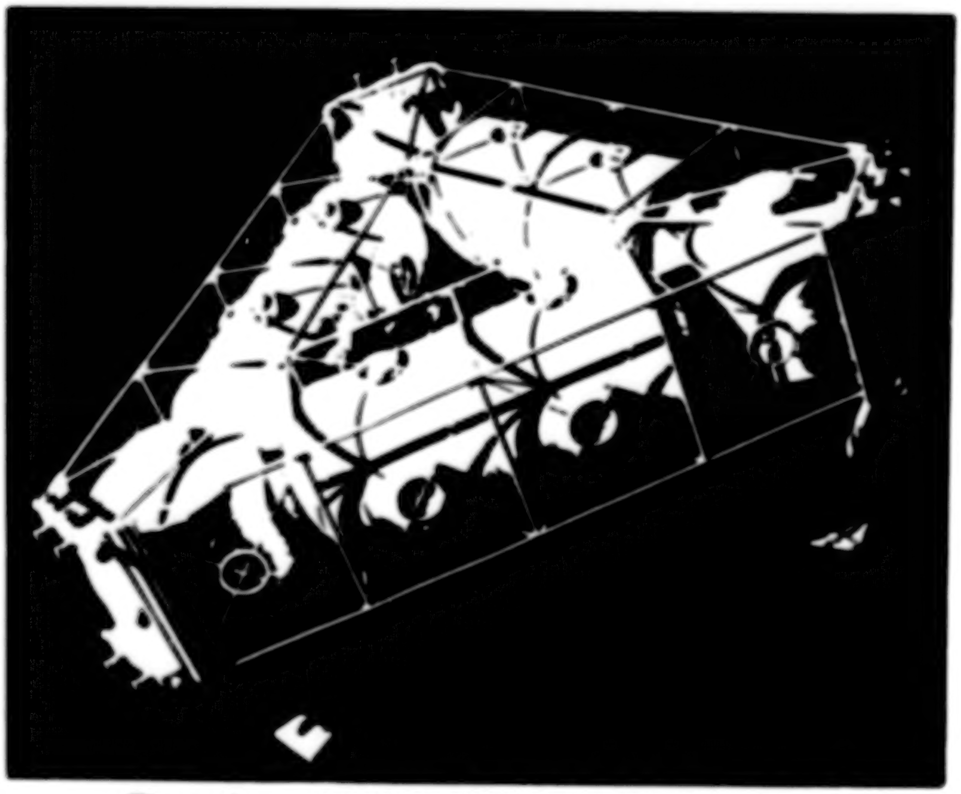

Figure 8. MALEO LHB-1 Assembly Complete

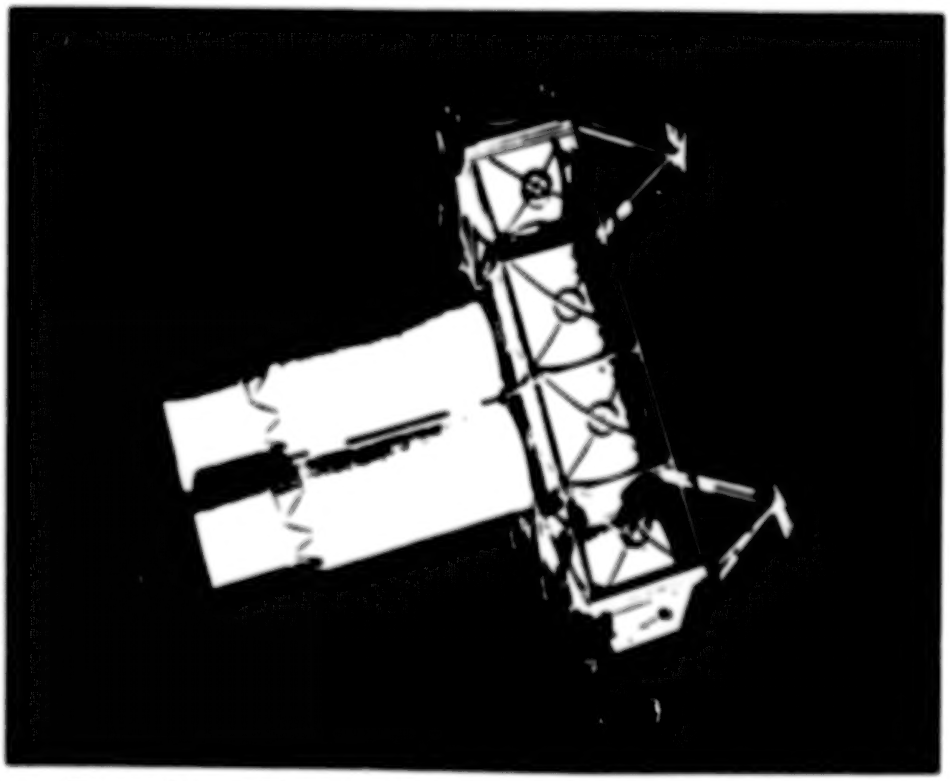

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Figure 9. Phase 1 LHB-1/mOTV Translunar Injection

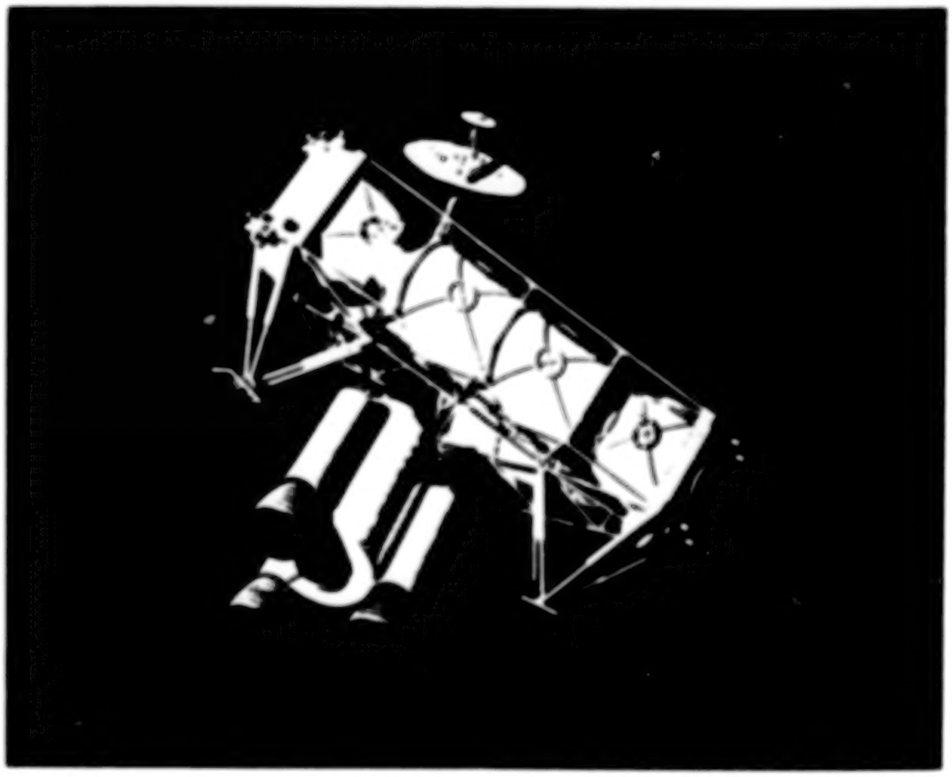

Figure 10. LLS/LHB-1 Lunar Orbit Rendezvous(LOR)

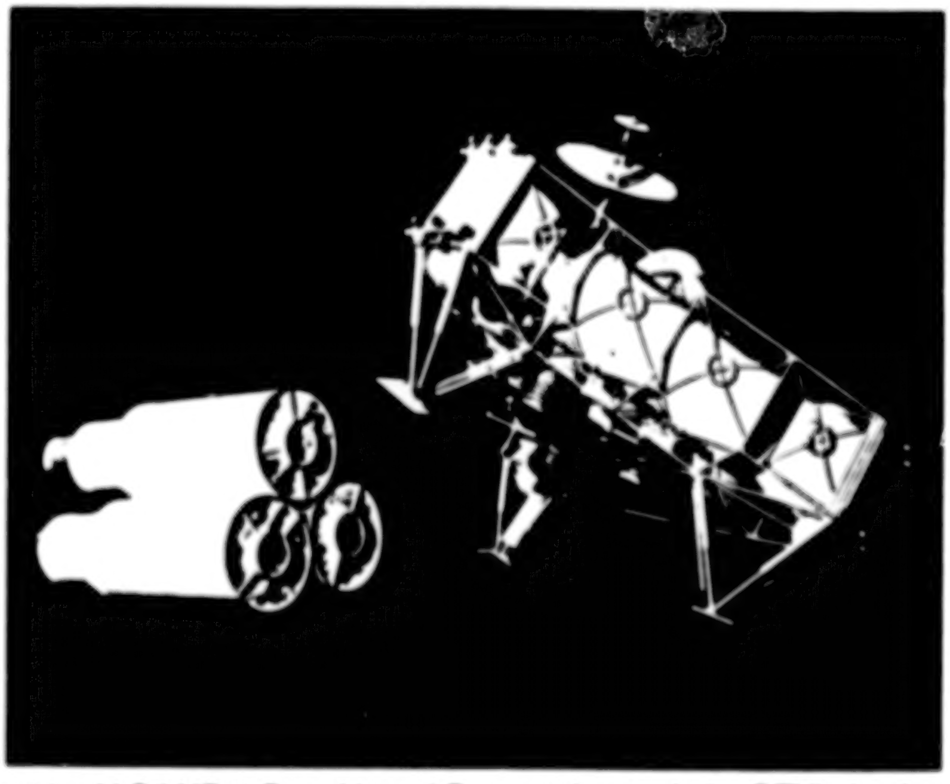

Figure 11. LLS/LHB-1 De-orbit and Descent (expended mOTVin background)

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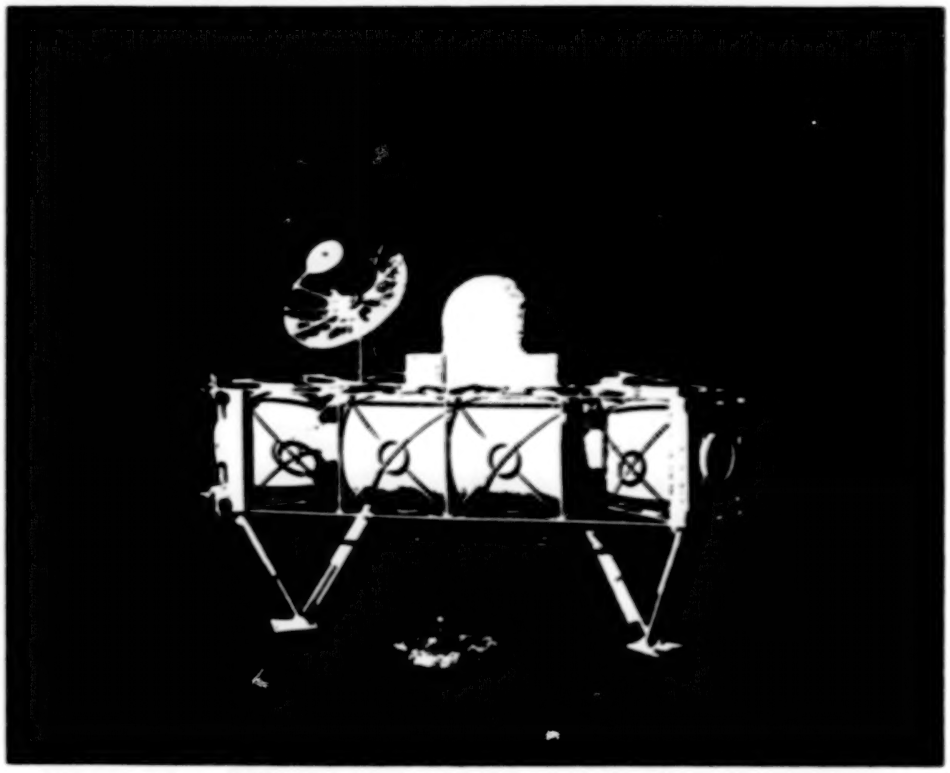

Figure 12. MALEO LHB-1/LLS Lunar Surface Touchdown

## 8 LHB-1 Structural System

From the operations listed above in sequence, it is evident that the truss superstructure of the LHB-1 and the complementing structures of the lunar landing system(LLS) and the modular orbital transfer vehicle need to be highly efficient, light weight and reliable. Historically, spacecraft have used few, if any, members in tension(Cohen,1987).

Inherently stiff tension members are suggested in the MALEO truss superstructure pretensioning system in order to conserve mass, minimize structural deflection, and to distribute the thrusting forces optimally among the truss members(Schierle,1990). The configuration is structurally strengthened by suspending the LHB-1 modules within a truss superstructure so that LHB-1 will be able to uniformly absorb the stresses induced on it during translunar injection(TLI), lunar orbit insertion(LOI), and lunar surface touchdown. The thrust structure, which includes the truss superstructure,the lunar landing system and modular orbital transfer vehicle interconnections, are selected so that the forces are applied symmetrically about the truss superstructure. The mOTV and the LLS share the same thrusting points on the truss superstructure. The truss superstructure is designed to facilitate quick and easy EVA assembly in LEO. An erectable/deployable philosophy is employed so that the structure is partially assembled even before launch to LEO. This philosophy is applied again upon lunar surface touchdown, after which much of the truss superstructure is disassembled and used for other building purposes on the lunar surface.

Though a stressed skin module configuration could be designed to take the stresses arising from TLI and LOI, which are estimated to be about 2g maximum, module safety and other factors dictated that a separate truss superstructure be employed for the thrusting load distribution.

## 9 Advantages of the MALEO Strategy

If space station derived modules are to be employed in the construction of a phase 1 lunar base, then the MALEO strategy offers the following advantages:

1. The safer LEO radiation environment for EVA, is also less risky and more economical than EVA on the lunar surface(Bufkin et al., 1988).
2. Spares and replacements are easily and more economically flown to LEO or borrowed from the LEO SSF-1 infrastructure.
3. The clean LEO assembly environment avoids dealing with the lunar soil which has undesirable abrasive and cohesive properties. Regolith has interfered with manned EVA systems in the past(Weaver,1988). During phase 1 build-up, paucity of man and equipment demands high reliability, and MALEO avoids the initial lunar surface assembly operations entirely.
4. In the event of an assembly accident, a crew rescue is more feasible from LEO using the assured crew return capability(ACRC) of SSF-1, than from the lunar surface.
7. The inheritability and commonality with Space Station Freedom (SSF-1) hardware enhances cost / unit economy as well as spares and replacement units for both the MALEO as well as the SSF-1 program.
6. LEO offers SSF-1 assisted assembly and the possibility of Earth-based real-time telerobotic assembly operations.
8. Avoids the risk of the repeated Earth to Orbit(ETO) launchings, TLIs, LOIs and lunar landings associated with the typical componentwise sequential build-up prescribed by the assembly on lunar surface strategy.
9. The LHB-1 is safely configured for habitation upon touchdown. Conventional strategies involving lunar surface assembly cannot offer a comparably safe environment during base build-up without additional investment.
10. In this primary extended duration mission, the MALEO LHB-1 offers an assured and substantial safe haven at IOC, which will help to diminish anxiety and enhance productivity among the astronaut crew.

## 10 The Disadvantages

1. The risk of losing the entire LHB-1 in the event of an accident during the transportation and landing demands high reliability.
2. Prefabricated MALEO base strategy deprives the assembly crew of the initial experience of learning to work on the lunar surface.
3. MALEO is a large manned spacecraft. All the MALEO systems, the LHB-1, the LLS, and the mOTV needs to be studied, their dynamic characteristics examined, and their limitiations confirmed.

## 11 Conclusion

The importance of highly reliable structural systems for the truss superstructure of the LHB-1, the lunar landing system, and the modular orbital transfer vehicle, are clearly evident for the success of such a mission. The vibration and resonance characteristics of such a large structure during lunar descent and a heavy impact or unsymmetric touchdown on the lunar surface, are being studied. Advances in structural materials and on-orbit assembly techniques acquired while assembling and operating SSF-1 will enhance this strategy. If a decision is made for the rapid deployment of a lunar base about the year 2000, the MALEO strategy holds promise and needs detailed engineering and systems analyses along with other strategies.

## 12 Acknowledgements

This paper is dedicated to my mother, Nanoo Saraswathy Thangavelu. I would like to thank P. Diamandis, T. Hawley, G.E. Dorrington, Dr.R.Shaefer and the whole crew at ISU for helping me to carry through the idea. I would like to thank Profs. R. Harris, G.G. Schierle, R.F. Brodsky, D. Vergun, E. Rechtin , R.K. Miller, E.P. Muntz, all of whom helped me to develop a framework in which to carry out a detailed study of the MALEO concept at the University of Southern California.

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N91-22171

# DESIGN STRATEGIES FOR THE INTERNATIONAL SPACE UNIVERSITY'S VARIABLE GRAVITY RESEARCH FACILITY

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## ABSTRACT

A variable gravity research facility named "Newton" was designed by fifty-eight students from thirteen countries at the International Space University's 1989 summer session at the Universite Louis Pasteur, Strasbourg, France. The project was comprehensive in scope including a political and legal foundation for international cooperation, development, and financing; technological, science, and engineering issues; architectural design; plausible schedules; operations, crew issues, and maintenance. Since exposure to long-duration zero gravity is known to be harmful to the human body, the main goal was to design a unique variable gravity research facility which would find a practical solution to this problem, permitting a manned mission to Mars. The facility would not duplicate other space-based facilities and provide the flexibility for examing a number of gravity levels including lunar and martian gravities. Major design alternatives included a truss versus tether based system which also involved the question of docking while spinning or despinning to dock. These design issues are described. The relative advantages and disadvantages are discussed including comments on the necessary research and technology development required for each.

## INTRODUCTION

The 1989 International Space University (ISU) convened July 1st in Strasbourg, France at the University of Louis Pasteur. One hundred twenty five students from twenty-five countries came to interact, study, and participate in a multinational, multidisciplinary educational experience in all aspects of space. An international faculty presented core lectures in eight space disciplines: Architecture, Business and Management, Engineering, Life Science, Policy and Law, Resources and Manufacturing, Satellite Applications, and Physical Science, providing a common base of knowledge for all the students. Advanced and plenary lectures from reknowned experts in each of the eight disciplines provided specialized study in each student's particular area of interest.

To promote interdisciplinary interaction and integration, two design projects were chosen whose goals were to utilize the talents and creativity of the students. Each project included mission objectives, design, organization, finance, implementation, and operation for peaceful international use. The selected design projects for 1989 ISU were a lunar polar orbiter and a variable gravity research facility. The names for these projects selected by their participants were Artemis and Newton, respectively. Faculty served as expert advisors. Department assistants who were 1988 ISU students provided additional support. The focus of this paper is to present the design alternatives for the variable gravity research facility, Newton, studied by the design team listed in Table 1. The required cooperation, collaboration, and understanding of the diverse student participants in research, analysis, decision-making, and complilation of the concluding design makes this project a remarkable achievement not only for its technical merit and feasibility but as a working example of outstanding international cooperation.

**ISU STUDENTS**

| | |
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Table 1. Names of all individuals and their countries of citizenship who worked on the Variable Gravity Research Facility project during the 1989 Summer Session of the International Space University.

# MISSION STRATEGY AND TIMELINE

Prolonged exposure to micro-gravity is known to be harmful to the human body. Some of the major problems are loss of heart and lung capacity, inability to stand upright, muscular atrophy, and loss of bone calcium. These debilitating effects of long duration-reduced gravity exposure must be minimized or counteracted for any long term human space travel such as the mission to Mars. Creation of artificial gravity in a rotating centrifuge or spacecraft is believed to be one possible way of combating these long term effects. The gravity level, spin rate, and duration that are compatible with human survival and efficient engineering design must be determined before a long duration mission to Mars can be undertaken. The variable gravity research facility, Newton, will find a practical solution, as soon as possible, to the problem of human adaptation to artificial gravity so that humans can go to Mars and return safely. It will provide the flexibility for examining a number of gravity levels including lunar and martian gravities and will not duplicate other space-based facilities.

The anticipated Mars mission development drove the end-point decision for Newton's operational lifetime. The scenario presented in the figure below seemed reasonable considering the extensive international development and cooperation required. The international organization consists of the U.S.A. and the U.S.S.R. as the primary partners with E.S.A., Canada, and Japan being secondary partners. The anticipated time-line of Newton is illustrated in Fig. 1 below.

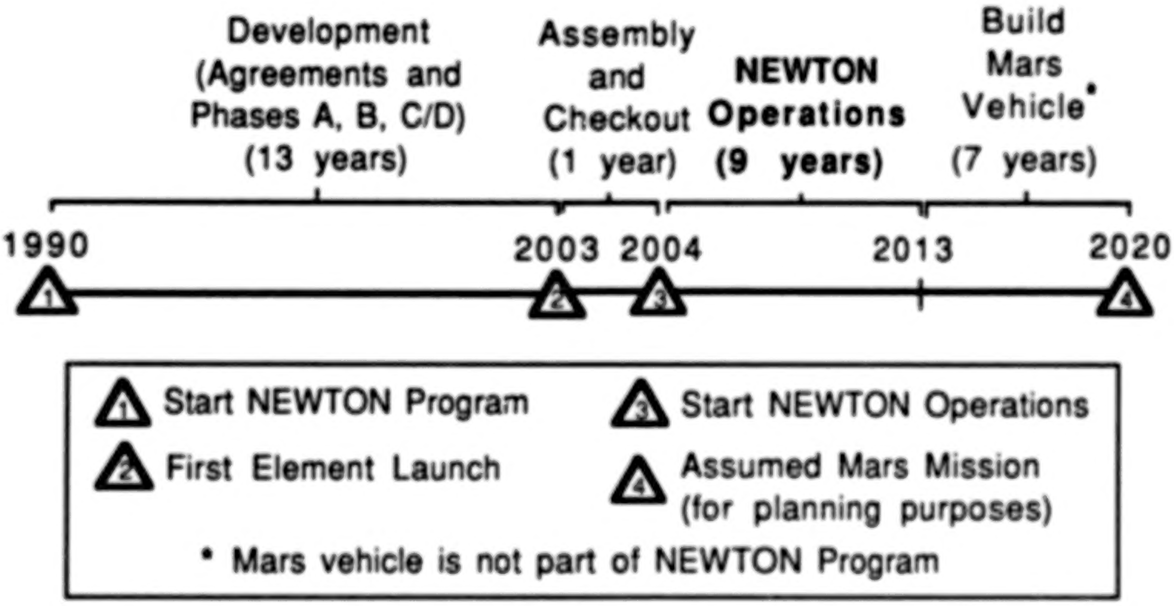
Development (Agreements and Phases A, B, C/D) (13 years)
Assembly and Checkout (1 year)
NEWTON Operations (9 years)
Build Mars Vehicle* (7 years)
1990
2003
2004
2013
2020
1
2
3
4
1 Start NEWTON Program
2 First Element Launch
3 Start NEWTON Operations
4 Assumed Mars Mission (for planning purposes)
* Mars vehicle is not part of NEWTON Program

Fig. 1. Newton Time-line

# FUNCTIONAL REQUIREMENTS DRIVING SYSTEM DESIGN

Certain assumptions were adopted to facilitate program design. They were as limited as possible in accordance with generally accepted projections for the timeframe listed above. The organizational structure assumes that the two major space-faring nations, the United States of America and the Union of Soviet Socialist Republics, will maintain the improving relations that have been demonstrated over the past few years. Furthermore, it is assumed that no major political problems will arise between or among the U.S.A., the U.S.S.R., or any of the other three partners. It is assumed that Newton will be constructed with technologies and launch vehicle capabilities that are currently in existence, which allows development costs and time requirements to be kept to a minimum. A notable exception to this is that Shuttle C, the future heavy-lift variant of the current U.S. Space Shuttle, is expected to be available when construction begins. Although international co-operative projects offer the benefit of shared costs, the price of Newton will be expensive for each of the partners. It is assumed that each partner has the necessary resources to build this facility, and the political motivation to do so. The U.S.S.R. and the U.S.A. will launch all required components and supplies for the Newton facility.

The functional requirements that were foremost in driving the design of Newton were as follows:

a) Newton must be capable of independently varying both spin rate and gravity level.
b) Newton must provide discrete gravity levels ranging between 0.1g and 1.0g including Martian and lunar gravities.
c) The maximum radius of rotation provides 1g of acceleration at 3rpm.
d) Newton will despin while docking.
e) Newton must accomodate a crew of six.
f) Newton design must permit phased development to allow replacement of modular lab racks and potential upgrade, such as replacing the counterbalance mass for laboratory/habitat modules. It will not be designed to permit additional mass at the end points.

# DESIGN CONFIGURATIONS

Two major design alternatives were considered: a truss-based configuration illustrated in Fig. 2 and a tether-based design shown in Fig. 3. Both systems permit implementation of the functional requirements of Newton. A brief description of each system follows and then a comparison of the advantages and disadvantages.

## Truss-based Configuration

The truss-based facility contains three major hardware sections: the module section, the despun section, and the counterweight section. The module section includes all of the pressurized habitation and laboratory modules, airlocks, escape vehicles, structural support hardware, and utility runs to provide continuous safe operation of the facility for up to six months. After six months the facility will be despun and resupplied through logistics modules. Long duration life support systems, thermal control systems, and meteoroid/space debris/radiation shielding are provided for the safe operation of Newton. The module section also includes a small reaction control system to provide control and maneuverability during zero gravity and artificial gravity conditions.

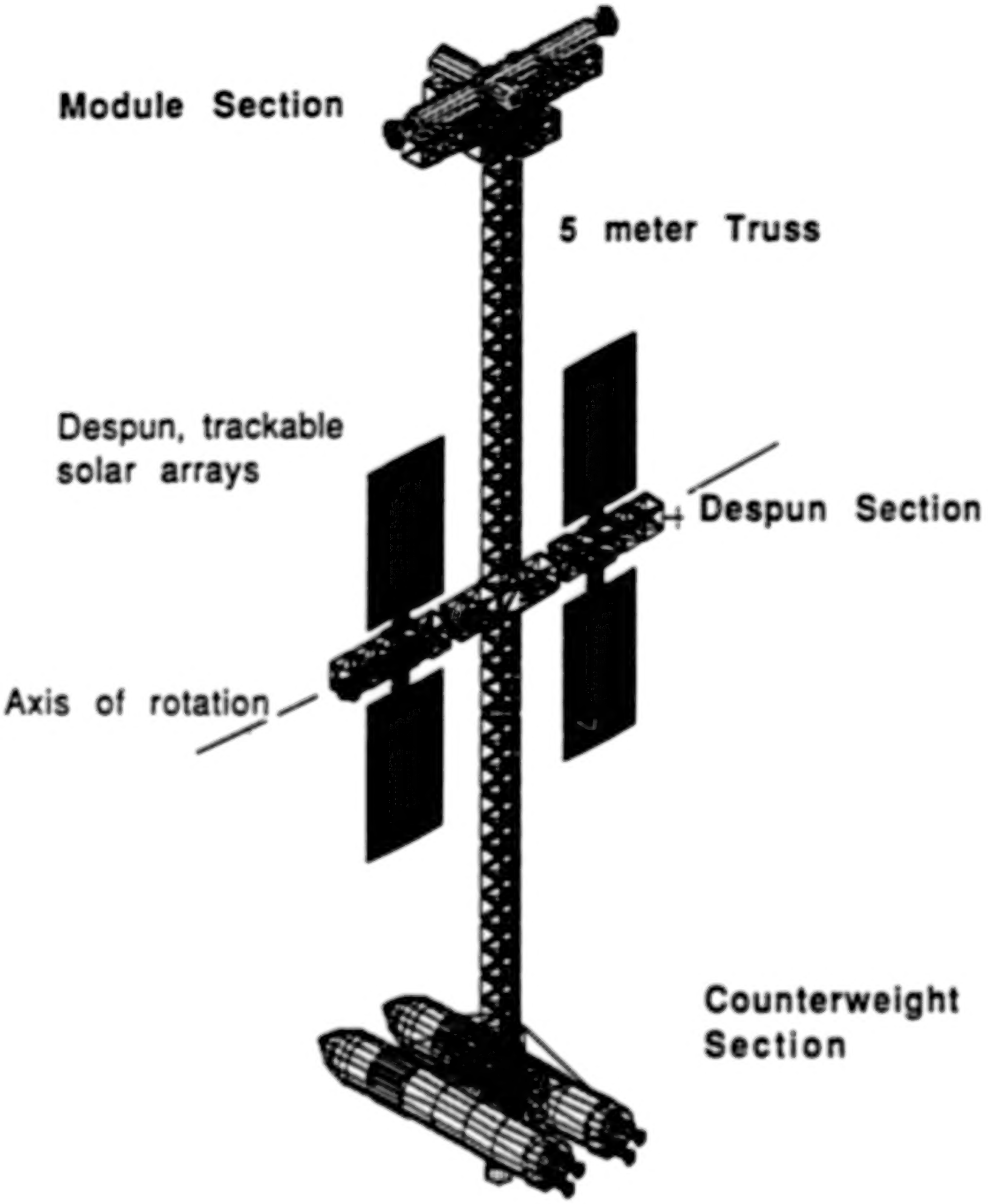
Module Section
5 meter Truss
Despun, trackable solar arrays
Despun Section
Axis of rotation
Counterweight Section

Fig. 2. Newton

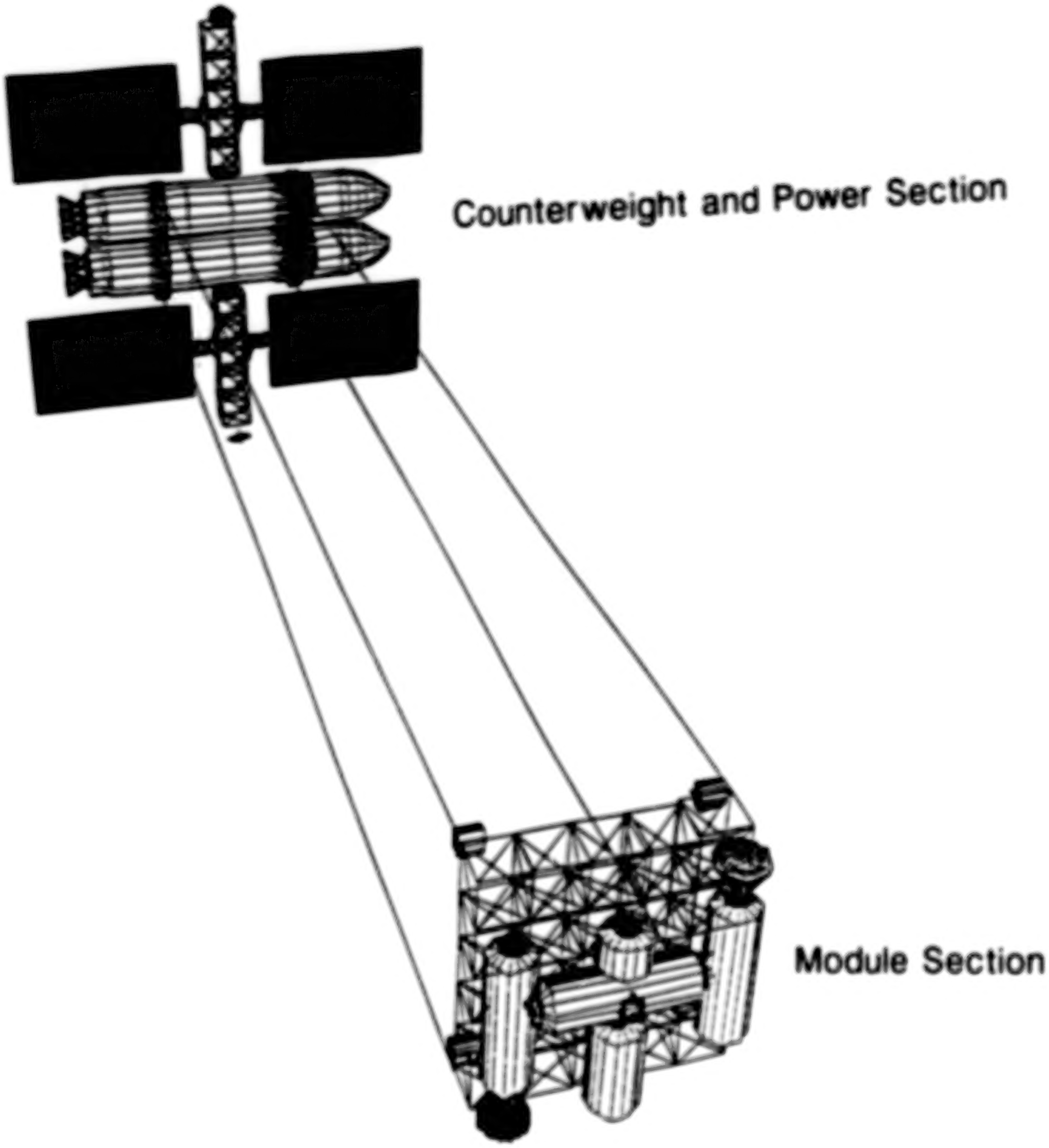
Counterweight and Power Section
Module Section

Fig. 3. Tether-based Newton

The despun section provides the power and external communication interfaces for Newton. Seventy-five kilowatts of continuous electrical power is provided through four solar array wings located on twin despun towers; each array is thus at zero gravity and capable of efficiently tracking the sun while the remainder of the facility is rotating. The twin communications antennae have the capablitiy to provide constant tracking of geostationary communication satellites. The despun section is mounted on a movable pallet to permit the section to remain at the center of gravity during radius or mass changes. A front view of Newton in Fig. 4 illustrates the despun section components.

The counterweight section balances the mass of the module section with a pair of spent Energiya core vehicles which have been orbitally outfitted with special mounting hardware to the truss assembly. The counterweight section is also capable of being moved to a new radial position by a mobile servicing unit. The variation in spin rate as a function of the movement of the counterweights is illustrated in Fig. 5. A larger reaction control system is located at the outer radius of the facility, and is capable of providing spinup and spindown thrust as well as boosting capability to higher orbits.

The truss assembly has been designed around a 5 m erectable bay, similar in size and composition to Space Station Freedom's truss structures. Fig. 6 illustrates the required 200 m truss structure necessary for Newton. Freedom is ~80 m in length not including the solar power modules. The struts used in Newton have a 3 cm radius and 2 mm wall thickness compared to the 2.54 cm radius and 1.83 mm wall thickness of Freedom's struts. The increased size of the struts is designed to account for material fatigue due to the rotation and hence induced structural tension.

### Tether-based Configuration

The tether-based facility contains two major hardware sections: the module section and the counterweight/power section, as illustrated in Fig. 3. During operation, the dominant load on the structural connection between the two ends of Newton is the tension load due to the centrifugal force. The load would be carried by a system of tethers or cables. Four tethers provide redundancy and torsional stability. The tethers can be reeled in and out from four pulley systems located at the habitat end. Rigid spacers would be placed between the tethers at regular intervals to minimize the free-floating length of a ruptured tether. Such a tether system would require location of the solar arrays on the core stages at the end of the facility. To enable the system to track the sun, the arrays have two degrees of freedom with alpha and beta joints as in the truss-based design. As the power system is located on a rotating end of the facility, the arrays are gravitationally loaded and need to be designed accordingly.

To facilitate control, it would be necessary to reel in the tethers prior to despinning. Alternatively, it is possible that docking could be accomplished without despinning by reeling out the tethers until the rotation rate becomes so slow that docking is possible directly at the module. This would require at least a kilometer of length and is only possible with a tether-based system. The decision to despin prior to docking was based primarily on safety considerations.

## TRUSS VERSUS TETHER: A COMPARISON

Four major design issues which must be discussed to assess the advantages and disadvantages of each system are: structural characteristics, assembly/deployment, operational use, and control.

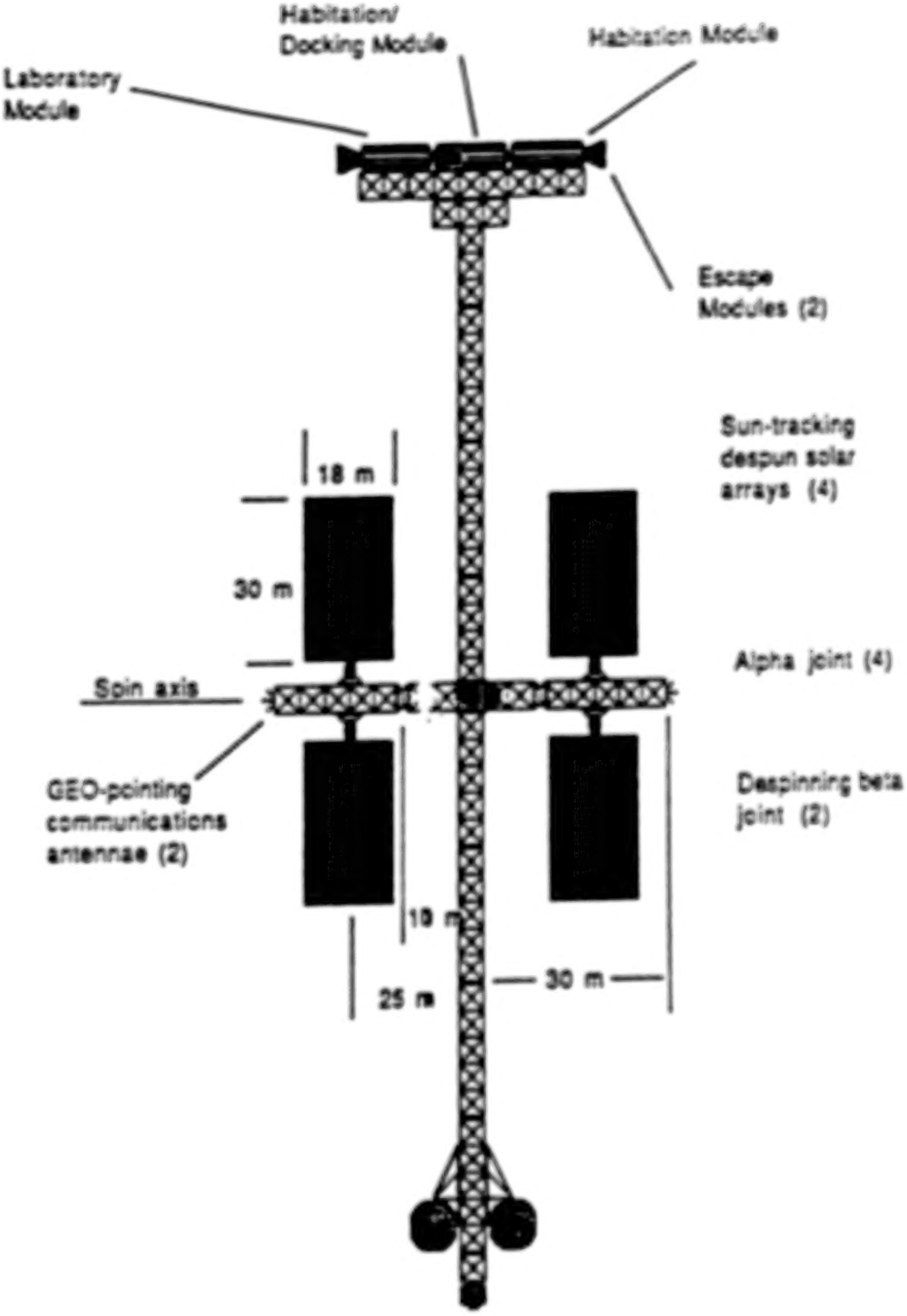
Habitation/
Docking Module
Habitation Module
Laboratory Module
Escape Modules (2)
Sun-tracking despun solar arrays (4)
18 m
30 m
Alpha joint (4)
Spin axis
GEO-pointing communications antennae (2)
Despinning beta joint (2)
10 m
25 m
30 m

Fig. 4. Front View of Newton

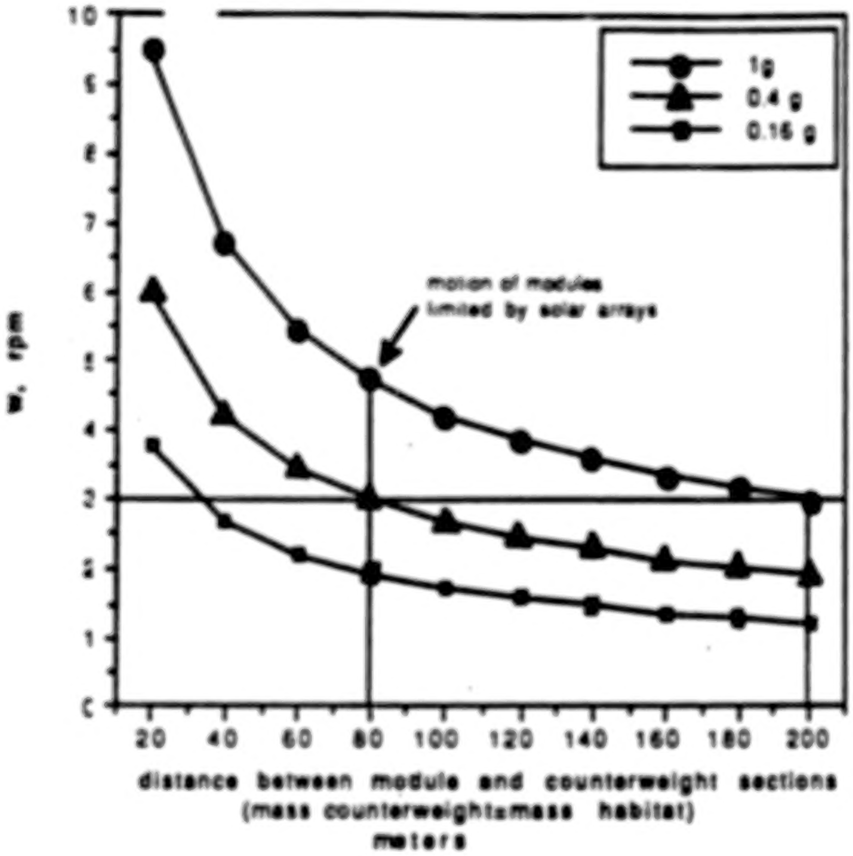

Fig. 5. Variation in spin rate for movement of the counterweights

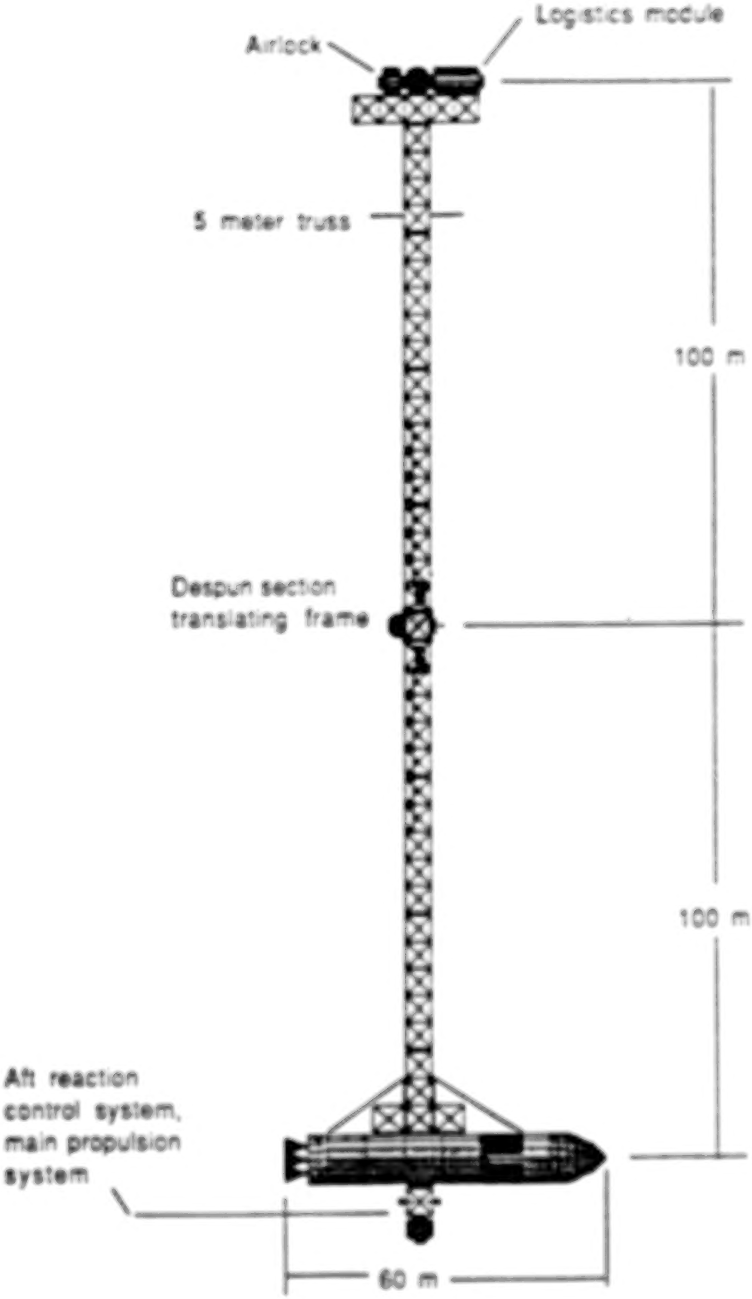
Airlock
Logistics module
5 meter truss
100 m
Despun section translating frame
100 m
Aft reaction control system, main propulsion system
60 m

Fig. 6. Side View of Newton

## CONCLUSION

A summary of some important considerations appear in Table 2. A truss structure clearly wins if more than two simultaneous gravity levels are required or for small radius structures where the added stiffness of a truss simplifies control. However for ease of gravity level variations permitting a variable radius rather than despinning, a tether system is preferable. Both systems permit phased development; however, a tether system excludes the possibility of expanding to simultaneous gravity levels. It is possible that a tether system would result in a total lower cost for a variable gravity research facility, however principally for safety reasons Newton was designed with a truss structure.

| TETHERS | TRUSS |
|---|---|
| | |
| Best in tension | Not optimal in tension |
| Higher strength to mass ratio (Kevlar) | Lower strength to mass ratio (Al/C composite) |
| Minimum volume to mass ratio (small rolled volume) | Higher volume to mass ratio |
| Continuous | Beams must be connected |
| Not rigid when not rotating | Always rigid |
| Easily deployable (quick) | Must be erected (time consuming) |
| Easy and quick length change by reeling | More complicated length change |
| Limited knowledge of deployment (Agena; Shuttle 1991) | Better knowledge of deployment (Freedom 1995) |
| No knowledge of dynamic behavior under rotation | No knowledge of dynamic behavior under rotation |

**Table 2. Tethers versus Truss: A Comparison**

## REFERENCES

1. "NEWTON: A Variable Gravity Research Facility, Volume I", The International Space University, 1989.

2. "NEWTON: A Variable Gravity Research Facility, Executive Summary", The International Space University, 1989.

## Structural Characteristics

The limiting factor on a truss's strength is the strength of the joints. An increase in the number of joints increases the potential failure points. A tether can be regarded as a homogenous, uniform tensile structure with a significant advantage in the strength to weight ratio under tension. Tethers do not have any static stiffness. Truss type structures have both static and dynamic stiffness. Some damping is thought to be initially present in both truss and tether systems. Active damping is greater in a truss type structure and requires active control. Damping considerations are particularly important during docking. In general it is easier to integrate utilities and power systems into a truss type structure. Power distribution requires significantly longer transmission lines in the tether system.

## Assembly/Deployment

Tethers are likely to require significantly less EVA than either deployable or erectable truss structures. However, the construction of Freedom will result in a gain of considerable experience in erectable truss assembly. A tether structure would be significantly simpler than a truss assembly to deploy. The tether also facilitates radius changes, whereas an erectable truss structure requires major operations to change the radius or system geometry. Tethers are most suitable when configured for two separate masses.

## Operational Use

Safety and reliability are of prime concern in considering a design choice. Whereas there is a lack of knowledge concerning the dynamic behavior under rotation of both a 200 m truss or tether, it is clear that tethers lack static stability. During assembly, they should be reeled together and then spun out. The major concern regarding tethers is maintaining control during spinup and spindown operation. If the facility were designed to continue spinning while docking tethers would be capable of being lengthened to permit a lower centripetal acceleration at the modules. This would permit docking directly with the module rather than to a central hub mechanism located at the center of gravity. Both structures are suitable for central docking, however eliminating the central hub docking facility and elevator transportation to the modules would increase the safety factor and reduce the complexity of the design. Tethers would have the further advantage of being able to absorb docking impacts by reeling out to absorb momentum and slowly retracting as oscillation subside. The truss system whether spinning or despun will use active control if there is any impact upon docking.

## Control

A finite element model of Newton's truss indicated that the fundamental frequency of the truss under worst case loading is approximately 0.8 Hz (this assumes a pinned constraint at each end of the facilitry). While tethers can approach this dynamic stiffness under worst case loading, their dynamic stiffness decreases with the square of the tensile loading, thus dramatically diminishing their controlability. A tether system would require a greater number of control systems than a truss structure. However it should be remembered that a long thin truss will also be very flexible. If the line of action of thrust through the end mass is not through the center of mass, there will be a resulting torque on the structure. The inherent stiffness in a truss structure will help to reduce the effect of this torque although a damping control system will be necessary. A more complex system will be required for a tether structure.

N91-22172

# A COMPARISON OF MICROWAVE VERSUS DIRECT SOLAR HEATING FOR LUNAR BRICK PRODUCTION

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## ABSTRACT

The Michigan Technological University Planetary Materials and Resource Utilization (PMRU) group has been examining the concept of fabricating bricks from lunar regolith. Such bricks are proposed for use in the construction of buildings that will provide protection from radiation and micrometeorite bombardment.

In this paper, two processing techniques considered suitable for producing dense bricks from lunar regolith are examined: direct solar heating and microwave heating. An analysis was performed to compare the two processes in terms of the amount of power and time required to fabricate bricks of various sizes. The regolith was considered to be a mare basalt of composition (in wt.%) 55% pyroxene, 20% plagioclase, 15% olivine, and 10% glass. Overall regolith density was taken to be 60% of the theoretical. Densification was assumed to take place by vitrification; several other mechanisms were considered but rejected since vitrification uses moderate amounts of energy and time while producing dense products. The average ambient temperature was assumed to be 50° C, while 1000° C was used as the temperature sufficient to achieve a viscous silica glass suitable for vitrification. Microwave heating was shown to be significantly faster compared to solar furnace heating for rapid production of realistic-size bricks. However, the relative simplicity of the solar collector(s) used for a solar furnace compared to the equipment necessary for microwave generation may present an economic trade-off. The relative costs and engineering complexity associated with the appropriate furnace design for these processes were not included in this analysis, although the final choice of a processing method will require such considerations.

## INTRODUCTION

There has been a renewed interest in space exploration, particularly the establishment of manned lunar/Martian bases, as a result of the proposals outlined by President Bush in his speech on July 20, 1989, commemorating the 20th anniversary of the Apollo 11 mission. The design of such bases can take two general paths. The first would involve bringing up all construction materials or importing pre-fabricated modules. However, the economics of material transportation from earth dictate that a second path be explored, in which local resources such as lunar regolith be used whenever possible, *e.g.*, as construction materials. Toward this end, the PMRU group at Michigan Tech has utilized a multi-disciplinary approach to study the design and fabrication of construction "bricks" made of lunar regolith.

The mechanical behavior of a shelter constructed from "silo stave" bricks with tongue-and-groove joints was examined in a previous study (ref. 1). The technique chosen to fabricate this (or any) brick design will play an important role in lunar/Martian base design and construction in terms of brick production rate, ease of automation, brick production cost, etc. Although a variety of fabrication techniques are possible, the most promising methods to densify regolith into bricks are microwave heating and direct (focused) solar heating. The former is generally considered to be very economical in terms of the energy expended during sintering while the latter does not require a special power generation source.

Microwave radiation for lunar brick production would be generated via an electrical source such as solar cells or a nuclear power plant generator. This radiation must be "tuned" to a desired frequency, which would correspond to a strong absorption peak of the phase to be heated. If that phase is reasonably dispersed throughout an agglomerate, heating takes place uniformly within the green body. A relatively high heating rate and heating efficiency can be achieved since thermal conduction is required only over short distances. In contrast, a solar collector directs a wide range of frequencies (determined by the collector's reflectivity) onto a target. A large portion of the wavelengths fail to achieve maximum energy transfer to any particular phase; thus thermal conduction over larger distances is required. The resulting heating efficiency and rate are expected to be low since the thermal conductivity of the green body is quite low, due to the vacuum of space.

The purpose of the current study was to make a preliminary analysis of the power and time requirements necessary to form lunar bricks using either microwave or direct solar heating. Although there are a number of potential factors that have to be considered, we will limit our consideration to the power and time characteristics of each process in order to determine which technique represents the better choice for a given brick size.

## PROBLEM BOUNDARY CONDITIONS

Several assumptions are necessary in order to proceed with the calculations. Table 1 describes the average regolith composition used in this study (ref. 2). This raw material was assumed to have an apparent density of 60% of theoretical (ref. 3). The average ambient temperature at the lunar surface during "daylight" was taken as 50° C. The brick morphology was simplified to that of a parallelepiped (see Figure 1). All calculations were made assuming 100% efficiency, *i.e.*, there was no heat loss to mold walls, all incident energy was absorbed by the regolith, and no power losses occurred in the solar collector or microwave generating circuit. These assumptions will be discussed later in more detail.

Table 1. Assumed Regolith Composition

| Mineral Class | Mineral Name | Composition | Fraction (wt.%) |
|---|---|---|---|
| Pyroxene | Diopside | $CaMgSi_2O_6$ | 55 |
| Plagioclase | Anorthite | $CaAl_2Si_2O_8$ | 20 |
| Olivine | Forsterite | $MgSiO_4$ | 15 |
| Glass | Silica | $SiO_2$ | 10 |

A variety of potential densification mechanisms exist, including solid-state sintering, liquid-phase sintering, and complete melting and solidification. A variation of liquid phase sintering known as vitrification was chosen for analysis due to the relatively low temperatures and times required for densification. Vitrification relies on the softening and viscous flow of a glassy phase (*e.g.*, $SiO_2$) to densify the regolith mass. Therefore, a microwave frequency (43 GHz) was chosen to selectively excite the Si-O bond and a processing temperature of 1000° C was chosen in order to soften the glassy $SiO_2$ phase (ref. 4).

Figure 1. Schematic of the brick morphology.

## CALCULATIONS

The two processing methods were compared on the basis of how much power was available to heat regolith bricks of varying size and the time required to attain the appropriate vitrification temperature.

The power available for direct solar heating is a function of the sunlight incident on the lunar surface and the surface area of the collector. The former has a value of 1400 W/m$^2$ , and a representative collector size of 10 m$^2$ was selected. This results in an available power of 14 kilowatts which is independent of the mass of the regolith brick (see Figure 2).

The power per unit volume [W/m$^3$] deposited in a dielectric by an electromagnetic field is given by equation (1),

$$P = 5.56 \times 10^{-11}\, k'\, \tan(\delta)\, f\, E^2 \tag{1}$$

where k' is the relative dielectric constant, tan(δ) is the loss tangent, f is the frequency [Hz], and E [volts/m] is the magnitude of the internal field. Work on a large number of lunar soil samples (ref. 5) has led to empirical relations to describe the relative dielectric constant and loss tangent of these materials as a function of sample density and are shown in expressions (2) and (3). The rule of mixtures, along with the data in Table 1, was used to calculate the theoretical regolith density.

$$k' = 1.919^{\rho} \tag{2}$$

$$\tan(\delta) = 10^{(0.44\rho - 2.943)} \tag{3}$$

The theoretical density was used to calculate k' and tan(δ) rather than integrating power over the range of densities (60-100%) that would result during the sintering of an actual regolith brick; this procedure was used to simplify the calculations, and the results represent an upper bound value for power. The frequency was chosen as 43 GHz since this value corresponds to a characteristic rotational transition in Si-O bonds (ref. 6). Previous work on microwave heating of ceramics has utilized applied voltages of 300-400V; a value of 400V was chosen for this analysis since dielectric breakdown will occur at higher voltages under hard vacuum conditions. Using these values, the microwave power available to heat the bricks as a function of brick volume was calculated and plotted in Figure 2.

A calculation of the time required to raise the regolith mass from a temperature of 50° C to 1000° C is possible using the definition of power as the time rate at which work is done, equation (4). E [J] is the heat required to raise

$$P = \frac{E}{t} \tag{4}$$

the regolith mass from 50° C to 1000° C in time t. The heat is calculated using equation (5) where m is the mass [g] of the regolith brick and $C_p$ is the heat capacity [J/mole K]. $C_p$ values were obtained utilizing known expressions for the heat capacity of the component minerals. The brick mass was calculated for different brick volumes using an apparent regolith density of 60% of theoretical. This value represents a lower bound since some compaction of the regolith may

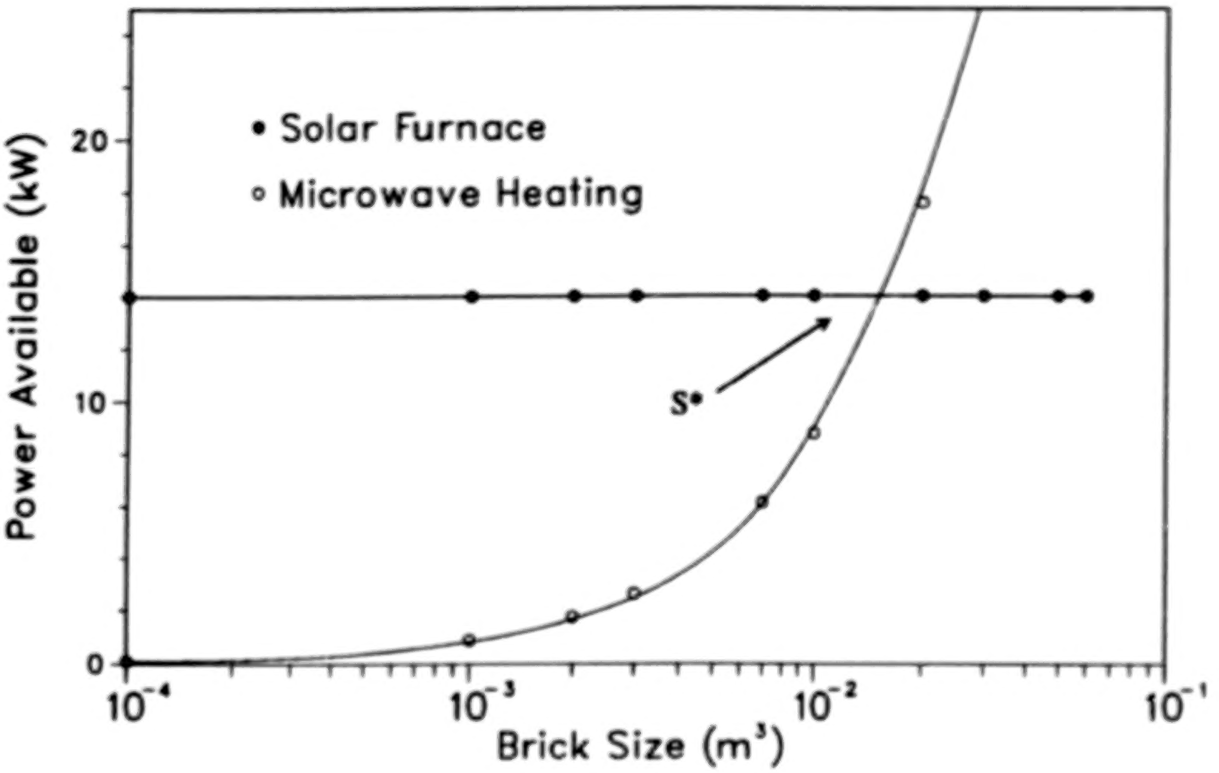


Figure 2. Power available for heating the regolith as a function of brick size. The solar power curve is based on a 10 $m^2$ collector.

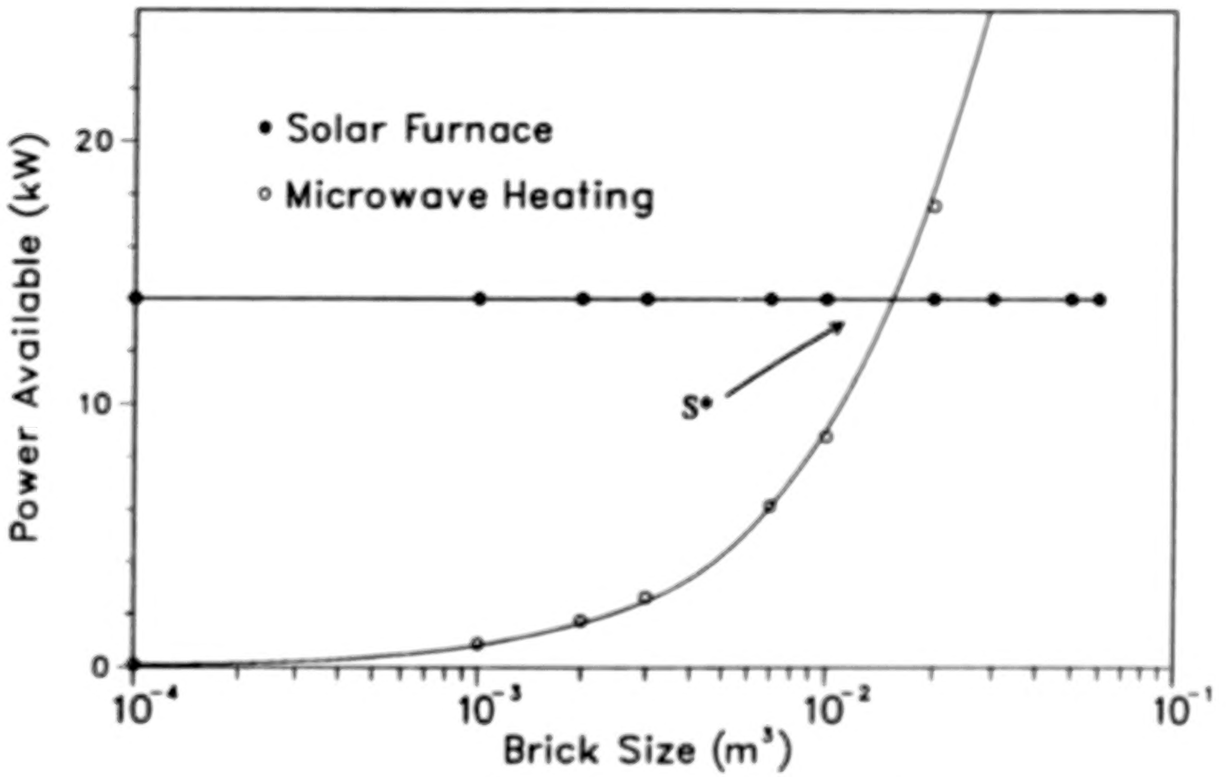
be necessary to form the brick shape. It was assumed that no change took place in the density of the green regolith brick as the temperature was increased to 1000° C. With this information, the heat required to raise the regolith temperature by 950° C was determined from Equation (5) as a function of brick size; the assumption was also made that no phase changes took place in that temperature range. The time required to attain a temperature of 1000° C could then be calculated for either direct solar heating or microwave heating as a function of the brick mass using Equation 4 (see Figure 3). The actual time required for vitrification to take place at 1000° C was assumed to be equal for both processes.

$$E = m \int C_p \, dt \qquad (5)$$

## DISCUSSION

An intersection (S*) occurs in the two curves plotted on Figure 2 since the power available from direct solar heating is a constant while the microwave power increases with increasing brick volume. The plot indicates that more power is available from direct solar heating than microwave heating at smaller brick volumes while the reverse is true at larger brick volumes. The intersection, S*, occurs at a brick volume of 0.016 $m^3$. If a brick thickness of 0.1 m is assumed, typical dimensions of the brick at S* would be 0.1 m x 0.4 m x 0.4 m. This yields a brick size comparable to those found in traditional terrestrial buildings. Thus, even though direct solar heating appears to be more efficient at smaller brick volumes, more power is available from microwave heating for realistic brick sizes.

It should be pointed out that the intersection point will vary directly with the size of the solar collector(s) used. That is, as the collector area increases, direct solar heating becomes more competitive with microwave heating. However, solar heating of the regolith relies on transfer of heat from the outside of the brick toward the interior. As brick



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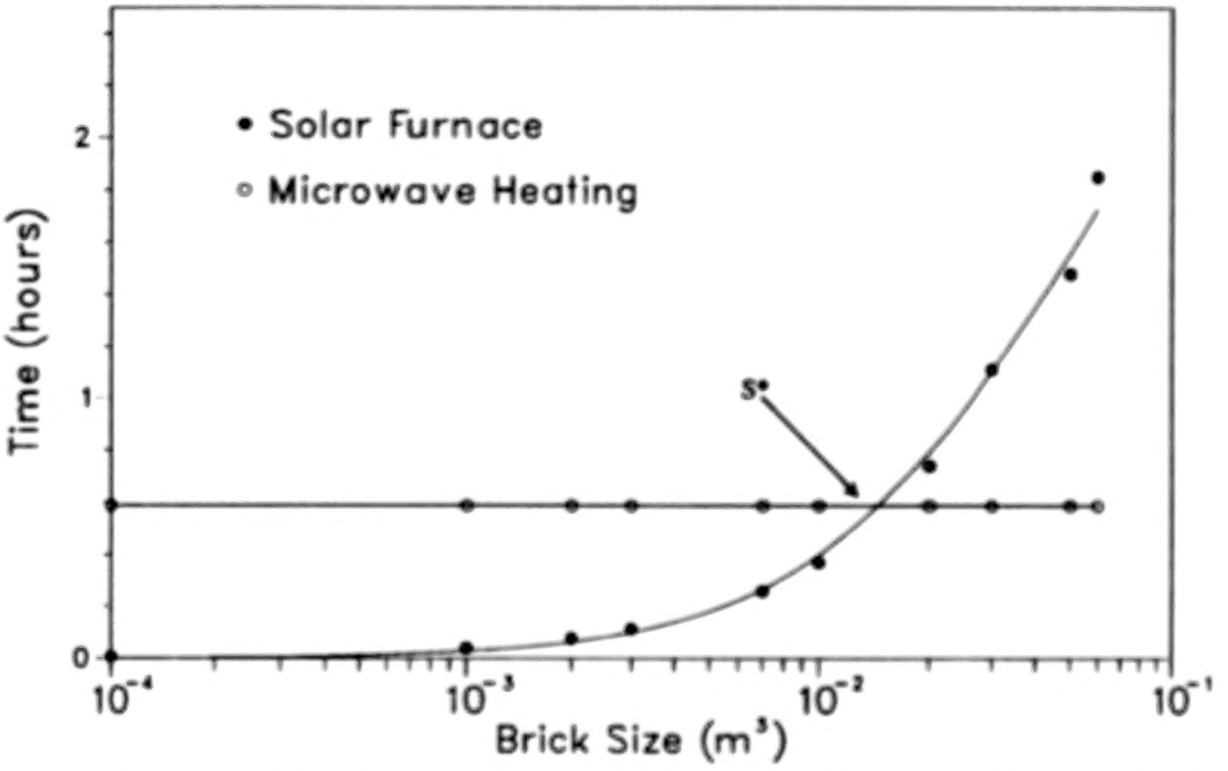

Figure 3. Time required to supply the regolith charge with the calculated amount of energy as a function of brick size. The solar furnace curve is based on a $10 m^2$ collector.

size increases, the very poor thermal conductivity of the regolith (~60 µW/cm K) (ref. 7) may result in very high surface temperatures with little interior heating. This temperature gradient could result in differential shrinkage of the brick, producing internal stresses that could degrade the brick properties. The low thermal conductivity would also increase the processing time for vitrification to take place.

We have also neglected the radiative heat loss that will occur at the brick surface in our calculations for direct solar heating. The heat loss will be proportional to $T^4$, and as a result, additional energy will be required to compensate for this energy loss. The problem in trying to make the calculation is that emissivity data are required for the minerals and temperature range of interest. Thus, the calculations shown in Figure 2 represent an upper bound case. This would again support the selection of microwave heating although innovative design and engineering of a solar furnace will play an important role in determining the heating characteristics and degree of heat loss associated with brick production.

In contrast to direct solar heating, the temperature gradient in microwave heating is reversed. That is, heating takes place from the interior of a body outwards, which minimizes heat loss to the surroundings. Larger bricks can be more easily fabricated with microwave heating since heating can take place uniformly if the strongly absorbing phase is distributed evenly within the green body, and since thermal conduction is required over much shorter distances. It has also been shown that as densification proceeds, heating via microwave coupling tends to concentrate at pores and other defects, leading to faster and more complete densification (refs. 7 and 8).

The plot of the time required to supply the regolith charge with the energy required to raise its temperature to 1000° C (Figure 3) shows that microwave heating requires less time at the brick volumes that are realistic (*i.e.*, greater than 0.016 $m^3$) and the times themselves are practical for a continuous production process. It should also be noted that the processing time for microwave heating will be considerably shorter than that calculated strictly with the use of equations 1-3. The tuned frequency of 43 GHz was chosen to heat primarily the glassy phase in order to achieve densification via vitrification. However, the other regolith components are also silicates and contain Si-O bonds. As a result, very uniform heating and shorter sintering times should be achieved.

As described earlier, solar heating relies on heat transfer from the brick surface to the interior. As the brick size increases, the surface area to volume ratio decreases, thus leading to sintering times which increase rapidly as brick size increases (see Figure 3). In contrast, microwave heating relies on the excitation of specific bond types which are assumed to be distributed uniformly throughout the regolith. Since the number of these bonds scales directly with the amount of regolith present, the time required to reach a given temperature is independent of brick size. These differences in the heating mechanism of each process lead to the form of the curves shown in Figure 3. It should be noted, however, that the solar furnace curve is a significant underestimate of the time necessary to raise the temperature of the entire brick to 1000° C. The curve in Figure 3 was calculated with the assumption that the entire brick instantaneously attains thermal equilibrium as its temperature is raised to 1000°C. To obtain more realistic results, a one dimensional heat flow model incorporating the thermal conductivity of the regolith was employed. The model (ref.9) assumes conductive heat flow from two faces of a flat plate of thickness z with large or infinite surface dimensions (see figure 1) and non-steady-state conditions. An analytical solution of the one dimensional heat flow equation was obtained using this model. The time required for the center line of the brick to reach 1000°C, as a function of brick thickness, could then be calculated using appropriate values for the parameters (e.g., thermal conductivity = 60 μW/cm K). For comparison purposes, a large constant surface area of one square meter was assumed to insure the assumption of one dimensional heat flow. These calculations, which incorporate a temperature gradient, are plotted in Figure 4 and compared with the solar data from Figure 3, which assumes instantaneous thermal equilibrium. It is apparent that accounting for the regolith thermal conductivity increases the processing time by several orders of magnitude for realistic brick thicknesses. This is primarily due to the nearly perfect vacuum in the void space between regolith particles, which acts as insulation and prevents heat convection between particles. The shaded region of Figure 4 represents a probable processing time "window" for the fabrication of lunar regolith bricks using direct solar heating.

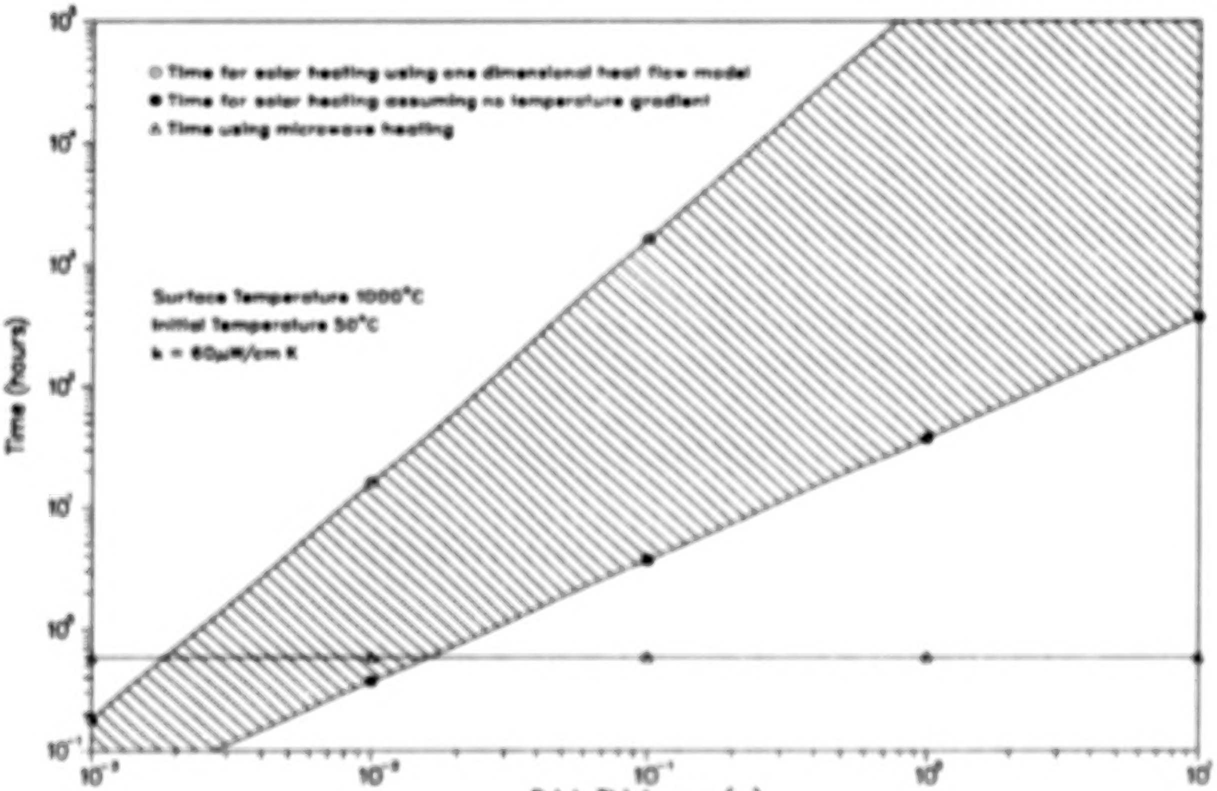

Figure 4. Time required to supply the regolith charge with the calculated amount of energy as a function of brick thickness. The upper most curve takes into account the thermal conductivity of the regolith and the resulting temperature gradients.

This preliminary analysis has ignored economic issues which will have an important impact on the selection process. For example, although solar furnace power increases as collector area increases, the size, mass, and transport of very large collectors must be considered. However, the relative simplicity of a solar collector and furnace is attractive compared to a microwave generation/transmission unit. A solar furnace has the disadvantage of only functioning during the lunar day while a microwave facility would be capable of continuous operation. Electricity for microwave generation would need to come from solar cells or a nuclear generator, although solar cells have been developed recently to operate at 20% efficiency (ref. 10). The conversion of electricity to microwaves is approximately 50% efficient at present, while solar collectors are capable of focusing 90% of incident sunlight. These efficiencies must then be contrasted with the actual sintering efficiency of the two processes. It is apparent that the final choice of technique will have to be based on a combination of technical and economic concerns, as well as environmental factors.

## CONCLUSIONS

Microwave and direct solar heating processes have been examined as methods of producing bricks from lunar regolith for potential construction applications. Microwave sintering offers a number of advantages in terms of the power available for densification and the short processing times required for realistic brick volumes based on the assumed regolith composition and selected values for the processing parameters. However, a number of other factors, primarily the economics of power generation, have to be taken into account before an optimal process is developed.

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N91-22173

# A HYDROPONIC DESIGN FOR MICROGRAVITY AND GRAVITY INSTALLATIONS

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A hydroponic system is presented that is designed for use in microgravity or gravity environments. The system utilizes a sponge-like growing medium installed in tubular modules. The modules contain the plant roots and manage the flow of nutrient solution. The physical design and materials considerations are discussed as are modifications of the basic design for use in microgravity or gravity environments. The major external environmental requirements are also presented.

## INTRODUCTION

This paper presents a design for a hydroponic system capable of supporting plant growth from seed through maturity. This system can operate in either microgravity or lunar/planetary base environments with only minor modifications required to transition between these environments.

The hydroponic system provides a dark, moist, nutrient-rich, well-aerated environment that is suited for the growth of plant roots. A supporting medium holds the seeds and growing plants which have access to a lighted external environment. In addition, containment is provided to keep solution flows under control in a microgravity environment.

## PHYSICAL DESIGN

The basic hydroponic design consists of tubular structures of sufficient size to accommodate the types of plants being grown. Each tubular structure is a cylindrical module with an open slit extending for the majority of the length of the cylinder. This open slit accommodates a sandwich of a foam-like, spongy substance that holds seeds or growing plants. The slit is oriented towards the lighting system.

The sandwich consists of two strips of spongy material that are inserted into the open slit to fill the entire length of the slit in the cylinder. Crop seeds or plants are placed between the two layers. The portion of the strips exposed to the light is coated with an impermeable, opaque coating.

The tubular structures are installed in arrays that make optimum use of the lighting system and the available pressurized volume. Each tube is connected to a nutrient solution feed line and a nutrient solution removal line. Both the feed and removal lines have valves and bypass lines so that individual modules can be taken out of service without disrupting the operation of the entire growing system.

## NUTRIENT FLOWS

In the microgravity configuration, the nutrient solution feed system is a flattened pressurized feed line inserted near the surface of each strip of sponge. (Fig. 1) A pressurized flow of aerated nutrient solution is introduced into the sponge strips. The flow is directed away from the exterior towards the "bottom" of the cylinder. This flow is maintained and reinforced by a longitudinal air flow established within the cylinder and by the removal of excess and spent nutrient solution by a suction system. (Fig. 2) In a microgravity environment this pressurized flow of nutrient solution may aid in orienting root growth towards the interior of the cylinder. What roots do grow into the exterior will likely be air-pruned into directionality.

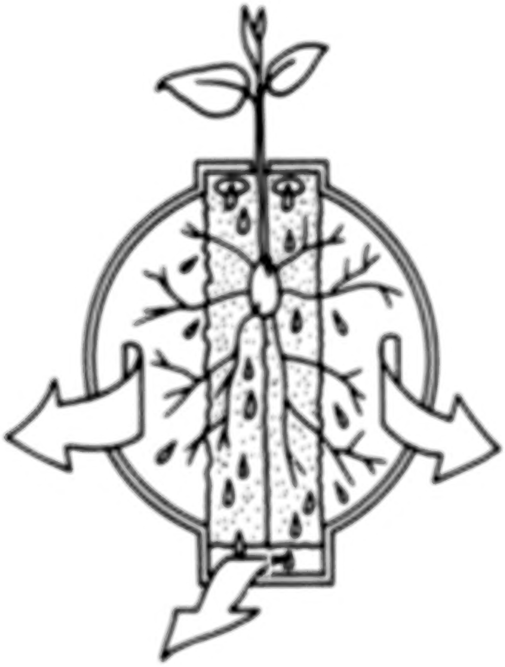

Figure 1. Microgravity system configuration.

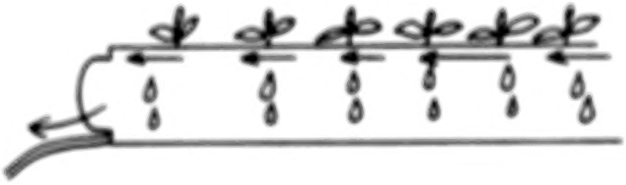

Figure 2. Nutrient flow in microgravity.

In the gravity environment, the nutrient solution is introduced into the cylinder in an ebb and flow schedule. The cylinder is allowed to fill with nutrient solution and then the nutrient solution is drained away. (Fig. 3). This ebb and

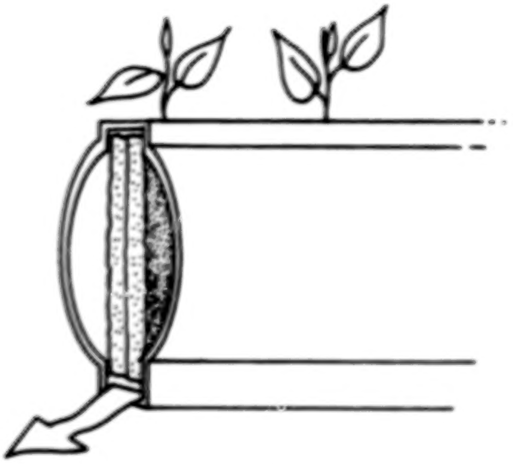

Figure 3. Nutrient flow in gravity.

flow cycling ensures that the roots are well aerated in addition to being supplied with nutrient solution. The ebb and flow cycles are more frequent during periods when the plants are lighted and transpiration is at a peak than during the dark periods.

## NUTRIENT MANAGEMENT AND CONTROLS

The nutrient solutions used in this system must contain all macro nutrients and micronutrients required for plant germination, growth, and maturation. The solutions will need to be formulated for individual crop types to ensure optimum growth rates and yields.

Hydroponic systems tend to be unforgiving of failures in nutrient flows. This system partially mitigates this problem with the limited nutrient solution holding capacity of the sponge. Redundant reservoirs of pre-mixed nutrient solution will be required to replace the existing nutrient solution system reservoir should it become contaminated or lost.

Periodic sterilization of the circulating nutrient solution is recommended to control pathogens. In-line filters will limit the movement of plant materials and some types of pathogens through the system. Plant toxins, pH, and soluble solids must be monitored on a continuous basis.

The circulating nutrient solution must be refreshed and periodically replaced in entirety. Provision should be made for the reconstituting of the spent nutrient solution and for the recycling of organic wastes. These major requirements are not easily achieved in a hydroponic system.

## EXTERNAL ENVIRONMENT

The environment outside of the cylinder should replicate many of the features of the earth's environment. The plants should be fully and evenly lit with light provided in a regulated day/night cycle. Broad spectrum lighting in the 400 to 800 nanometer range is required with infrared emissions kept to a minimum. The light intensity should be 200 to 400 watts per square meter for optimum plant response. This lighting system will generate waste heat that must be removed from the agricultural area.

Temperature in the agricultural areas should be maintained in the range of 15°C to 32°C. A standard nitrogen, oxygen, and carbon dioxide atmosphere is required. A pressure of 800 mb or above is strongly recommended. This atmosphere is intermittently moved by the plants at a speed of 1-6 km/hour to circulate fresh air, prevent $CO_2$ depletion, facilitate temperature control, and provide some thigmonastical stimulation.

The plants may need other environmental features such as the presence of a steady state magnetic field. The exploration of these features should be an important mission for space station and lunar base plant growth experiments.

## MATERIALS ENGINEERING CONSIDERATIONS

All materials used in this system must be non-toxic to both plants and crew members. This includes the cylinders, sponge strips, feed lines, valves, circulating pumps, and other components that come in contact with the plants or the nutrient solution. Coated toxic materials are not appropriate due to the possibility of coating failure either by accidental impact or abrasion.

The foam-like sponge strips must be engineered for appropriate water retention and aeration properties required by their gravity environment. It would be advantageous to create the sponge out of a recyclable material that does not have to be imported from the earth.

All of the materials in contact with the plants and nutrient solutions must be highly resistant to corrosion. There should be little or no interaction between the materials, nutrient solutions, plant-released organic acids, and other compounds. Aluminum components in particular should be avoided because of their corrosion susceptibility and toxicity to plants.

# NATURAL VACUUM ELECTRONICS

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The ambient natural vacuum of space is proposed as a basis for electron valves. Each valve is an electron-controlling structure similar to a vacuum tube that is operated without a vacuum sustaining envelope.

The natural vacuum electron valves discussed offer a viable substitute for solid state devices. The natural vacuum valve is highly resistant to ionizing radiation, system generated electromagnetic pulse, current transients, and direct exposure to space conditions.

## INTRODUCTION

This paper proposes the use of the natural vacuum of space as a basis for electronic components. The space environment offers a natural vacuum that is suitable for the passage of electron beams over rather long distances. These electron beams can be used in electron valve components.

## PHYSICAL OPERATION

The basic natural vacuum component uses a stream of electrons that is emitted from a source. This electron stream travels through the ambient vacuum environment towards a positively charged destination. This basic flow corresponds to the flow of electrons within the diode vacuum tube of electronics history. In the diode vacuum tube, electrons are emitted by a heated cathode (thermionic emission). The emitted electrons are attracted across the vacuum by a positively charged metallic plate (the anode).

Natural vacuum components use this basic vacuum tube flow without the need for the vacuum sustaining enclosure. The cathode and anode are directly exposed to the ambient environment of space. The natural vacuum diode "tube" provides the service of rectification where an alternating current input is converted to pulses of direct current output.

Additional functions are provided by installing electron flow modifying structures in the electron beam. A metallic screen or grid placed in the electron beam between the cathode and anode provides a valving action. Varying the electric charge on the grid varies the electron beam current traveling between the cathode and the anode. Additional grids can be inserted into the electron flow to provide additional control and to isolate against undesired coupling of stray signals through the system.

This type of valving is quite familiar to people in electronics who have worked with multi-grid vacuum tubes. In the multigrid natural vacuum component, the artificial vacuum maintained by the glass tube is replaced by the ambient environment of space. Figure 1 shows a natural vacuum component with concentric grids.

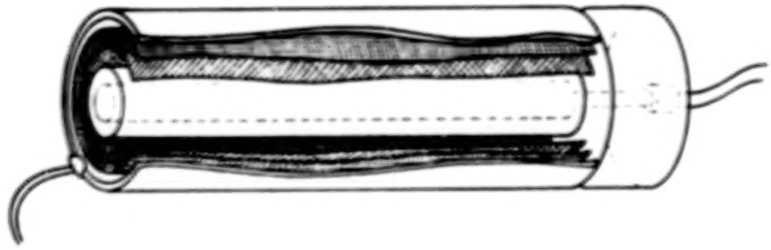

Figure 1. Natural vacuum component.

Magnetic fields can be used to control the electron beam also. Magnetic fields can direct, deflect, reverse, or contain the electron beam. A limited example of this control is the common television picture tube. The ambient environment of space frees the natural vacuum component from the heavy picture tube containment. A natural vacuum component can hang the electron gun and magnetic coils in an open structure.

## COMPONENT INNOVATIONS

Natural vacuum components are not limited to revivals of older vacuum tube designs. The large volume of the ambient space environment allows the component designer to produce components that are not practical in small artificial vacuum tube environments. For example, the grids within a natural vacuum component can be mechanically deployable. Different sets of grids can be deployed for different operating conditions.

The cathode can be heated by directly focused solar energy. This configuration avoids the problem of filament failure that was a common problem with terrestrial vacuum tubes.

The most profitable approach to natural vacuum electronics is to think of the ambient environment of space as a gymnasium for electrons. You can place electron sources, destinations (anodes), and beam modifying devices where you need them within the ambient environment. Your design does not have to be limited to connected electron-valving modules. A continuous structure can perform operations on a continuous electron beam that passes through the structure. Connections between the stages of processing are provided by the electron beam itself (although a "ground" return conductor is required as well).

## NON-ELECTRON CARRIERS

The natural vacuum approach can use carriers other than electrons. For example, accelerator technology can be used to generate streams of protons to carry signals or power. While protons may not have clearly-visible advantages over electrons, it is worthwhile to examine the concept of proton-based circuits.

One possible advantage of proton-based circuits is that protons are heavier than electrons. Thus a beam of protons would be less perturbed by ambient magnetic fields—a possible advantage in some space environments.

Atomic nuclei can also be used in natural vacuum systems. Similarly, larger masses of material can be pushed about by mass driver and magnetic separator types of "circuit" components. The scope of electronics is expanded to a continuum of natural vacuum functions that includes large scale energy beam manipulation and materials separation and processing.

## USES

The basic natural vacuum components provide the familiar active electronics functions of:

- switching
- rectification
- oscillation
- amplification
- detection
- mixing

These are the same functions provided by transistors or vacuum tubes. The natural vacuum component is a very robust replacement for conventional solid-state components. Entire electronic circuits can be built from natural vacuum components or natural vacuum components can be used in conjunction with conventional components.

## ADVANTAGES

Natural vacuum components are very robust devices that can survive the stresses of the space environment. Each natural vacuum component is a largely open-metal structure that is directly exposed to the vacuum environment. This structure is resistant to damage from ionizing radiation, system generated electromagnetic pulse (SGEMP), and micro-particle impacts.

The natural vacuum components can be installed outside of the spacecraft conserving interior space for other uses. (Fig. 2) These components can be operated at very high power levels when suitable radiative cooling means are provided.

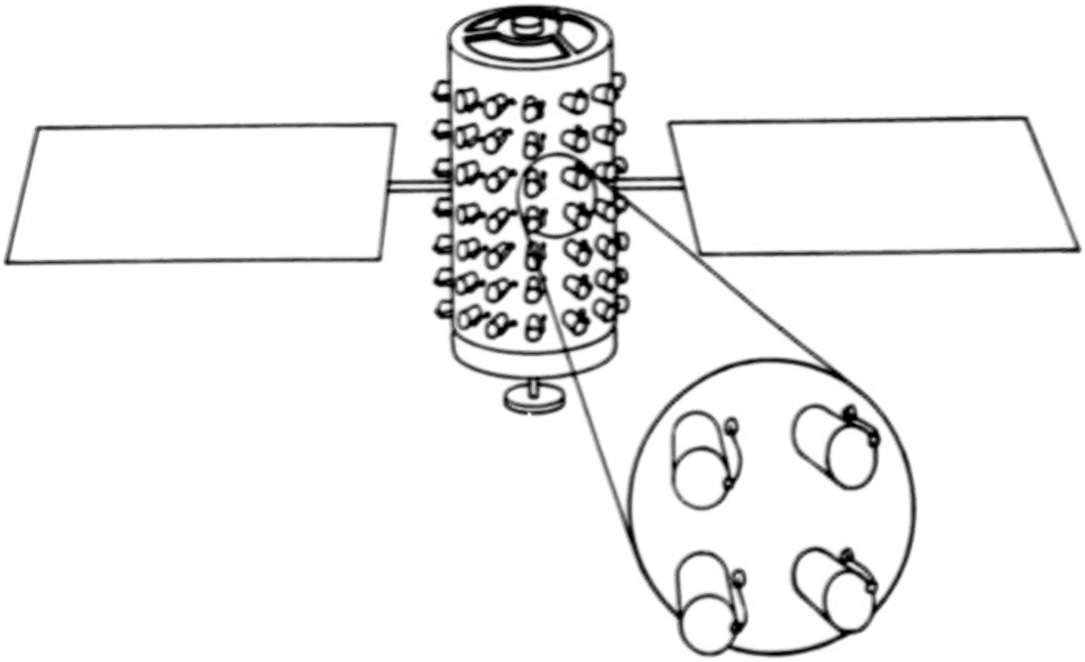

Figure 2. Configuration of externally mounted components.

The natural vacuum components can be made from space-derived metals and insulating materials, when these materials become available. This increases the appeal of the natural vacuum approach for worlds such as the moon that have excellent ambient vacuum environments and locally available materials.

## DISADVANTAGES

The operation of natural vacuum components could be negatively impacted by the poor quality of "vacuum" present in low earth orbit or near other bodies with atmospheres. One such impact is the blockage of the free flow of electrons. Erosion of the components due to the action of ambient molecules could occur as well.

Another potential problem is the size and weight of natural vacuum components that may be larger than their solid state equivalents.

## CONCLUSION

Natural vacuum components should be considered for space craft and lunar base designs. These components change the vacuum of space from a problem into an asset.

N91-22174

# LOW-TEMPERATURE PLASMA TECHNOLOGY AS PART OF A CLOSED-LOOP RESOURCE MANAGEMENT SYSTEM

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## ABSTRACT

More comprehensive resource management systems must be developed if the United States is to continue extraterrestrial exploration involving extended lengths of time in space. Testing was performed to obtain additional information concerning the feasibility of using a low-temperature plasma reactor as the centerpiece of a closed-loop processing (CLP) resource management system. Low- and high-ash carbon, freeze-dried human feces, sunflower, and plastic were processed in a low-temperature oxygen plasma reactor system in both stationary and agitated modes. Conversion of carbon, hydrogen, and nitrogen to gaseous species was determined for each of the test materials. Sample agitation greatly reduced the residence times required for species conversion. Virtually complete carbon conversion was achieved at residence times ranging from 2 to 8 hr. The inorganic matrix was unchanged by the processing technique. Based upon the results of this testing, scale-up and further testing of this technology are warranted.

## INTRODUCTION

More comprehensive resource management systems must be developed if the United States is to continue extraterrestrial exploration involving extended lengths of time in space. Three types of resource management systems are available for this type of mission: 1) systems in which consumables such as oxygen and food are not recycled, 2) totally closed-loop systems in which all consumables are recovered and reused in some manner, and 3) a combination of 1 and 2. The quantity of consumable material to be recycled (i.e., 50% or 90%) and/or ferried back and forth from Earth will be based primarily on the mass, volume, and energy requirements of the closed-loop processing (CLP) system. An extraterrestrial base will require a CLP system capable of integrating all functions necessary to support life in a remote setting, including manufacturing and biological by-product processing.

A low-temperature plasma reactor is capable of oxidizing or reducing the organic components of a stream while leaving the inorganic matrix unaffected. In effect, separation and conversion take place in a single reactor. The organic fraction of plants, human waste, and plastics can be converted to gases, while the inorganic fraction remains virtually unchanged. The inorganic materials can then be recycled directly to other operations or subsystems.

The feasibility of using this technology as part of a CLP resource management system can be measured by reactor size, efficiency, and energy requirements, as well as by the capability of the reactor to be closely integrated into the closed-loop system. Stationary-bed batch tests were performed previously at EERC to determine conversion rates and efficiencies for a variety of organic wastes (ref. 1). The results of these tests showed that the technology held promise as part of a CLP resource management system, with its ultimate potential dependent upon lowering residence time requirements. The current research focused on determining if the residence times could be reduced to a point where it is likely that scale-up and reactor system optimization could result in a small, energy-efficient system capable of being closely integrated into a closed-loop resource management system.

## METHOD AND MATERIALS

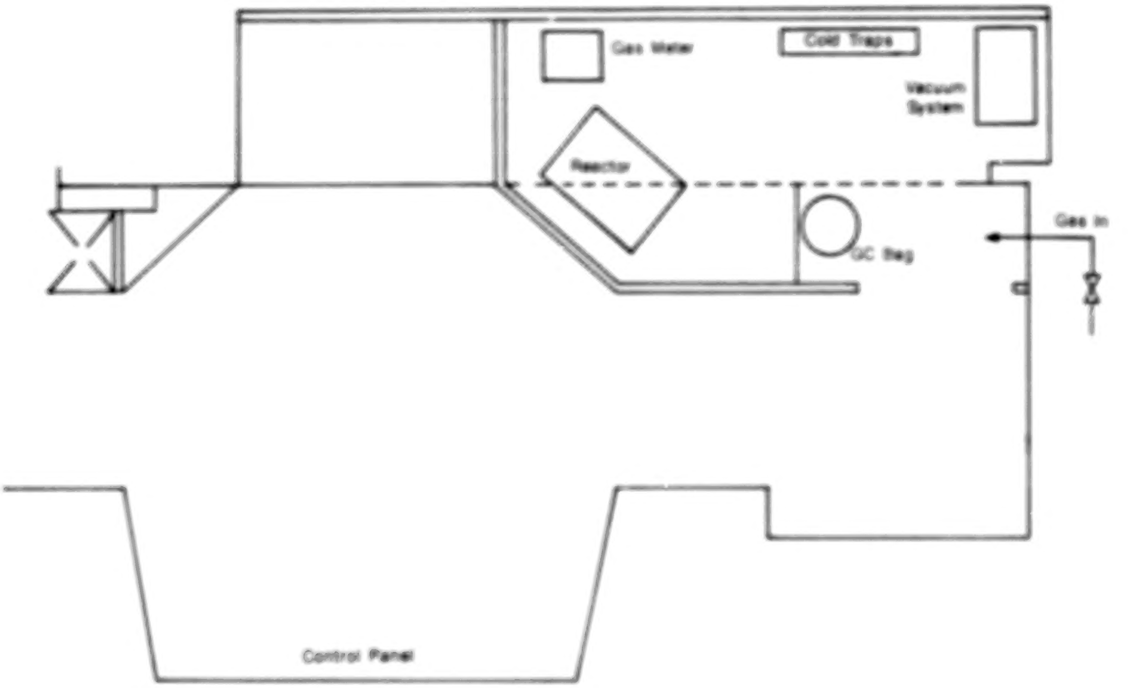
### Equipment

The low-temperature plasma tests were performed using three pieces of equipment: a radio frequency (rf) generator, a reactor center, and a vacuum system. The rf generator and the reactor center were manufactured by the International Plasma Corporation. The rf generator can generate up to 500 W of rf power at a frequency of 13.6 MHz. The reactor center houses the reactor chamber, the electrodes that generate the plasma field, and a control panel. The floor plan of the work area supporting this system is shown in Figure 1. The control panel is used to regulate the gas flow and the vacuum in the reactor chamber. It also balances the load on the rf generator. The quartz reactor chamber is eight inches long and four inches in diameter and has been modified to contain the hardware necessary for agitation during operation. A vacuum pulls the plasma-producing gas through the chamber, while the rf energy is transmitted across the electrodes surrounding the sides of the reactor chamber. The reactor chamber is shown in Figure 2.

As part of earlier research, the reactor chamber was fitted with three types of agitation. The first was a fritted quartz cylinder attached to a rotary coupling (the reactor agitation system originally proposed for this testing). Samples were placed inside this cylinder and tumbled during processing. Two other methods of sample agitation during processing were also established with the existing reactor system.

The second device consisted of a pair of rotating aluminum trays. The trays were each constructed with a lip on one longitudinal edge and were aligned as shown in Figure 3. They were attached to the rotary coupling and fitted inside the quartz reactor chamber. At this small scale, some of the particle sizes of the samples tested were not large enough to overcome the electrostatic forces which held them to the aluminum plates. To overcome this problem, the system was "internally" agitated. The method chosen was essentially a ball mill. An inert solid (sand or silica gel) was added to the test material to provide a mixing, crushing action. It also ensured that some type of sample would be present at extinction for analysis.

The third tumbling device consisted of a thin blade of aluminum which was attached to the rotary coupling. The blade continuously scraped the reactor chamber walls, thereby agitating the sample, as shown in Figure 4. This method also made use of the addition of an inert solid to the test materials for the purpose of providing further mixing action.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Cold Traps
Vacuum System
Reactor
GC Bag
Gas In
Control Panel

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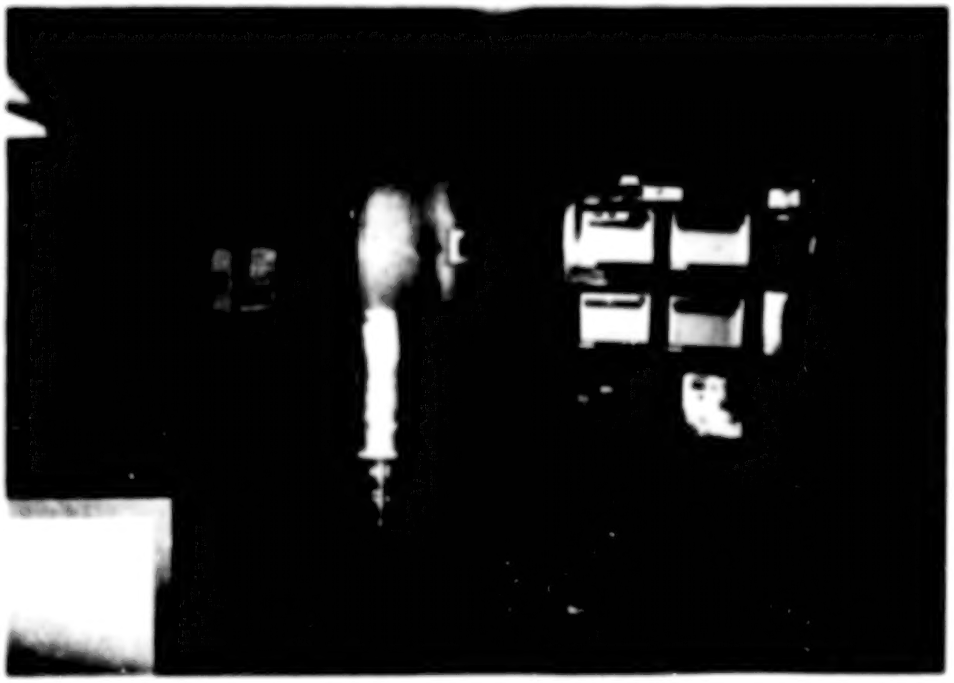

Figure 1. (a) Floor plan of low-temperature plasma system work area.
(b) Low-temperature plasma system control area.

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Figure 2. Low-temperature plasma system quartz reactor chamber.

During the tests, nominal reactor chamber pressure was held at 67 to 107 $N/m^2$, rf power was set at 100 W, and the gas flow rate was less than 0.1 cc/min.

**Experimental Method**

Irrespective of the agitation method used, sand or silica gel was added to the test material in a mass ratio of 9:1. The composite sample was then loaded into the reactor. When the aluminum trays were used, the sample was placed on the bottom tray. As the trays rotated, the sample was dumped from one tray to another, falling through the plasma as it did so. At the completion of the test, the trays were removed and the sample collected. Any sample that had fallen off of the trays was reclaimed from the bottom of the reactor chamber and added back to the product container.

In the case of the method involving the rotating aluminum blade, the sample was loaded onto, and collected from, the bottom of the reactor chamber. This method was used for the majority of the kinetics tests involving agitation.

To provide a yardstick against which to measure the results of the tests involving agitation, tests were also performed in a stationary mode. The same quartz reactor chamber was used for these tests, but without agitation. Instead, the sample was loaded onto quartz trays which were placed inside the reactor chamber.

Quality control measures taken during this study are described in reference 2.

feed). These test points were not included in the plots. As expected, conversion to gaseous products generally took place more rapidly when samples were agitated, although the difference between the results of the agitated and stationary tests was more pronounced for some of the test materials.

Figure 5 shows the conversion of carbon in low-ash carbon as a function of time. In this instance, there was a rather dramatic difference between the results of the stationary and agitated tests. This can be compared with the conversion of carbon in high-ash carbon as a function of time, shown in Figure 6. The same trend is noted in that higher conversion levels were attained for the high-ash carbon sample when it was processed in the agitated reactor configuration. However, under stationary conditions, carbon conversions were higher for the high-ash carbon than for the low-ash carbon.

Of the five test materials, it was more difficult to obtain a representative sample of freeze-dried human fecal material for analysis due to the variations in particle size within the composite sample. As such, only one viable test point was obtained for the agitated tests. Conclusions can still be drawn from the plot shown in Figure 7. From the fecal material data point, it appears that the carbon conversion rate was enhanced by agitation and was of the same order of magnitude as that observed with carbon.

The results of processing the sunflower samples are presented in Figure 8. Again, agitation enhanced the conversion of carbon to gaseous species. The carbon conversions obtained for the sunflower samples were higher than those of any other materials tested within the one-hour kinetics evaluation series.

Conversions of hydrogen and nitrogen to gaseous products were also determined for the sunflower samples and are shown in Figures 9 and 10, respectively. As was the case with the carbon, agitation hastened the conversion of both the hydrogen and the nitrogen. Figure 10 indicates that the nitrogen conversion rate was more rapid during the first 20 min of agitated processing.

The plastic processed consisted of ground plexiglas. During the agitated processing, the particles had a tendency to melt together, making it difficult to contact the sample with the plasma. However, one viable test point was obtained for the agitated processing (60 min). As Figure 11 shows, this value is less than the 57% conversion obtained during stationary processing for 60 min. This is most likely due to the fact that the sample did not agglomerate during the stationary processing.

Samples of each of the test materials were processed to extinction using the rotating aluminum plate reactor configuration. Due to the tendency of the silica gel to stick to the reactor plates as a result of static electricity, the processing was performed using sand as the inert matrix. The product samples could not be analyzed for carbon, hydrogen, and nitrogen content because not enough test material remained. The time necessary for complete extinction varied from sample to sample, with the plastic requiring approximately 8 hr and the remaining samples requiring 3 hr or less. The five test materials are shown before and after processing in Figures 12 through 16.

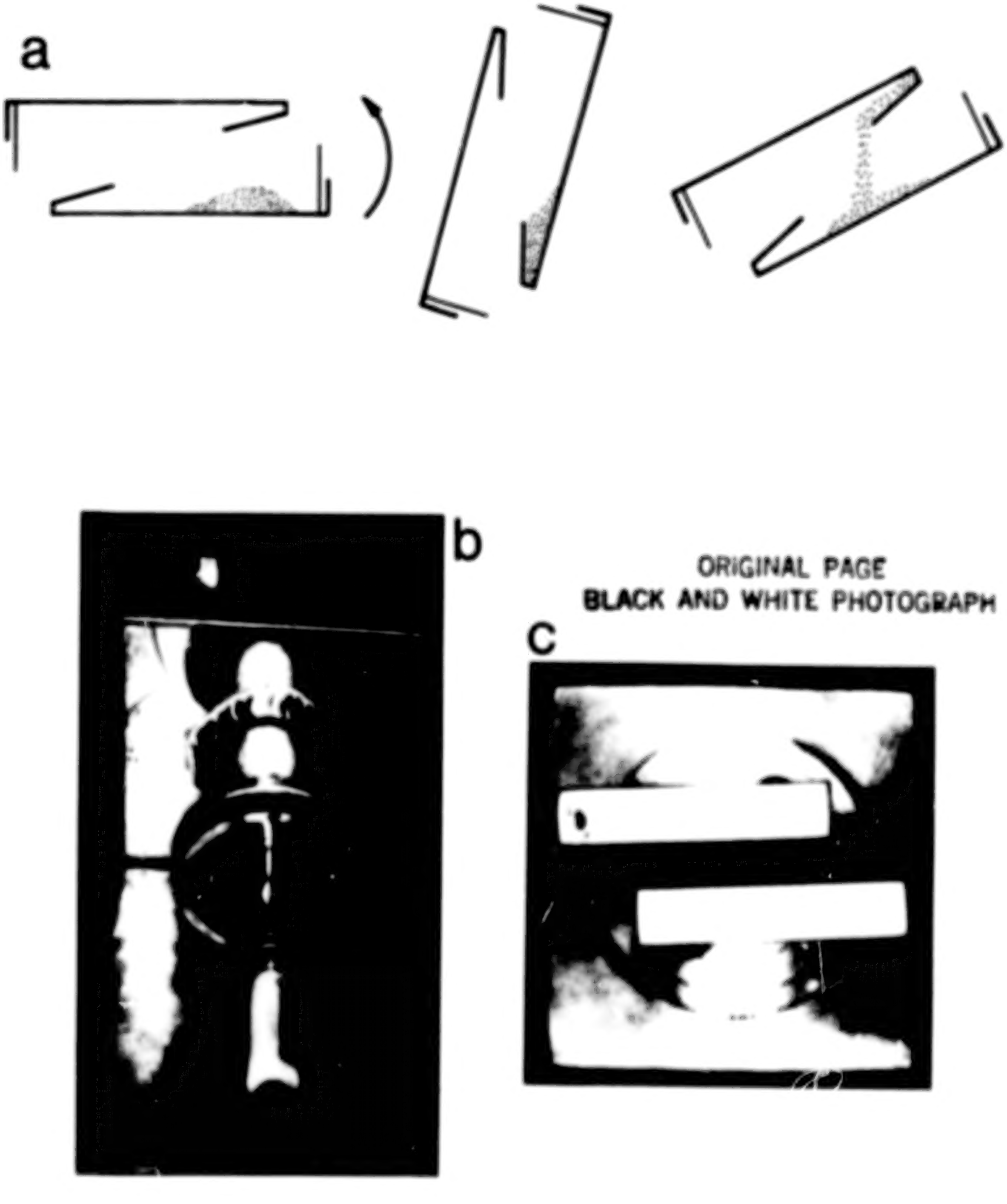

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Figure 3. (a) Diagram of rotating tray agitation system.
(b) Rotary coupling.
(c) Rotating tray agitation system.

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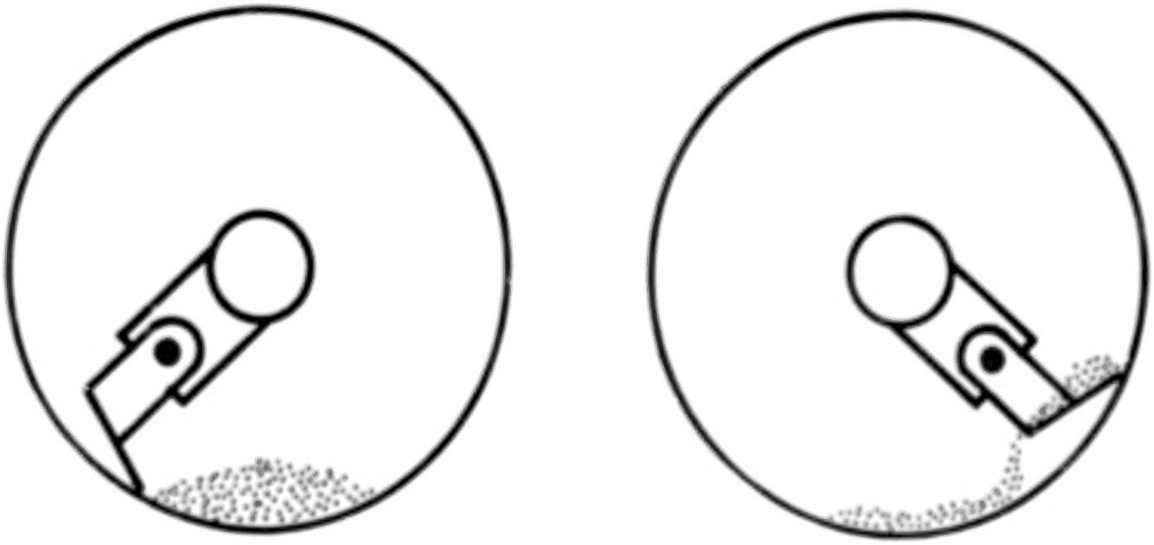

Figure 4. Diagram of aluminum blade agitation system.

**Test Matrix**

The tests were performed using oxygen plasma and the following materials:

1. Low-ash carbon
2. High-ash carbon
3. Freeze-dried human fecal material
4. Sunflower roots, stalk, and head
5. Plastic (ground plexiglas)

Tests were performed in both stationary and agitated modes for times of 20, 40, 60 min, and to extinction (less than 3 hr in most cases). The rotating aluminum blade method of agitation was used for the testing, except for the extinction tests, which were performed using the aluminum plate reactor. When possible, the product samples were analyzed for carbon, hydrogen, and nitrogen content.

## ANALYTICAL RESULTS AND INTERPRETATION

Carbon, hydrogen, and nitrogen (CHN) contents were determined for the five feed samples and 29 of the products. The theoretical limits of calculations involving these results are the same as the detectability limits of the analytical instrumentation (i.e., ±0.5% for carbon and ±0.1% for hydrogen and nitrogen).

Conversions of the carbon, hydrogen, and nitrogen to gaseous species were determined using the CHN information. The conversion data were plotted as functions of residence time for ease of interpretation. In some cases, negative conversions were calculated (i.e., the analytical results indicated that more carbon was present in the product than had been present in the

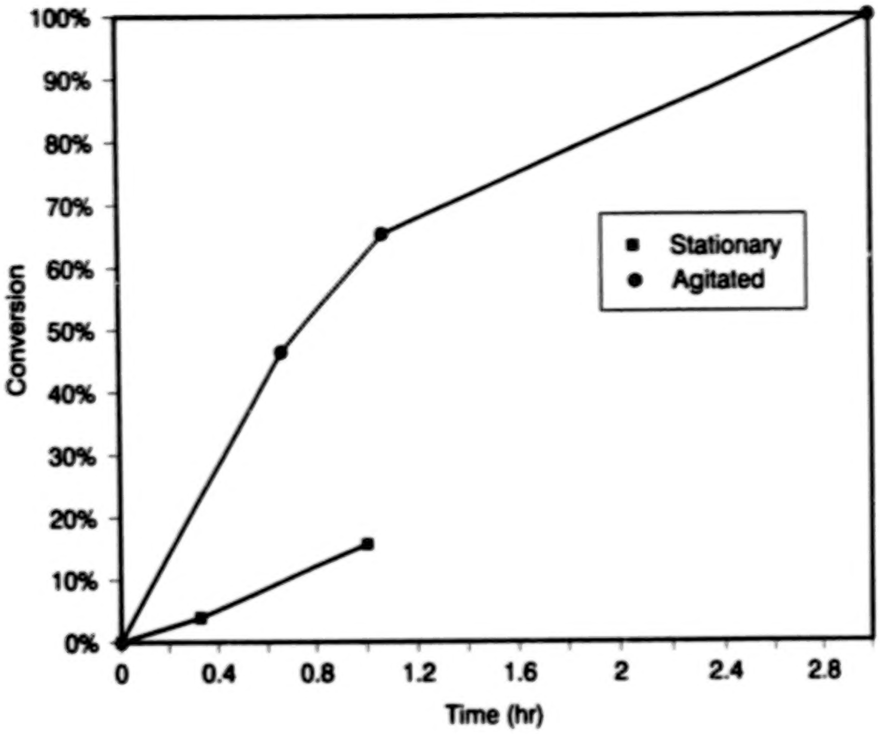

Figure 5. Carbon conversion in low-ash carbon sample as a function of time.

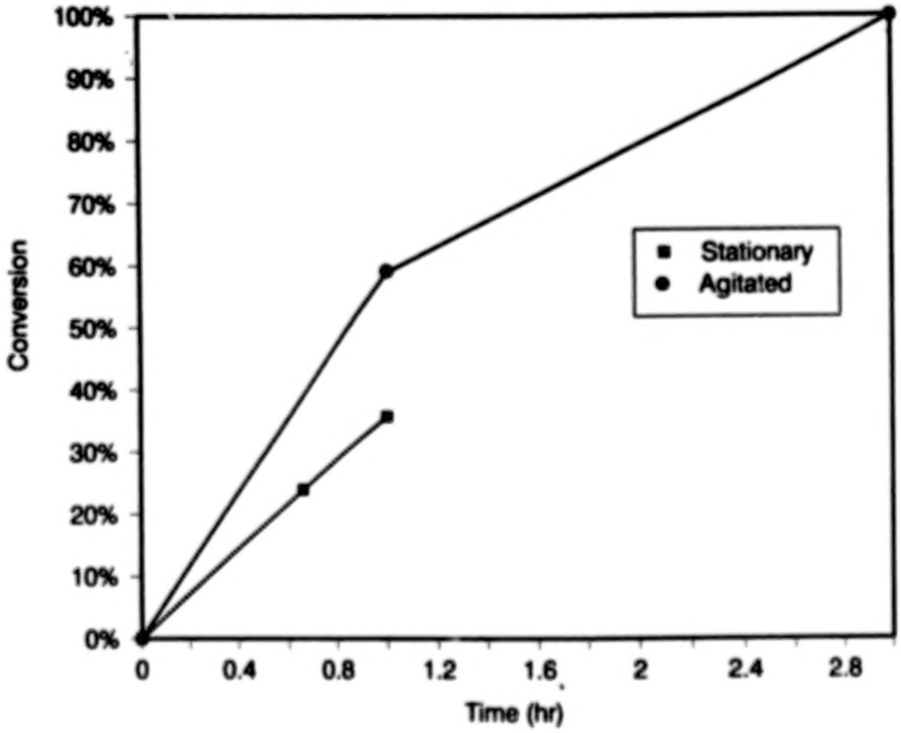

Figure 6. Carbon conversion in high-ash carbon sample as a function of time.

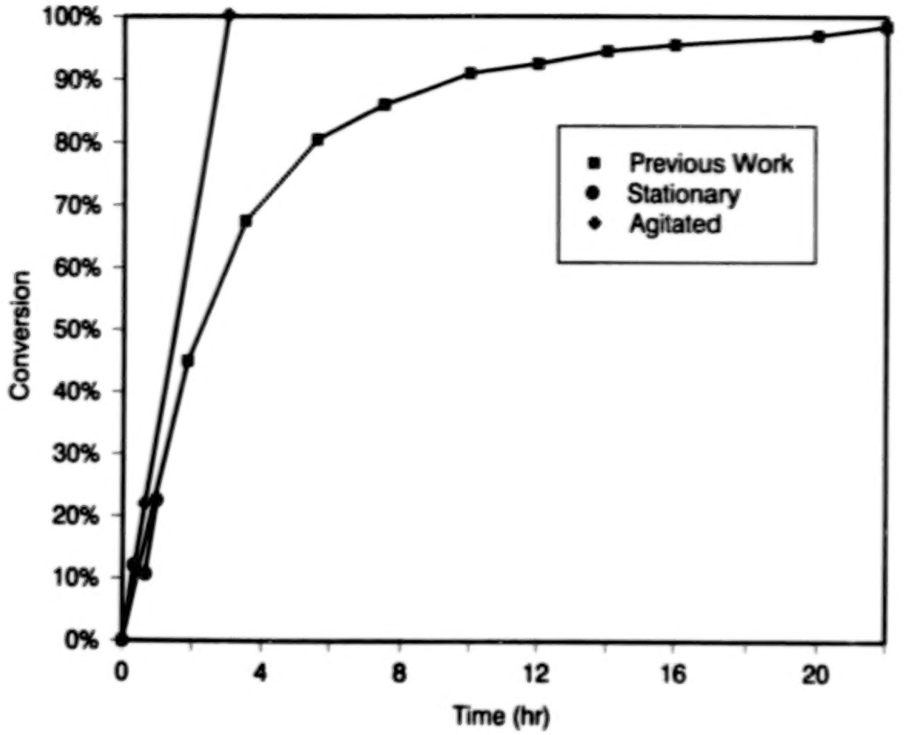

Figure 7. Carbon conversion in feces sample as a function of time.

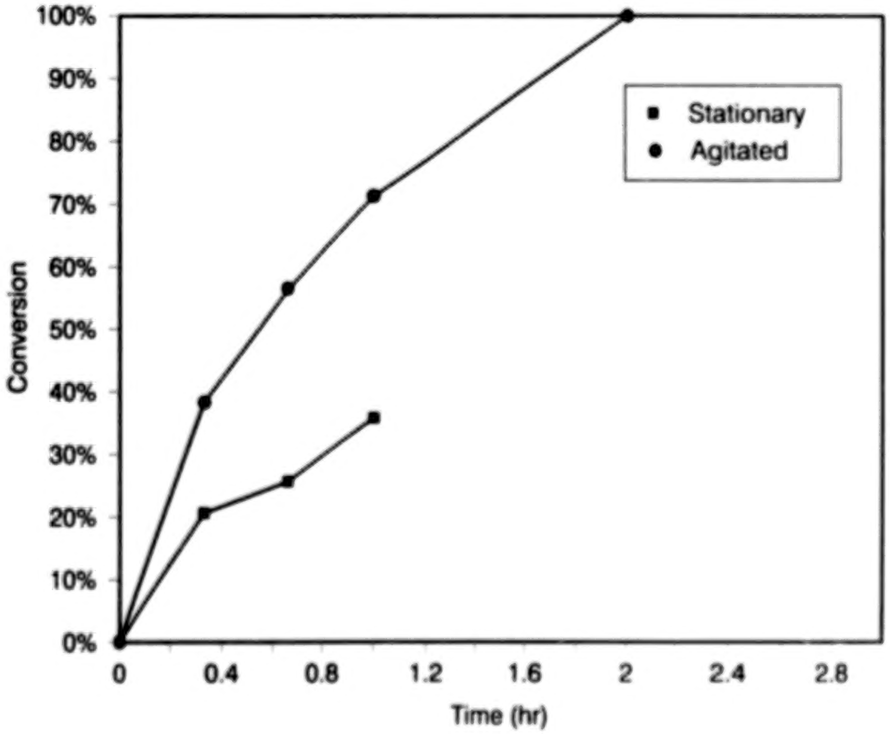

Figure 8. Carbon conversion in sunflower sample as a function of time.

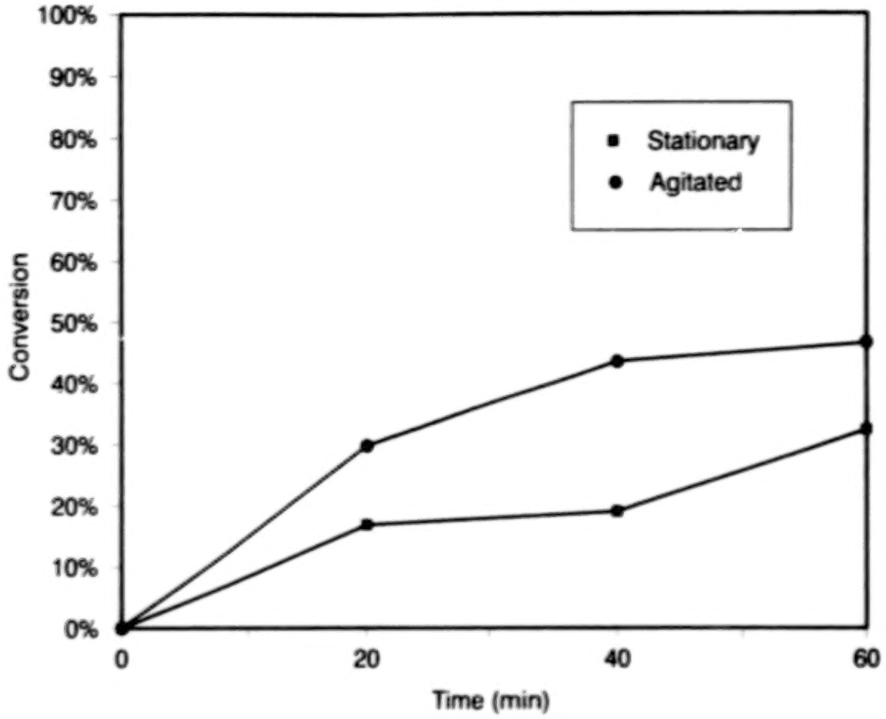

Figure 9. Hydrogen conversion in sunflower sample as a function of time.

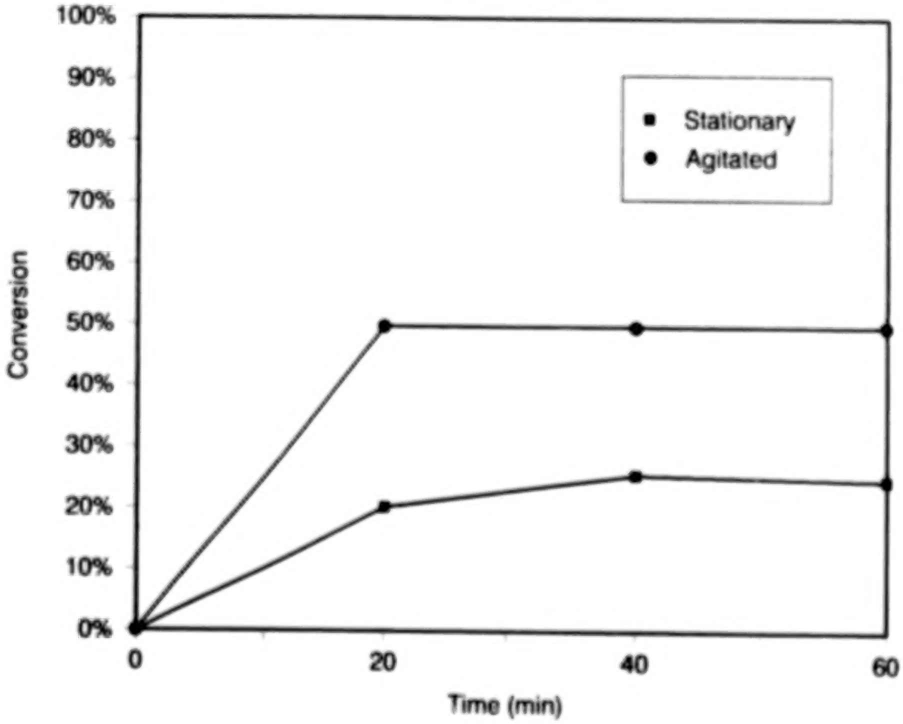

Figure 10. Nitrogen conversion in sunflower sample as a function of time.

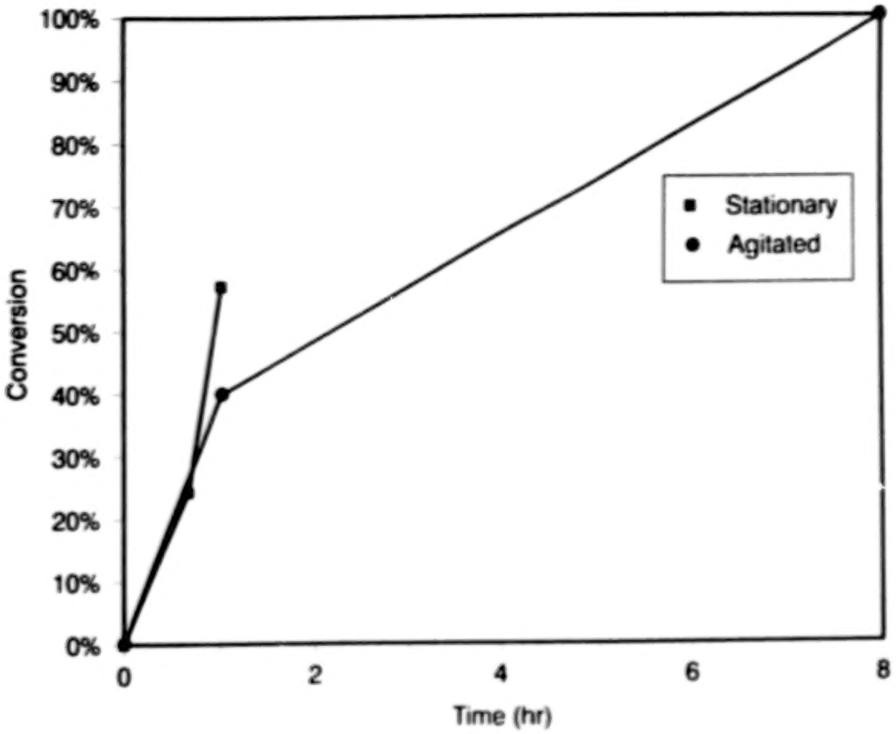

Figure 11. Carbon conversion in plastic sample as a function of time.

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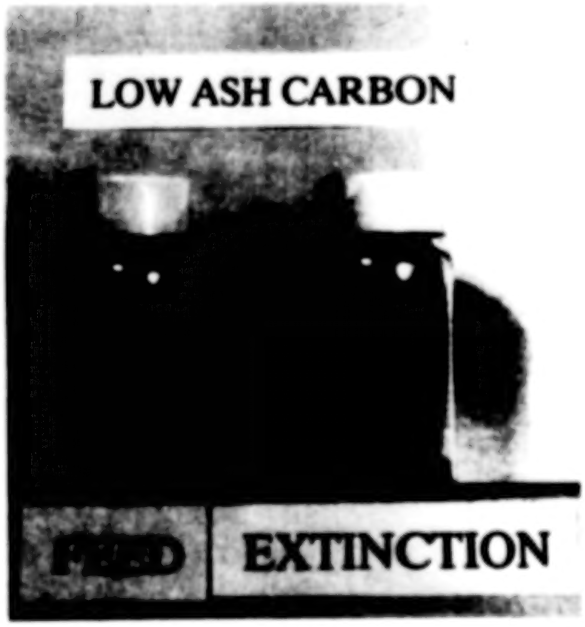

LOW ASH CARBON
FEED
EXTINCTION

Figure 12. Low-ash carbon sample before and after processing to extinction.

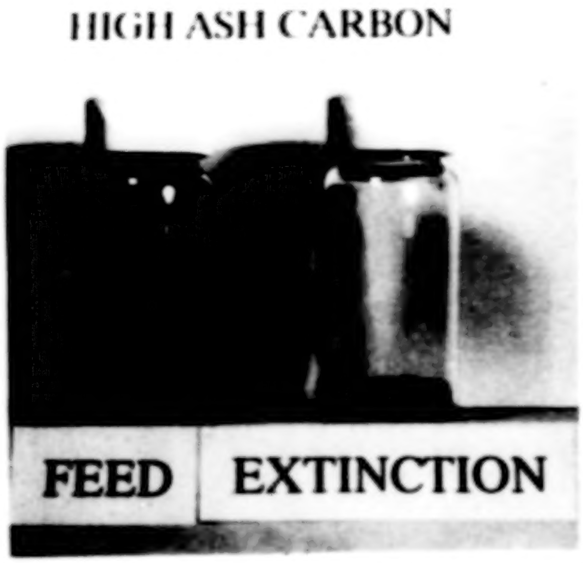
HIGH ASH CARBON
FEED
EXTINCTION

Figure 13. High-ash carbon sample before and after processing to extinction.

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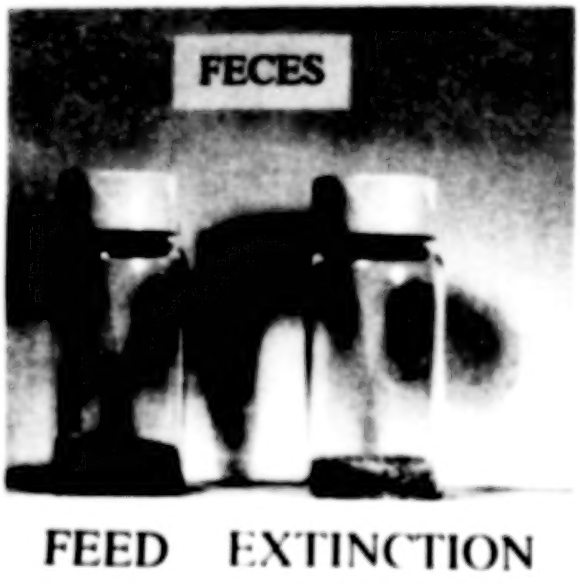
FECES
FEED
EXTINCTION

Figure 14. Freeze-dried feces sample before and after processing to extinction.

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SUNFLOWER

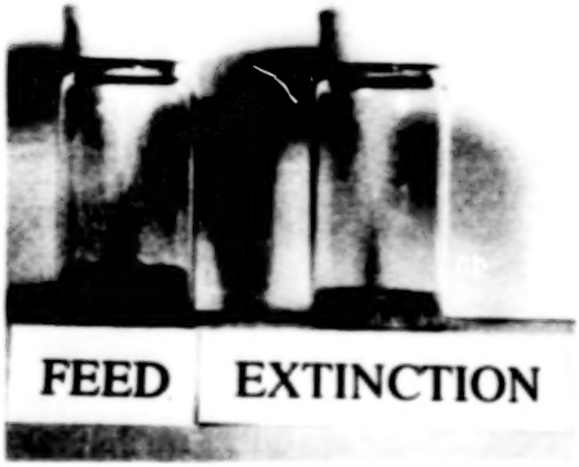
FEED
EXTINCTION

Figure 15. Sunflower sample before and after processing to extinction.

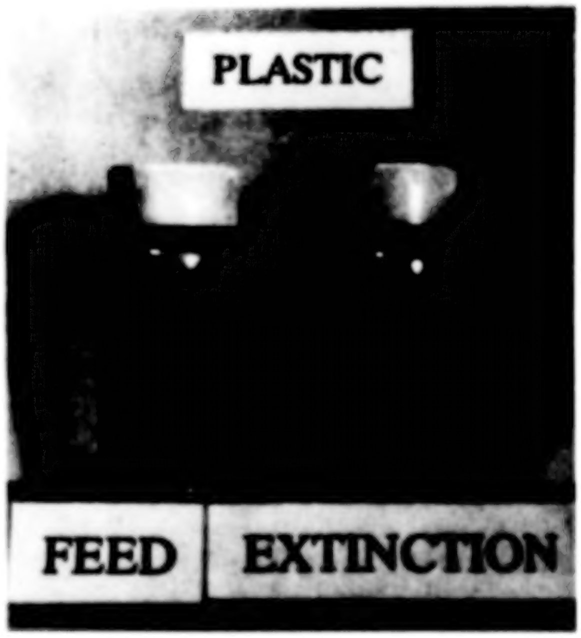
PLASTIC
FEED
EXTINCTION

Figure 16. Plastic sample before and after processing to extinction.

The results obtained during this research were compared to those obtained during earlier bench-scale testing at EERC (ref. 1). The comparison is summarized in Table 1. The products of the agitated runs were very similar in conversion and solubility characteristics to the products of the stationary tests performed during earlier work. The ranges of residence times required for complete carbon conversion are indicated in Figure 17. As the figure shows, the residence times required to completely convert the carbon were reduced dramatically by agitation, in one case by a factor of 12.

Early in the testing, it was observed that the carbon conversion rates were related to particle size as viewed by gross examination. As a result, select samples were submitted for size analysis. The carbon conversion rates of the three samples analyzed were not related solely to particle size as determined by small particle analysis. This discrepancy suggests that the conversion rate may be related instead to other physical parameters such as surface area and/or physical ash or residue formation or structure relative to the reactant species. Sample preparation will therefore be a necessary area of research in defining efficiencies for this system application.

Samples of the inert matrices (i.e., sand and silica gel) taken before and after processing were analyzed using x-ray diffraction and x-ray fluorescence. The results of these analyses indicated that the processing had not caused any significant morphological changes.

## CONCLUSIONS

The results of this testing indicate that the agitated low-temperature plasma reactor system successfully converted carbon, hydrogen, and nitrogen to gaseous products at residence times that were approximately ten times shorter than those achieved by stationary processing. The inorganic matrix present was virtually unchanged by the processing technique. It is concluded that, at this stage of development, this processing technique is feasible for use as part of a CLP resource management system. Further investigation to optimize reactor configuration, energy requirements, and scale-up are warranted based upon the results of this work.

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2. Hetland, M.D., Rindt, J.R., Jones, F.A., and Sauer, R.S., "Further Investigation of the Feasibility of Applying Low-Temperature Plasma Technology to a Closed-Loop Processing Resource Management System," Final report submitted to NASA, Contract No. NASW-4455, Project No. 10-49501 (April 1990).

## TABLE 1

## SUMMARY OF CONVERSION AND SOLUBILITY DATA FROM BOTH CURRENT AND EARLIER WORK (REF. 1)

| | Feces (earlier work) | Feces | Sunflower (earlier work) | Sunflower | Low-Ash Carbon | High-Ash Carbon | Plastic (earlier work) | Plastic |
|---|---|---|---|---|---|---|---|---|
| % Residue[a] | 31.00 | 29.82 | 18.52 | 14.84 | 4.24 | 29.85 | 1.40 | <1.00 |
| Time (hr in plasma) | 22 ± 1 | 3 ± 0.1 | 24 ± 4 | 2 ± 0.1 | 3 ± 0.5 | 3 ± 0.5 | 48 ± 6 | 8 ± 2 |
| CHN Analysis of Residue[b] | | | | | | | | |
| % Carbon | 5.205 ± 0.145 | BDL[d] | 5.220 ± 0.27 | BDL | BDL | BDL | --- | BDL |
| % Nitrogen | 0.885 ± 0.045 | BDL | 0.440 ± 0.04 | BDL | BDL | BDL | --- | BDL |
| % Hydrogen | 1.380 ± 0.06 | BDL | 1.655 ± 0.095 | BDL | BDL | BDL | --- | BDL |
| % Conversion[c] | 98.39 | 100[e] | 99.03 | 100 | 100 | 100 | 98.60 | 100 |
| % Solubility, $H_2O$ | 32.33 | ---[f] | 82.36 | 84.09 | 46.67 | 16.67 | --- | --- |
| % Solubility, 6M HCl | 73.86 | --- | --- | 94.44 | 70.83 | 22.22 | --- | --- |

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[a] Based on wt. of residue out of plasma reactor/wt. of samples into reactor

[b] Standard CHN on a Control Equipment Corp. unit

[c] Based on
$$1 - \frac{\text{residue - carbon wt. in residue - non-organic wt.}}{\text{sample wt. - non-organic wt.}}$$

[d] Below detectability limits

[e] None detected; i.e., in/out indicated 100% loss

[f] Not enough sample for analysis

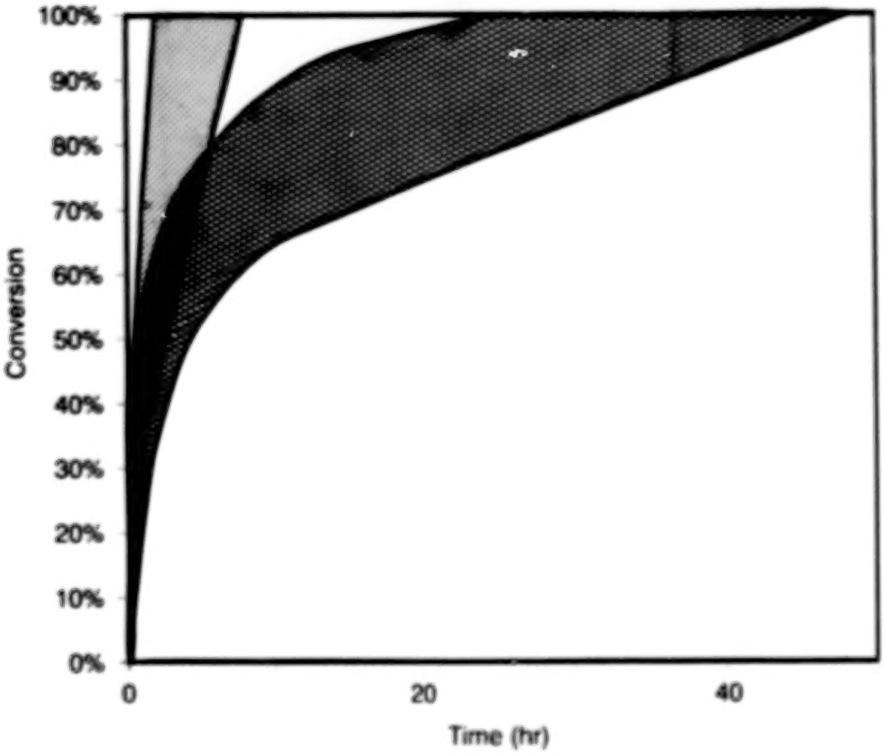

Figure 17. Ranges of residence times required for complete carbon conversion.

N91-22175

# CHARACTERIZATION OF BLOOD DRAWN RAPIDLY FOR USE IN BLOOD VOLUME EXPANSION STUDIES: AN ANIMAL MODEL FOR SIMULATED WEIGHTLESSNESS

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This study demonstrates that up to 8ml of blood can be drawn from donor rats without significantly increasing volume and stress sensitive hormones and thus, can be used for volume expansion studies. Infusion of whole blood allows more physiological changes than can be seen with volume expansion by saline or other ionic solutions. Furthermore, the infusion of whole blood to induce hypervolemia may provide an improved model to study the fluid balance and control mechanisms operative in weightlessness. In this study, blood samples were drawn as rapidly as possible from femoral artery catheters (microrenathane) chronically implanted in Sprague Dawley rats and analyzed for hematocrit, plasma sodium, potassium, osmolality, corticosterone, epinephrine, norepinephrine and vasopressin. The levels were found to be comparable to those of normal rats. Future studies of simulated weightlessness utilizing compatible and physiologically comparable whole blood will provide data that can be used to develop protective measures for the debilitating effects of space flight.

## INTRODUCTION

Blood volume expansion studies have been conducted to examine fluid adaptation and cardiovascular control mechanisms using numerous fluids including hypertonic, hypotonic and isotonic saline, water, dextran solution, albumin and polyethylene glycol (references 1,2). Less frequently, investigators have used whole blood, plasma or packed red blood cells from compatible donors (references 3,4). Infusion of whole blood allows the composition of the recipient's blood to remain relatively unaltered. Infusion of saline or other solutions may cause intracompartmental fluid shifts that modify the normal physiological adaptive processes. To prevent the introduction of aberrant hormones and ions that may alter or skew the adaptation processes, the blood infused must be as close as possible in composition to that of the recipient. It is well known that surgery, handling, restraint, injections and commonly used blood sampling techniques can cause fluctuations and changes in the levels of plasma catecholamines and other stress and volume dependent hormones, such as ACTH, renin, corticosterone and prolactin (references 5-8). However, little work has been done to determine at what point in time and/or at what quantity of blood drawn, the blood levels of the volume and stress sensitive hormones change significantly.

In these studies, blood samples were drawn as rapidly as possible from femoral artery cannulated unanesthetized rats. The blood samples were analyzed for hematocrit, plasma sodium, potassium, osmolality, corticosterone, epinephrine, norepinephrine and vasopressin. The data were analyzed in regard to the volume drawn to examine the effects of blood sampling on the levels of hormones and electrolytes listed above.

## METHOD

Male Sprague Dawley rats weighing between 350-400 grams were obtained from Charles River Laboratories (Raleigh, NC). All the rats were housed in single cages and maintained on a 12:12 hour light:dark cycle, with the lighted phase beginning at 6 a.m. They were used in the studies when they achieved a minimum body weight of 515 grams, previously determined to be an optimum size for cannulation. Standard rat chow and water were available ad libitum throughout the experiment. Room temperature and humidity were controlled as specified by the National Institutes of Health Animal Care Guide (reference 9).

The rats were anesthetized using a 1:1 mixture of KETASET (ketamine hydrochloride- Aveco Co., Inc.) and GEMINI (xylazine- The Butler Co.) for the cannulation procedure. The cocktail was injected (0.1ml/100gr) intramuscularly (references 10,11). Microrenathane (i.d. 0.025" o.d. 0.040"- Braintree Scientific, Inc.) catheters were inserted into the femoral artery in the right leg of 17 rats and from there led subcutaneously under the skin to emerge from the back of the neck. Microrenathane has been reported to reduce the probability of intravascular thrombosis (reference 12). The catheters were filled with heparin to maintain patency, heat sealed and coiled in a protective stainless steel button on the animal's back. Immediately after surgery, the animals received 0.1ml/100gr injection of DI-TRIM (trimethoprim sulfadiazim, Syntex Animal Health) subcutaneously, as a prophylactic agent against infection (reference 13). The rats were allowed to recover from the surgery for one week. During this week, they were conditioned to the manipulation of their catheters by daily handling, in the morning. This has been reported to prevent the stress-induced release of corticosterone (reference 14). Non-specific stressful stimuli (i.e., sudden or loud noises, excessive movement, unnecessary handling) were kept to a minimum during the experiment.

At the end of the recovery period, blood samples (3ml samples, n=4; 6ml samples, n=7 or 8ml samples, n=6) were drawn as quickly as possible from conscious animals. The blood samples were obtained between 9 a.m. and 2 p.m. to avoid the effects of circadian variations (reference 15). Prior to obtaining the sample for the assays, the heparin solution filling the dead space of the cannula (about 0.25ml) was removed. Approximately one-third of each of the blood samples was mixed with the anticoagulant EGTA (Ethylene glycol-bis tetraacetic acid) for the catecholamine assay specimen. Approximately 150ul was used to fill three heparinized microhematocrit tubes and the remaining portion of whole blood was mixed with lithium heparin for the other assays. The blood was centrifuged immediately (1500 rpm) at 4°C. The EGTA and heparinized plasma specimens were removed from the red blood cells and frozen in separate microcentrifuge tubes at -20°C until assayed.

The hematocrit was assayed using a previously described microhematocrit procedure (reference 15). The plasma osmolality was determined using vapor pressure osmometry (reference 16). The plasma electrolytes (sodium and potassium) were assayed by flame photometry (reference 17). The catecholamines (epinephrine and norepinephrine were analyzed using a radioenzymatic assay method (reference 18). The other hormones, corticosterone and arginine vasopressin, were determined using a radioimmunoassay method (references 19,20).

### Statistics

The data was analyzed in two different ways. Analysis of the statistical difference between the 3ml, 6ml and 8ml samples was performed by one-way analysis of variance. The Mann Whitney Confidence Interval and Test was used to compare each of the blood parameters and the blood sampling time for the 3ml blood sample to the 6ml and the 8ml blood samples individually, since the 3ml sample was the best control for these experiments. Differences were considered significant when $p < 0.05$ (reference 21).

## RESULTS

One way analysis of variance was used to determine the statistically significant difference between each of the blood parameters for the volumes of blood obtained. The F test was calculated at a 95% confidence interval for the mean differences at each volume collected (3ml, 6ml 8ml) for each of the blood assays (hematocrit, plasma sodium, potassium, epinephrine, norepinephrine, osmolality, corticosterone and vasopressin) and the time required to obtain the volume. The observed F for each of the blood parameters was less than the critical F (19.4) determined from the F table [hematocrit: F(2,16)= 0.13; osmolality: F(2,16)= 0.29; sodium: F(2,15)= 5.26; potassium: F(2,15)= 12.51; corticosterone: F(2,16)= 0.68; epinephrine: F(2,16)= 1.37; norepinephrine: F(2,16)= 1.54; vasopressin: F(2,16)= 0.07] including time: F(2,16)= 2.89]. Therefore, there is no significant difference between the groups (i.e., blood volumes drawn).

The Mann Whitney Confidence Interval and Test was also computed for the mean differences in the blood parameters listed above. In this analysis, the 3ml blood samples were compared versus first the 6ml blood samples and then versus the 8ml blood samples. The time required to draw the volumes ranged from 17.2 (3ml sample) to 106 seconds (8ml sample). The time required to draw sufficient blood from the donor animals does increase as a function of time, as anticipated. However, there was only a significant difference in the time required to draw the 8ml sample as compared to the 3ml sample, not the 6ml sample. Interestingly, there was a significant difference in the sodium and potassium levels for the 3ml blood samples versus both the 6ml and the 8ml samples (Table 1).

**TABLE 1. Blood Volumes Drawn in Respect to Time, Plasma Electrolytes (Sodium & Potassium), Osmolality and Hematocrit Levels.**

| Volume (ml) | 3<br>(n=4) | 6<br>(n=7) | 8<br>(n=6) | Reference Values |
|---|---|---|---|---|
| Time (sec.) | 28.8 ± 7.26 | 49.1 ± 7.54 | 61.3 ± 10.2* | ---------- |
| Sodium (mEq/L) | 136.5 ± 1.00+ | 141.9 ± 1.33* | 142.7 ± 0.9* | 138-148 ($\bar{x}$ =143) |
| Potassium (mEq/L) | 3.9 ± 0.29+ | 5.01 ± 0.165* | 4.39 ± 0.11* | 4.0-9.2 ($\bar{x}$ = 6.6) |
| Osmolality (mosm/kg) | 281.9 ± 7.5 | 277.4 ± 3.37 | 278.2 ± 2.19 | 294 ± 1.4 |
| Hematocrit (%) | 41.9 ± 3.83 | 42.57 ± 3.17 | 41.8 ± 0.93 | 49 ± 4 |

Values are ± S.E.M.
Statistical analysis performed using the Mann Whitney U Test.
+ n= 3
* Significant difference compared versus 3ml blood sample, $p< 0.05$

The values were comparable with the range of the reference values reported for normal rats (reference 22). The values for hematocrit, plasma osmolality, epinephrine, norepinephrine, corticosterone and vasopressin measured in the 3ml samples were not significantly different from the 6ml or 8ml samples (see Figures 1,2 and 3).

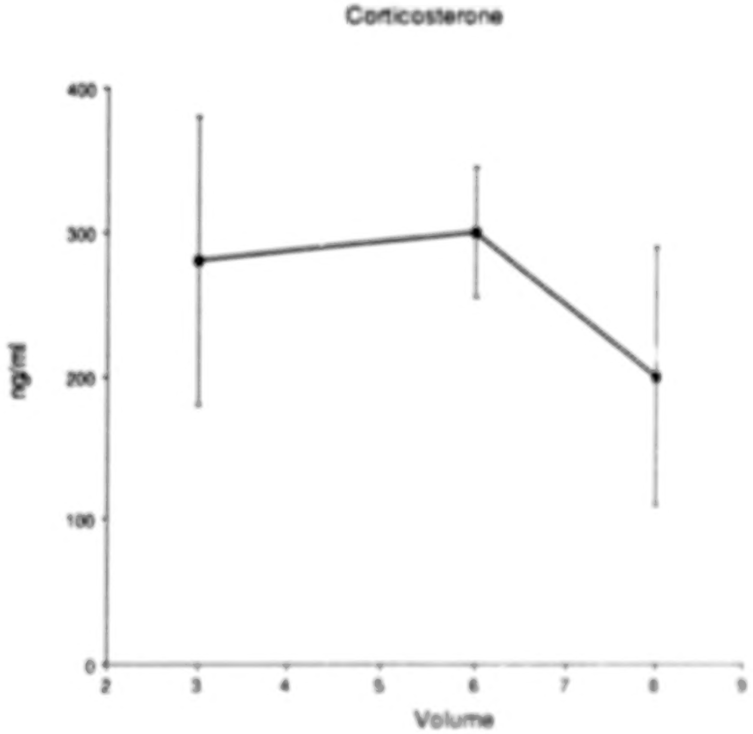

FIGURE 1. Plasma levels of corticosterone in 3ml, 6ml, and 8ml sample volumes. Samples were removed from cannulated animals as quickly as possible (< 2 minutes) and analyzed for corticosterone using a radioimmunoassay. There was no significant difference between the three groups.

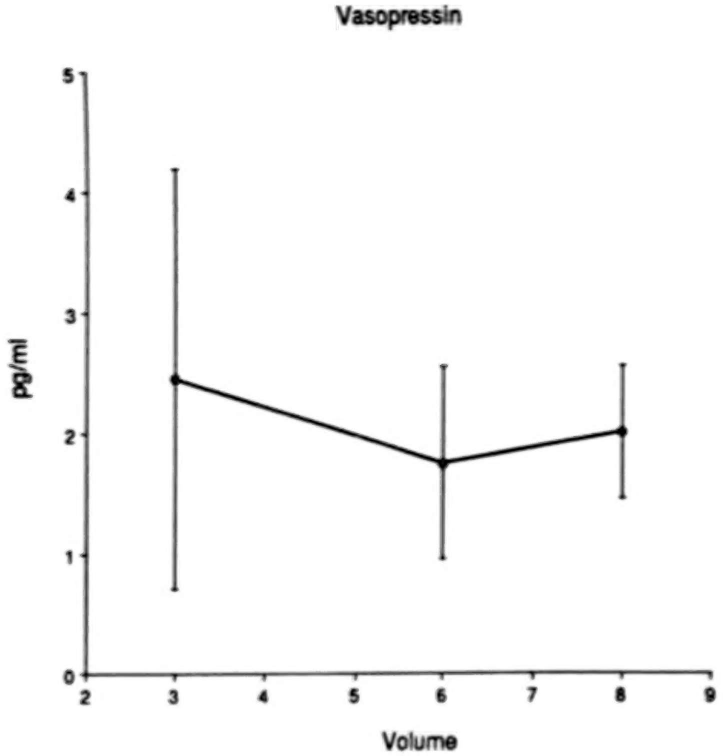

**FIGURE 2.** Plasma levels of vasopressin in 3ml, 6ml, and 8ml sample volumes. Samples were removed from cannulated animals as quickly as possible (< 2 minutes) and analyzed for vasopressin using a radioimmunoassay. There was no significant difference between the three groups.

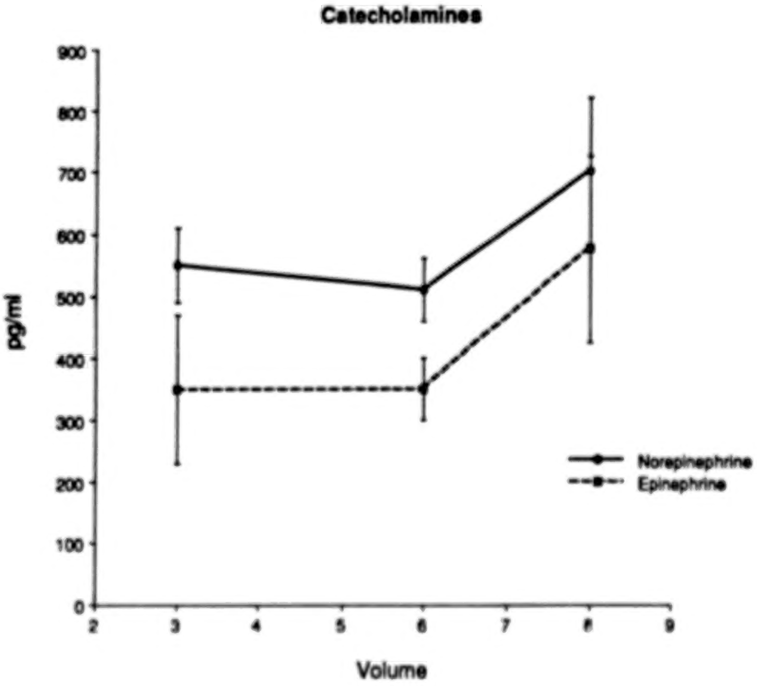

**FIGURE 3.** Plasma levels of catecholamimes (epinephrine & norepinephrine) in 3ml, 6ml, and 8ml samples volumes. Samples were removed from cannulated animals as quickly as possible and analyzed for the catecholamines using a radioenzymatic assay. There was no significant difference between the three groups.

Furthermore, the values for corticosterone, epinephrine and norepinephrine appear to be within the limits given in other investigations (references 7,8).

## DISCUSSION

As expected, the time required to draw blood from donor animals did increase as a function of volume. Surprisingly, there was a significant difference ($p < 0.05$) in the sodium and potassium levels for the 3ml blood samples versus the 6ml and 8ml samples. However, these values were comparable to the reference values observed in blood samples obtained from control animals in other studies (reference 22). The mean hematocrit, osmolality, corticosterone, epinephrine, norepinephrine and vasopressin levels for the 3ml, 6ml and 8ml samples were not significantly different from each other. Moreover, the mean values for the volume and stress sensitive hormones appear to be comparable to the reference levels reported in other studies (references 7,8).

Investigators have studied the numerous and complex adrenocortical and adrenomedullary responses to hemorrhage and transfusion, but, little has been done regarding the plasma electrolyte responses (reference 23). It has been postulated that the electrolyte responses during weightlessness are the result of endocrine changes acting on the kidney to restore "normal" cardiopulmonary volume. The electrolyte response and the endocrine interrelationships need to be clarified.

Plasma catecholamines are a sensitive indicator of sympatho-adrenal medullary activity. The source of the epinephrine has been demonstrated to be predominately from the adrenal medulla (reference 8). The source of the norepinephrine has been demonstrated to be primarily from the sympathetic nerves (reference 8). A recognized component of the immediate physiological response to stress is the release of catecholamines into the blood. Conventional animal manipulations are well known stressors that elicit changes which release catecholamines. The opening of the cage, transfer of animals into another room and short term handling has been reported (reference 24) to cause epinephrine and norepinephrine levels to increase as much as eight fold. In the current studies, withdrawing blood from the femoral artery catheter elicited an increase in the catecholamine levels that correspond with the animal manipulations reported in other studies. Although the values tend to increase concomitant with the volume of blood drawn from the animal, the values for the 3ml samples are not significantly different from the values for the 6ml or 8ml samples.

Corticosterone is secreted by the adrenal cortices of rats and controlled by a complex negative feedback system involving the central nervous system, hypothalamus, pituitary and adrenals (reference 23). Increased plasma corticosterone levels can be induced by the same stress inducing manipulations that increase plasma catecholamine levels (references 7,8). Corticosterone is also a volume sensitive hormone. Studies by Gann in dogs (ref. 23) have demonstrated that plasma cortisol levels are the function of a semi-logarithmic relationship with ACTH and an exponential relationship with the magnitude and rate of the hemorrhage. Other studies have shown that up to 10 minutes is required to allow stimulation of the aforementioned feedback mechanisms to significantly alter the blood levels of corticosterone (reference 25). In the current study, the values for corticosterone in the 3ml samples was not significantly different from the levels in the 6ml or 8ml samples. These samples were drawn in less than 2 minutes, which was not adequate time to stimulate the mechanisms that would significantly influence the corticosterone blood levels.

Arginine vasopressin (AVP) is a hypothalamic hormone, synthesized primarily by hypothalmic neurosecretory neurons whose axons terminate in the pars nervosa of the posterior lobe of the pituitary gland. Vasopressin secretion is controlled by osmotic and nonosmotic factors (reference 26). Vasopressin release is stimulated by a fall in blood pressure, an increase in plasma osmolality or a decrease in plasma volume (reference 27). There is an exponential nature to AVP release in response to hypovolemia, much like the corticosterone response (reference 28). The threshold for stimulation of vasopressin release in hypovolemia is generally to be between 10 and 20% of the blood volume (reference 27). Ginsberg and Brown (ref. 29) studied changes in vasopressin activity after slow and rapid hemorrhages in anesthesized animals and found greater increases in the antidiuretic activity in blood samples drawn more slowly. In the current study, there were no significant differences in the vasopressin levels of the 3ml, 6ml and 8ml samples, which represents hemorrhages of 7.9%, 15.8% and 21% respectively. It can be speculated that in the current study, the degree of hemorrhage or the time for collection was not sufficient to elicit an increase in plasma vasopressin.

This study and the values reported in other studies seem to indicate that "donor" rats' blood values reflect levels that are seen under standard experimental conditions. Therefore, a maximum of 8ml of blood from donor rats can be used for infusion into recipient rats to study the physiological adaptation mechanisms of volume expansion with whole blood.

In space flights, a redistribution of body fluid toward the head and chest as a result of a decrease in hydrostatic pressure in the vasculature of the lower limbs (reference 30) has been observed. As blood from the lower extremities shifts to the cardiopulmonary space due to loss of gravity, there is an engorgement of the central circulation where the mechanoreceptors involved in plasma volume regulation are located (reference 30). In essence, the body perceives a volume expansion. The short term cardiovascular response mechanisms include an increased venous return, cardiac output, arterial pressure and individual organ blood flow (reference 31,32). The resultant overperfusion of individual organs may bring about long-term autoregulatory changes in arteriolar and venular diameter, number and length (reference 33). The long-term regulatory mechanisms also involve the volume regulating hormones (renin, vasopressin, aldosterone and atrial natriuretic factor). The short-term effects involve the autonomic nervous system and the plasma electrolytes, such as, sodium, potassium and chloride. Alterations are also seen in urine and/or blood levels of kinins, corticosterone and catecholamines (reference 34).

Numerous studies of the effects of microgravity or the weightlessness of space flight have demonstrated that the body's homeostatic mechanisms adapt reasonably soon (days) after exposure to it. Although readaptation also occurs upon return to normal gravity, the process is slow (weeks) and cardiovascular deconditioning (i.e. tachycardia, impaired locomotion, reduced exercise tolerance and orthostatic intolerance) has caused problems for astronauts returning from space ( reference 32).

These same changes have been seen in simulations of weightlessness (i.e., bedrest, head-down tilt or immersion) (reference 35). In human and animal studies alike, consistent simulation conditions are difficult to maintain for long periods. Thus, the current data is incomplete, inconsistent and conflicting. Furthermore, the findings during space flight are often difficult to interpret because of the lack of data from the start of launch, reduced possibility to obtain blood samples every day and the lack of simultaneous collection of hemodynamic, hormonal and electrolyte data. It is also difficult to compare the fluid shift data between missions because of the difference between the duration of the flights, nature of the missions and the exercise regime, eating habits and genetics of the individual astronauts (and cosmonauts). Therefore, improved human and animal models need to be developed for simulated weightlessness to better standardize and control the experimental conditions so that the cardiovascular and hormonal adaptations can be fully characterized.

We propose that rats infused with 20-25% excess blood from donor rats will mimic the increased cardiopulmonary volume seen in weightlessness and thus constitute a model for the study of fluid shifts and cardiovascular changes. We speculate that the hypervolemic/volume expansion model will not have the previously described deficiencies. Use of an animal model will allow blood collection and cardiovascular monitoring that is not feasible on human subjects. Direct comparisons can be made between these animal studies and those conducted during space flight. Furthermore, these ground support studies will provide valuable data to help differentiate and distinguish the fluid balance and control adaptation mechanisms from the bone and muscle changes observed in other animal models of weightlessness and during space flight.

In future studies, we will infuse compatible and physiologically comparable donor blood, as described and characterized above, into recipient rats, to simulate the fluid shifts of weightlessness. These experiments will describe the fluid balance factors (cardiovascular, renal and hormonal) operative in simulated weightlessness. In this manner, we will obtain a complete picture of the fluid shifts during simulated weightlessness and the attendant compensatory adjustments. These experiments may also provide data that can be used to develop protective measures for the debilitating effects of space flight and determine whether weightlessness will accelerate or ameliorate some cardiovascular diseases.

## ACKNOWLEDGEMENTS

This work was supported by a National Aeronautics and Space Administration (NASA) Graduate Student Training Grant (NGT-70081); NASA contract NAS2-10523 and National Institutes of Health (NIH) Grant NS22492.The technical assistance of Jane M. Fulton is greatly appreciated. The catecholamines were assayed in the laboratory of Dr. Natalie Alexander at the University of Southern California; Department of Medicine; Los Angeles, CA.

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N91-22176

# THE THEORY OF AN AUTO-RESONANT FIELD EMISSION CATHODE RELATIVISTIC ELECTRON ACCELERATOR FOR HIGH EFFICIENCY MICROWAVE TO DIRECT CURRENT POWER CONVERSION

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A novel method of microwave power conversion to direct current is discussed that relies on a modification of well known resonant linear relativistic electron accelerator techniques. In particular, an analysis will be presented that shows how, by establishing a "slow" electromagnetic field (in particular, a traveling TE wave) in a waveguide, electrons liberated from an array of field emission cathodes (FEC's), are resonantly (with respect to the phase of the electromagnetic wave) accelerated to several times their rest energy, thus establishing an electric current over a large potential difference. Such an approach is not limited to the relatively low frequencies that characterize the operation of rectennas (i.e., 2.45 GHz) and can, with appropriate waveguide and slow wave structure design, be employed in the 300-600 GHz range where much smaller transmitting and receiving antennas are needed.

## 1. INTRODUCTION

The advent of future space and planetary missions brings to the fore the usual energy transfer problems, e.g., launching payloads form earth or from other planetary surfaces, LEO to GEO orbital transfer and, especially with such proposed missions as the Mars mission, the transfer of energy, in particular, in the form of electrical power, to the surface of planets to support the various organized exploration activities (for example, powering the "Mars Rover"). The need to distribute energy from a minimal number of centralized sources (e.g., from a nuclear reactor to an ion engine-driven LEO to GEO space vehicle, or from an orbiting reactor to several widely separated planetary exploration sites and/or an electrically driven rover) in an efficient manner suggests the reconsideration of microwave power transmission and its related conversion from and to useful electrical power. In the past, microwave power transmission concepts were fraught with one major problem, the source of which is directly attributable to the particular microwave to DC conversion device used, i.e., the rectenna. The problem encountered is that unrealistically large transmitting and receiving antennas need to be used for efficient transfer of the microwave radiation; the source of this obstacle is that the rectennas employed (and the only efficient ones still available) work in the 2.45 GHz (i.e., ~ 10 cm wavelength) band.

The purpose of this paper is to demonstrate the theoretical feasibility of the basis of a novel microwave power conversion scheme that is not bound to such a low frequency of operation. Here, the resonant relativistic acceleration of electrons, obtained from an FEC array in a waveguide by action of the electric field of the EM wave field, accelerated by the electric and magnetic fields of the same EM wave field allows one to obtain potential differences on the order of several million volts. Of course, the true test of any conversion scheme is the overall conversion efficiency that prevails. However, this will by highly dependent on the design of the waveguide and the field emission cathode within it. This aspect of the conversion scheme will not be addressed here but incentive will certainly be provided for such a study by the results given here.

## 2. DESCRIPTION OF THE OVERALL POWER CONVERSION SCHEME

The overall power conversion scheme envisioned here is as follows: a waveguide is fitted with a slow wave structure and a field emission cathode. The FEC can in the form of a helix subtending an angular range of $2\pi$ radians with a pitch that is commensurate with that of the traveling, circularly polarized wave the electric field of which induces the electron field emission. In the presence of an appropriately specified, externally applied magnetic field (it will be shown in Section 3 why this external field is necessary) the electrons are then accelerated in the transverse plane of the waveguide by the electric field of the wave while, at the same time, the transverse momentum is converted into longitudinal momentum by the attendant magnetic field of the wave. Resonance (or actually, anti-resonance) of the accelerating electrons with the oscillating electric field of the wave is constantly maintained by the external magnetic field. A point is reached where all of the transverse motion of the electrons is converted into longitudinal motion; at this point the electrons are collected, thus establishing an electric current across an induced potential difference in a circuit joining the collector and waveguide wall. The concept is depicted in Figure 1.

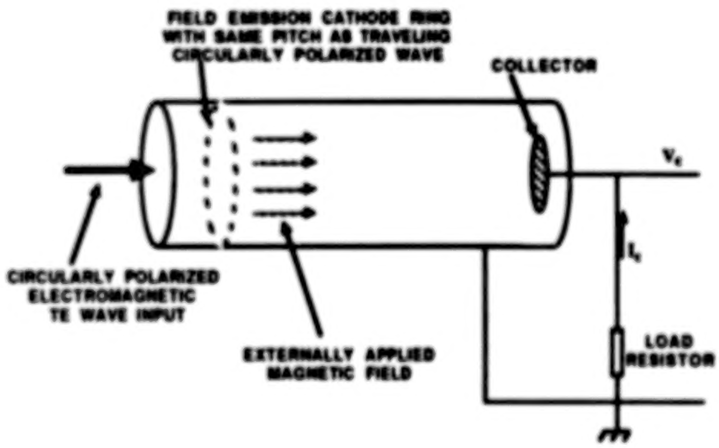

**FIGURE 1.**
**SLOW WAVE FIELD-EMISSION ELECTRON ACCELERATOR FOR HIGH EFFICIENCY MICROWAVE TO DIRECT-CURRENT POWER CONVERSION**

In what is to follow in Section 3 below, the theoretical basis of the foundation of the proposed conversion method will be developed under rather palatable assumptions. Even though the acceleration of the field emission electrons relies on a properly designed slow wave structure and waveguide, these latter elements will not be addressed here since they already represent a well developed area of methodology. However, such considerations are necessary for a complete assessment of conversion efficiencies; only with a complete accounting of the various losses in the slow wave / waveguide system can one calculate the available electric field intensities at the FEC's, the beam loading power, etc. Thus, the analysis below will provide a calculation of the potential difference $V_C$ realized by this method of power conversion. The attendant beam current $I_C$, easily found from the well known Fowler-Nordheim equation, and the and the related converted power $P_C = V_C I_C$ will be functions of the particular slow wave structure and waveguide design and will be considered in a future study. For the same reasons, analyses concerning the maximum transverse and longitudinal dimensions will not be included in the discussion below, although it is not too difficult to obtain these results.

## 3. THEORETICAL DEVELOPMENT OF THE ACCELERATION TECHNIQUE

Consider a circularly polarized TE wave traveling along the axial direction of a circular waveguide in the presence of an externally applied magnetic field that is also directed along the axis. (The fact that this must be a slow TE wave will be established below.) Let there exist at the input to the waveguide an appropriately arranged field emission cathode array from which electrons are continuously liberated by the action of the electric field of the wave. Since the outcome of the acceleration technique is the increase of electron energies to several times their rest energies, a relativistic description of motion is in order. Neglecting 1) the presence of the necessary radial variation of the external magnetic field and 2) electron interactions and the action of "beam currents" on the traveling wave, the equation of motion of an emitted electron in such a situation is

$$\frac{d\mathbf{p}}{dt} = e\left[\left(1 - \frac{\beta_z}{\beta_{ph}}\right)\mathbf{E}\right] + \frac{e}{c\beta_{ph}}(\mathbf{v}\cdot\mathbf{E})\hat{\mathbf{z}} + \frac{e}{c}\mathbf{v}\times\mathbf{B}_e \tag{3.1}$$

where $\mathbf{p}$ is the momentum and e is the elementary charge of the electron, c is the velocity of light, $\mathbf{v}$ is the velocity of the electron, $\beta_z = v_z/c$ for the component of $\mathbf{v}$ in the longitudinal z direction ( $\hat{\mathbf{z}}$ is the unit vector in this direction), $\beta_{ph} = v_{ph}/c$ where $v_{ph}$ is the phase velocity of the wave, $\mathbf{E}$ is the electric field attendant with the TE wave, and $\mathbf{B}_e$ is the externally applied magnetic field the spatial form of which, at this point, remains to be determined. The first term on the right side of Eq.(3.1) describes the transverse force caused by the combined electric field of the wave and the Lorentz force of the magnetic field of the wave, the second term describes the longitudinal Lorentz force due to the magnetic field of the wave, and the third term is the total force that results from the external magnetic field. It is through this "externally induced" force that one can design and control the acceleration process. Also, from considerations of total energy of the electron, one has the relation

$$m_0 c^2 \frac{d\gamma}{dt} = e\mathbf{v}\cdot\mathbf{E} \tag{3.2}$$

where $\gamma = 1/\sqrt{(1-v/c)}$. Adopting plane polar coordinates in the transverse plane of the electron motion, Eqs.(3.1) and (3.2) give

$$m_0\frac{d}{dt}\left(\gamma \mathbf{v}_\perp\right) = e\left(1 - \frac{\beta_z}{\beta_{ph}}\right)\mathbf{E} + \frac{e}{c}B_z\left(\rho\theta'\boldsymbol{\rho} - \rho'\boldsymbol{\theta}\right) \tag{3.3}$$

$$m_0\frac{d}{dt}\left(\gamma z'\right) = \frac{e}{c\beta_{ph}}\mathbf{E}\cdot\mathbf{v} \tag{3.4}$$

$$m_0c^2\frac{d\gamma}{dt} = e\mathbf{E}\cdot\mathbf{v} \tag{3.5}$$

where the primes denote differentiation with respect to time, $m_0$ is the rest mass of the electron and $\mathbf{v}_\perp = \rho'\boldsymbol{\rho} + \rho\theta'\boldsymbol{\theta}$ is the transverse velocity of the electron. The last two equations yield a prevailing constant of motion, I, viz,

$$\gamma\left(1 - \beta_z\beta_{ph}\right) = I \tag{3.6a}$$

This constant can easily be specified by the initial conditions that exist for an electron of initial total energy $\gamma_0$ and initial velocity $\beta_{z0}$ in the longitudinal z direction; hence, one has

$$\gamma\left(1 - \beta_z\beta_{ph}\right) = \gamma_0\left(1 - \beta_{z0}\beta_{ph}\right) \tag{3.6b}$$

At this point, it is convenient to represent the circularly polarized traveling TE wave as well as the transverse velocity as

$$\mathbf{E} = E\left(\cos\theta_w\boldsymbol{\rho} + \sin\theta_w\boldsymbol{\theta}\right),\quad \theta_w = \omega\left(t - \frac{z}{c\beta_{ph}}\right) + \phi_w \tag{3.7}$$

$$\mathbf{v}_\perp = v_\perp\left(\cos\theta_p\boldsymbol{\rho} + \sin\theta_p\boldsymbol{\theta}\right),\quad \theta_p = \omega\left(t - \frac{z}{c\beta_{ph}}\right) + \phi_p \tag{3.8}$$

where $\omega$ is the angular frequency of the electromagnetic radiation, $\phi_w$ is the phase of the wave and $\phi_p$ is that of the particle. Substituting Eqs.(3.7) and (3.8) into Eq.(3.3), expanding the derivative on the left side and simplifying yields a relation that governs the relative phases of the wave and particle motions, i.e.,

$$\frac{d\left(\phi_p - \phi_w\right)}{dt} = \omega\left(1 - \frac{\beta_z}{\beta_{ph}}\right) - \frac{\omega_B}{c\gamma} - \frac{1}{\gamma v_\perp}\left(\frac{d\gamma v_\perp}{dt}\right)\tan\left(\phi_p - \phi_w\right) \tag{3.9}$$

where $\omega_B$ is the cyclotron frequency of the electron in the B-field given by

$$\omega_B = \frac{eB_e}{m_0}$$

The relationship of Eq.(3.9) demonstrates the fact that if the condition

$$\omega\left(1 - \frac{\beta_z}{\beta_{ph}}\right) = \frac{\omega_B}{c\gamma} \tag{3.10}$$

is enforced throughout the motion, and if $\tan(\phi_p - \phi_w) = 0$ initially, the electron and TE wave will remain in resonance (or anti-resonance). If this is the case, then for an electron introduced into the crossed fields within the waveguide by field emission from an appropriately arranged cathode induced by the attendant electric field, $\phi_p - \phi_w = \pi$, anti-resonance is thus achieved, and as will be shown below, the electron will be accelerated in the longitudinal direction.

Assuming that the condition of Eq.(3.10) prevails throughout the motion of the electron and, hence, the electron remains in anti-resonance with the electric field of the wave, and taking the scalar product of Eq.(3.3) with $\gamma \mathbf{v}_\perp$ and simplifying gives

$$m_0 \frac{d\gamma v_\perp}{dt} = e\left(1 - \frac{\beta_z}{\beta_{ph}}\right) E\cos(\phi_p - \phi_w) = -e\left(1 - \frac{\beta_z}{\beta_{ph}}\right) E \tag{3.11}$$

showing that the electron is accelerated toward the axis of the waveguide so long as $\beta_z < \beta_{ph}$; when the condition $\beta_z > \beta_{ph}$ prevails, the electron reverses its direction of acceleration. But as found from Eq.(3.5),

$$\frac{d\gamma}{dt} = \frac{e}{m_0 c^2} E v_\perp \cos(\phi_p - \phi_w) = -\frac{e}{m_0 c^2} E v_\perp \tag{3.12}$$

Hence, as indicated by Eqs.(3.12) and (3.13), the electron energy will not only increase when $\beta_z < \beta_{ph}$ but continue to increase when $\beta_z > \beta_{ph}$ so long as $v_\perp < 0$, i.e, the electron continues to move toward the axis of the waveguide. With $v_\perp = 0$, the acceleration process stops and the the direction of motion is totally along the axis of the waveguide making electron collection straightforward. This situation can only be realized when one deals with a slow TE wave where $\beta_{ph} < c$.

Expressions for the evolution of the energy and the longitudinal velocity can be obtained from Eqs.(3.3) and (3.5) but are unwieldy and, in this case where electron interactions and the attendant beam currents are neglected, unnecessary to use. One can easily obtain such relationships from the integral of motion, Eq.(3.6b), and the well known relativistic identity

$$\gamma^2\left(1-\beta_\perp^2-\beta_z^2\right)=1 \tag{3.13}$$

Solving these equations for $\beta_z$ yields

$$\beta_z=\frac{\beta_{ph}\pm\gamma_0\left(1-\beta_{z0}\beta_{ph}\right)\sqrt{\left[\gamma_0^2\left(1-\beta_{z0}\beta_{ph}\right)^2+\beta_{ph}^2\right]\left(1-\beta_\perp^2\right)-1}}{\gamma_0^2\left(1-\beta_{z0}\beta_{ph}\right)^2+\beta_{ph}^2} \tag{3.14}$$

The choice of which root to use in this solution is dictated by the initial conditions; at the point of field emission of the electron into the crossed fields, $\beta_\perp << 1$ and $\beta_{z0} \approx 0$. Therefore, one has $\gamma_0 \approx 1$ and Eq.(3.14) becomes $\beta_z = (\beta_{ph} \pm \beta_{ph})/(1+\beta_{ph}^2)$ thus indicating that the "−" sign be used for values of $\beta_\perp$ up to the point where the radical vanishes. This consideration also indicates that if the end of the acceleration process is defined where $\beta_\perp = 0$, the maximum longitudinal velocity attained by the electron is

$$\beta_{z,\,max}=\frac{2\beta_{ph}}{1+\beta_{ph}^2} \tag{3.15}$$

The vanishing of the expression under the radical in Eq.(3.14) defines the maximum value of the transverse velocity $\beta_{\perp max}$ that is attained by the electron; equating this expression to zero and solving for $\beta_\perp$ yields

$$\beta_{\perp max}=\sqrt{\frac{\gamma_0^2\left(1-\beta_{z0}\beta_{ph}\right)^2+\beta_{ph}^2-1}{\gamma_0^2\left(1-\beta_{z0}\beta_{ph}\right)^2+\beta_{ph}^2}} \tag{3.16}$$

After this point where the two roots coincide, the root defined by the "+" sign prevails. The transverse velocity will begin to decline from $\beta_{\perp max}$ but the longitudinal velocity will continue to increase. The value of the declining $\beta_\perp$ at which $\beta_z = \beta_{ph}$ is found by applying this constraint to Eq.(3.14) and solving for $\beta_\perp$ which gives

$$\beta_\perp=\sqrt{\frac{\left(I^2+\beta_{ph}^2-1\right)\left[I^2-\beta_{ph}^2\left(I^2+\beta_{ph}^2-1\right)\right]}{I^2\left(I^2+\beta_{ph}^2\right)}} \tag{3.16}$$

where $I=\gamma_0(1-\beta_{z0}\beta_{ph})$. The fact that the event $\beta_z=\beta_{ph}$ occurs after $\beta_{\perp max}$ is realized is also borne out in the equations of motion, viz, Eqs.(3.11) and (3.12). Expanding the differentiation on the left side of Eq.(3.11) and using Eq.(3.12) yields

$$\gamma\frac{d\beta_\perp}{dt}=-\frac{eE}{m_0c}\left[1-\frac{\beta_z}{\beta_{ph}}-\beta_\perp^2\right]$$

Since $d\beta_\perp/dt=0$ at $\beta_\perp=\beta_{\perp max}$, one has

$$\beta_z=\beta_{ph}\left(\frac{I^2+\beta_{ph}^2-1}{I^2+\beta_{ph}^2}\right)<\beta_{ph}$$

This provides the key behavior of the electron in the acceleration process: The transverse velocity of the electron increases due to the action of the electric field of the traveling wave. Of course, some of the attendant transverse momentum gained by the electron is converted to longitudinal momentum by the action of the magnetic field of the wave but that in the transverse direction predominates over that in the longitudinal direction. At the transverse velocity $\beta_{\perp max}$, the situation reverses where the longitudinal momentum increases faster than that of the transverse. The longitudinal velocity continues to increase at the expense of the transverse until $\beta_\perp=0$.

Using Eq.(3.15) with Eq.(3.6b) yields an expression for the maximum energy gained by the electron, i.e.,

$$\gamma_{max}=\frac{1+\beta_{ph}^2}{1-\beta_{ph}^2} \tag{3.17}$$

Finally, it must be remembered that the entire analysis presented above presupposed the fact that the resonance condition of Eq.(3.10) prevails throughout the acceleration process. With the values of $\beta_z$ and $\gamma$ evolving during acceleration, it is apparent that the externally applied magnetic field must also change accordingly. Eliminating the factor $\gamma$ from Eq.(3.10) using Eq.(3.6) and solving for $B_e$ yields

$$B_e = \frac{m_0 c\omega}{e}\left(\frac{l}{1-\beta_z\beta_{ph}}\right)\left(\frac{\beta_{ph}-\beta_z}{\beta_{ph}}\right) \qquad (3.18)$$

in the general case and in the case of field emission considered above, I=1. In this specific case with $\beta_z = 0$ initially, $B_e = B_{e_0} = m_0 c\omega/e$. When $\beta_z = \beta_{ph}$, $B_e = 0$. Finally, at the other extreme where $\beta_z = \beta_{z,\,max}$ as given by Eq.(3.15), one has $B_e = -B_{e_0}$. Thus, the external magnetic field must initially have an intensity of $B_{e_0}$ and monotonically decrease to zero as the point where $\beta_z = \beta_{ph}$ is reached, and then change direction and monotonically increase to a value of $-B_{e_0}$.

As an example of the parameters and specifications encountered in the acceleration of field emission electrons within a waveguide by a traveling TE wave, consider the case where $\beta_{ph}$=0.9 at an operating frequency of 100 GHz ($\omega = 6.28 \times 10^{10}$ rad./sec). From Eqs.(3.15) and (3.17) one has $\beta_{z,\,max} = 0.9945$ and $\gamma_{max} = 9.53$. Thus, within the constraints of the theory presented above, an electron can be accelerated to 9.53 times its rest energy or 4.86 Mev. The maximum value of the externally applied longitudinal magnetic field needed to achieve the anti-resonance of the traveling electric field and accelerating electron os found to be $B_{e_0} = 100.7$ kG.

Electrons accelerated in such a manner will induce a potential difference $V_C$ between the FEC array and the collector. Since $V_C$ must be such that the quantity $eV_C$ is the total kinetic energy of the electrons, i.e.,

$$V_C = \frac{m_0 c^2\left(\gamma_{max}-1\right)}{e} \qquad (3.19)$$

one has from this relation and Eq.(3.17)

$$V_C = m_0 c^2\left(\frac{2\beta_{ph}^2}{1-\beta_{ph}^2}\right) \qquad (3.20)$$

From the values calculated above, one finds that $V_C = 4.35 \times 10^6$ volts can be realized.

Thus, the possibility of accelerating field emission electrons in a waveguide by the same electromagnetic field from which field emission was attained has been demonstrated. The key to the operation of such an acceleration scenario is the fact that momentum gained by the electron from the transverse electric field of the wave is converted into longitudinal momentum by the action of the attendant magnetic field of the wave. The process continues until all longitudinal momentum has been depleted. The ability for the accelerating electron to remain in synchronization with the traveling wave during this process is provided by an externally applied magnetic field that satisfies a specific intensity profile along the axis of the waveguide.

In the analysis above, consideration was not given to the transverse and longitudinal dimensions over which the acceleration process must take place. This requires a much more detailed exposition than that given here but will certainly be addressed in future publications on this subject. The purpose here was just to demonstrate the theoretical possibility of such a technique for power conversion purposes.

## 4. CONCLUSIONS NAD SUGGESTIONS FOR FURTHER WORK

It has been shown that under the rather non-restrictive assumptions that one neglect the ever-present transverse variation of the constant external B-field (at least in the paraxial region) and that the motion of the electrons are independent of one another and do not act back on the traveling wave, one can obtain linear acceleration of field emitted electrons to several times their rest energies. There is, however, another element of the process that has not been addressed; that is the stability of the accelerated electron motion with respect to the variations of the dynamical parameters of the field emission electrons. In the above treatment, it was only assumed that $\beta_{\perp 0} << 1$ and $\beta_{z0} \approx 0$. A rigorous analysis would require the full solution of Eqs.(3.3)-(3.5) (or Eqs.(3.6), (3.11), and (3.12) in light of the resonance condition Eq.(3.10)) with perturbations introduced into the initial conditions. One can then study the behavior of the electrons as they make the transition to stable motion. This, as well as the incorporation of the possibility of electron interactions into the theory, should be the next level of approximation that is considered so a thorough comparison can be made with the foregoing.

N91-22177

# NATURAL VACUUM ELECTRONICS

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The ambient natural vacuum of space is proposed as a basis for electron valves. Each valve is an electron-controlling structure similar to a vacuum tube that is operated without a vacuum sustaining envelope.

The natural vacuum electron valves discussed offer a viable substitute for solid state devices. The natural vacuum valve is highly resistant to ionizing radiation, system generated electromagnetic pulse, current transients, and direct exposure to space conditions.

## INTRODUCTION

This paper proposes the use of the natural vacuum of space as a basis for electronic components. The space environment offers a natural vacuum that is suitable for the passage of electron beams over rather long distances. These electron beams can be used in electron valve components.

## PHYSICAL OPERATION

The basic natural vacuum component uses a stream of electrons that is emitted from a source. This electron stream travels through the ambient vacuum environment towards a positively charged destination. This basic flow corresponds to the flow of electrons within the diode vacuum tube of electronics history. In the diode vacuum tube, electrons are emitted by a heated cathode (thermionic emission). The emitted electrons are attracted across the vacuum by a positively charged metallic plate (the anode).

Natural vacuum components use this basic vacuum tube flow without the need for the vacuum sustaining enclosure. The cathode and anode are directly exposed to the ambient environment of space. The natural vacuum diode "tube" provides the service of rectification where an alternating current input is converted to pulses of direct current output.

Additional functions are provided by installing electron flow modifying structures in the electron beam. A metallic screen or grid placed in the electron beam between the cathode and anode provides a valving action. Varying the electric charge on the grid varies the electron beam current traveling between the cathode and the anode. Additional grids can be inserted into the electron flow to provide additional control and to isolate against undesired coupling of stray signals through the system.

This type of valving is quite familiar to people in electronics who have worked with multi-grid vacuum tubes. In the multigrid natural vacuum component, the artificial vacuum maintained by the glass tube is replaced by the ambient environment of space. Figure 1 shows a natural vacuum component with concentric grids.

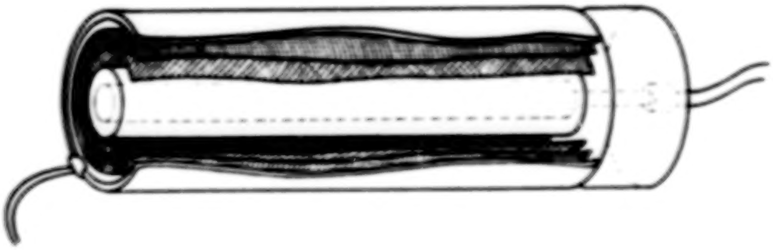

Figure 1. Natural vacuum component.

Magnetic fields can be used to control the electron beam also. Magnetic fields can direct, deflect, reverse, or contain the electron beam. A limited example of this control is the common television picture tube. The ambient environment of space frees the natural vacuum component from the heavy picture tube containment. A natural vacuum component can hang the electron gun and magnetic coils in an open structure.

## COMPONENT INNOVATIONS

Natural vacuum components are not limited to revivals of older vacuum tube designs. The large volume of the ambient space environment allows the component designer to produce components that are not practical in small artificial vacuum tube environments. For example, the grids within a natural vacuum component can be mechanically deployable. Different sets of grids can be deployed for different operating conditions.

The cathode can be heated by directly focused solar energy. This configuration avoids the problem of filament failure that was a common problem with terrestrial vacuum tubes.

The most profitable approach to natural vacuum electronics is to think of the ambient environment of space as a gymnasium for electrons. You can place electron sources, destinations (anodes), and beam modifying devices where you need them within the ambient environment. Your design does not have to be limited to connected electron-valving modules. A continuous structure can perform operations on a continuous electron beam that passes through the structure. Connections between the stages of processing are provided by the electron beam itself (although a "ground" return conductor is required as well).

## NON-ELECTRON CARRIERS

The natural vacuum approach can use carriers other than electrons. For example, accelerator technology can be used to generate streams of protons to carry signals or power. While protons may not have clearly-visible advantages over electrons, it is worthwhile to examine the concept of proton-based circuits.

One possible advantage of proton-based circuits is that protons are heavier than electrons. Thus a beam of protons would be less perturbed by ambient magnetic fields—a possible advantage in some space environments.

Atomic nuclei can also be used in natural vacuum systems. Similarly, larger masses of material can be pushed about by mass driver and magnetic separator types of "circuit" components. The scope of electronics is expanded to a continuum of natural vacuum functions that includes large scale energy beam manipulation and materials separation and processing.

## USES

The basic natural vacuum components provide the familiar active electronics functions of:

- switching
- rectification
- oscillation
- amplification
- detection
- mixing

These are the same functions provided by transistors or vacuum tubes. The natural vacuum component is a very robust replacement for conventional solid-state components. Entire electronic circuits can be built from natural vacuum components or natural vacuum components can be used in conjunction with conventional components.

## ADVANTAGES

Natural vacuum components are very robust devices that can survive the stresses of the space environment. Each natural vacuum component is a largely open-metal structure that is directly exposed to the vacuum environment. This structure is resistant to damage from ionizing radiation, system generated electromagnetic pulse (SGEMP), and micro-particle impacts.

The natural vacuum components can be installed outside of the spacecraft conserving interior space for other uses. (Fig. 2) These components can be operated at very high power levels when suitable radiative cooling means are provided.

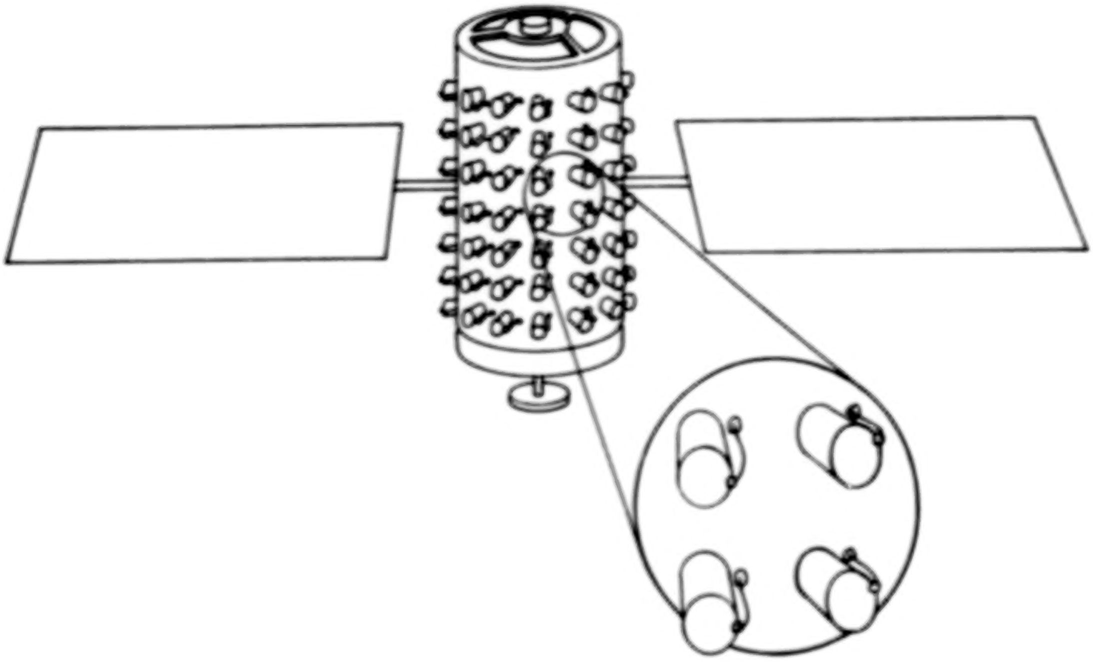

Figure 2. Configuration of externally mounted components.

The natural vacuum components can be made from space-derived metals and insulating materials, when these materials become available. This increases the appeal of the natural vacuum approach for worlds such as the moon that have excellent ambient vacuum environments and locally available materials.

## DISADVANTAGES

The operation of natural vacuum components could be negatively impacted by the poor quality of "vacuum" present in low earth orbit or near other bodies with atmospheres. One such impact is the blockage of the free flow of electrons. Erosion of the components due to the action of ambient molecules could occur as well.

Another potential problem is the size and weight of natural vacuum components that may be larger than their solid state equivalents.

## CONCLUSION

Natural vacuum components should be considered for space craft and lunar base designs. These components change the vacuum of space from a problem into an asset.

# A HYDROPONIC DESIGN FOR MICROGRAVITY AND GRAVITY INSTALLATIONS

Judith Fielder and Nickolaus Leggett
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A hydroponic system is presented that is designed for use in microgravity or gravity environments. The system utilizes a sponge-like growing medium installed in tubular modules. The modules contain the plant roots and manage the flow of nutrient solution. The physical design and materials considerations are discussed as are modifications of the basic design for use in microgravity or gravity environments. The major external environmental requirements are also presented.

## INTRODUCTION

This paper presents a design for a hydroponic system capable of supporting plant growth from seed through maturity. This system can operate in either microgravity or lunar/planetary base environments with only minor modifications required to transition between these environments.

The hydroponic system provides a dark, moist, nutrient-rich, well-aerated environment that is suited for the growth of plant roots. A supporting medium holds the seeds and growing plants which have access to a lighted external environment. In addition, containment is provided to keep solution flows under control in a microgravity environment.

## PHYSICAL DESIGN

The basic hydroponic design consists of tubular structures of sufficient size to accommodate the types of plants being grown. Each tubular structure is a cylindrical module with an open slit extending for the majority of the length of the cylinder. This open slit accommodates a sandwich of a foam-like, spongy substance that holds seeds or growing plants. The slit is oriented towards the lighting system.

The sandwich consists of two strips of spongy material that are inserted into the open slit to fill the entire length of the slit in the cylinder. Crop seeds or plants are placed between the two layers. The portion of the strips exposed to the light is coated with an impermeable, opaque coating.

The tubular structures are installed in arrays that make optimum use of the lighting system and the available pressurized volume. Each tube is connected to a nutrient solution feed line and a nutrient solution removal line. Both the feed and removal lines have valves and bypass lines so that individual modules can be taken out of service without disrupting the operation of the entire growing system.

## NUTRIENT FLOWS

In the microgravity configuration, the nutrient solution feed system is a flattened pressurized feed line inserted near the surface of each strip of sponge. (Fig. 1) A pressurized flow of aerated nutrient solution is introduced into the sponge strips. The flow is directed away from the exterior towards the "bottom" of the cylinder. This flow is maintained and reinforced by a longitudinal air flow established within the cylinder and by the removal of excess and spent nutrient solution by a suction system. (Fig. 2) In a microgravity environment this pressurized flow of nutrient solution may aid in orienting root growth towards the interior of the cylinder. What roots do grow into the exterior will likely be air-pruned into directionality.

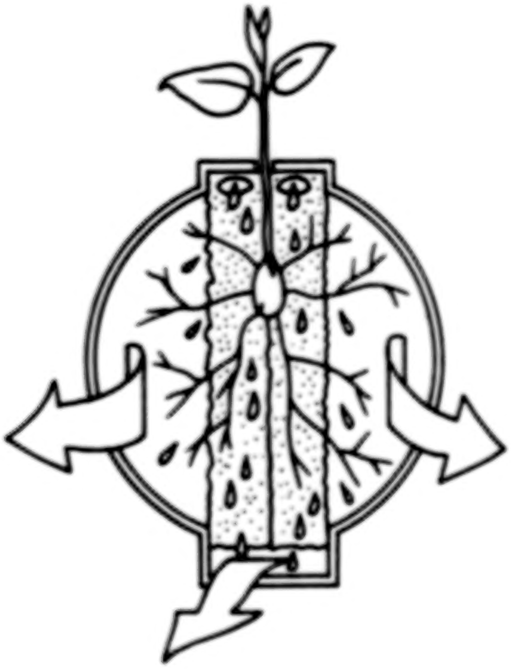

Figure 1. Microgravity system configuration.

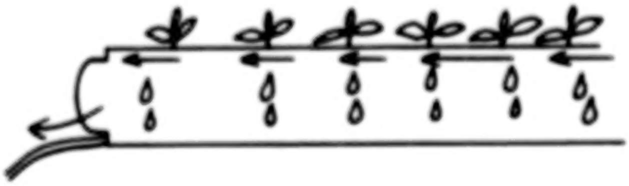

Figure 2. Nutrient flow in microgravity.

In the gravity environment, the nutrient solution is introduced into the cylinder in an ebb and flow schedule. The cylinder is allowed to fill with nutrient solution and then the nutrient solution is drained away. (Fig. 3). This ebb and

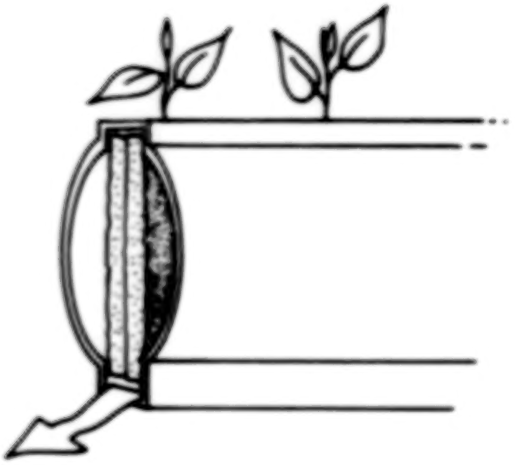

Figure 3. Nutrient flow in gravity.

flow cycling ensures that the roots are well aerated in addition to being supplied with nutrient solution. The ebb and flow cycles are more frequent during periods when the plants are lighted and transpiration is at a peak than during the dark periods.

## NUTRIENT MANAGEMENT AND CONTROLS

The nutrient solutions used in this system must contain all macro nutrients and micronutrients required for plant germination, growth, and maturation. The solutions will need to be formulated for individual crop types to ensure optimum growth rates and yields.

Hydroponic systems tend to be unforgiving of failures in nutrient flows. This system partially mitigates this problem with the limited nutrient solution holding capacity of the sponge. Redundant reservoirs of pre-mixed nutrient solution will be required to replace the existing nutrient solution system reservoir should it become contaminated or lost.

Periodic sterilization of the circulating nutrient solution is recommended to control pathogens. In-line filters will limit the movement of plant materials and some types of pathogens through the system. Plant toxins, pH, and soluble solids must be monitored on a continuous basis.

The circulating nutrient solution must be refreshed and periodically replaced in entirety. Provision should be made for the reconstituting of the spent nutrient solution and for the recycling of organic wastes. These major requirements are not easily achieved in a hydroponic system.

## EXTERNAL ENVIRONMENT

The environment outside of the cylinder should replicate many of the features of the earth's environment. The plants should be fully and evenly lit with light provided in a regulated day/night cycle. Broad spectrum lighting in the 400 to 800 nanometer range is required with infrared emissions kept to a minimum. The light intensity should be 200 to 400 watts per square meter for optimum plant response. This lighting system will generate waste heat that must be removed from the agricultural area.

Temperature in the agricultural areas should be maintained in the range of 15°C to 32°C. A standard nitrogen, oxygen, and carbon dioxide atmosphere is required. A pressure of 800 mb or above is strongly recommended. This atmosphere is intermittently moved by the plants at a speed of 1–6 km/hour to circulate fresh air, prevent $CO_2$ depletion, facilitate temperature control, and provide some thigmonastical stimulation.

The plants may need other environmental features such as the presence of a steady state magnetic field. The exploration of these features should be an important mission for space station and lunar base plant growth experiments.

## MATERIALS ENGINEERING CONSIDERATIONS

All materials used in this system must be non-toxic to both plants and crew members. This includes the cylinders, sponge strips, feed lines, valves, circulating pumps, and other components that come in contact with the plants or the nutrient solution. Coated toxic materials are not appropriate due to the possibility of coating failure either by accidental impact or abrasion.

The foam-like sponge strips must be engineered for appropriate water retention and aeration properties required by their gravity environment. It would be advantageous to create the sponge out of a recyclable material that does not have to be imported from the earth.

All of the materials in contact with the plants and nutrient solutions must be highly resistant to corrosion. There should be little or no interaction between the materials, nutrient solutions, plant-released organic acids, and other compounds. Aluminum components in particular should be avoided because of their corrosion susceptibility and toxicity to plants.

N91-22178

# POLAR LUNAR POWER RING - PROPULSION ENERGY RESOURCE

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ABSTRACT - A ring shaped grid of photovolaic solar collectors encircling a Lunar pole at 80 to 85 degrees lattitude, is proposed as the primary research, development and construction goal for an initial lunar base or development in general. The Polar Lunar Power Ring (LPR) is designed to provide continuous electrical power in ever increasing amounts as collectors are added to the ring grid. Initial lunar base function is suggested as production and distribution of power system components, most importantly the photovoltaic cells. Upon completion of phase 1, the LPR can provide electricity for any purpose indefinitely, barring a meteor strike. Capable of acting as the foundation for diverse Lunar developments ranging from agricultural life support systems to plasma based refinement and fabrication techniques(seamless CELSS), the associated rail infrastructure and inherently expendable power levels place LPR as an ideal tool to power an innovative propulsion research facility, and perhaps a trans Jovian Fleet. The basic polar power ring has utility on asteroids, and could suggest Earthly rectenna receiver positioning as well. Proposed initial output range 90 Mw to 90 Gw.

## INTRODUCTION

The moon can figure prominently in propulsion research in terms of energy and facilities for R and D testing, however, in order to achieve this goal as well as overcome the solar energy problem of the long Lunar Night, a rapid single minded development from an initial lunar base is required. The assumption is that in order to be able to pursue propulsion Research and development, all other potential energy users would likely need to be satisfied in order to reduce competition for energy. Although it is common knowledge that human demand is insatiable, this situation is limited to Lunar access, so we assume a limited set of users of energy including mining and benefaction, power set component manufacture, and perhaps direct energy export from moon to earth, as well as many other worthy energy uses. The following concept is an attempt to design a power system that is capable of generating sufficient continuous electrical energy to satisfy all users in a profitable manner so energy consumption trade-offs do not generate a negative impact on advances in propulsion required for deep space exploration. The power of the polar Lunar power ring could also be used to facilitate current propulsion requirements. Some problems are identified, unfortunately the initial lunar base would be dominated by the need to produce photovoltaics, the power ring could be very difficult to turn off even in the event we wished it would cease output, also the adventure of independent Lunar development would be limited to the ring builders, as later developments would probably tie into the established electrical grid upon landing. An adventurers point of view. Upon completion of the 90 Gw upgrade the ring would present a solid 300 foot high face of glittering amorphous silicon as the sun comes around. The ring would stretch off into the horizon, like a grey ribbon on edge. From space an almost invisible silver thread or Tiara, yet ring shade would be clearly visible from a polar orbit.

## Philosophical Introduction

In the course of designing strategies for the exploration of the Galaxy number of philosophical drivers were identified and developed as possible fundamental principles or skills for Space Exploration. One key strategic area is the initial phase of gaining access to space from a fully colonized hospitable planet to our moon(our current situation). If one considers as a fundamental principle of Space exploration the learning of skills in the application of light to pursue space related objectives as transportation, life support, and construction, one can abtract the process and frame it in the context of Life and reality as it now exists. To digress, at Stars, light is a certain resource, the usefulness of matter as it may be found will depend on our light skills. If then one considers that Terran life as it now exists depends on the light skill of photosynthesis and its associated 4 billion years of developmental biochemistry, and that the conditions and residents of the biosphere are almost entirely dependent on photosynthesis, the sentient species Humans should realize that to space is to learn new light skills. If one considers photosynthesis as the key light skill in the hands of life and its final success in building the biosphere of Earth, what extensions to the biosphere are possible, now that sentience( Humans) have appeared within the biosphere, how will photosynthesis be applied, and what are the next logical light skills that may be employed in the hands of life? In simpler terms can sentience apply photovoltaics to support photosynthesis, and what new light skills can be developed that can augment the life/light relationship? For example optimizing photon pressure devices. In addition the photoelectric effect, compton effect, reverse FEL, and inverse Cerenkov offer challenges. Perhaps even testing laser sails as resonators will produce some surprising results. Conceptually- In space, images of New habitats, distance, journey, inertia, momentum, travel, with light as the common thread, and light in transportation emerge, with a question regarding photon pressure, and the potential for optimization, with links to reaction mass. Since most new habitat for life lies in orbit around other stars the application of light in regards to access to those stars suggests a driver for research and development. There clearly is a challenge to sentience to make itself useful. A possible retribution for the extinctions one species is inflicting on the community of the biospheres. A valid question- Are we capable of sustaining life? We go to the moon to face the Galaxy. The Polar Lunar Power Ring will keep the lights on, but it will likely fall on Space Exploration Engineers to prove someone is home.

A corollary is that if we create such vast and powerful devices, we must prevent their misuse through the application of new goals as a species. These goals, with education, can do as much to turn around the human conditions as any new power system. Ingenuity can give rise to new hope. A tremendous future lies ahead for this solar system, provided we apply new resources for the long term. We can conquer the moon by a simple circle of Photovoltaics, but this can only spell disaster unless the victory is used to lay the foundation for address greater challenges. I strongly urge discussion on these matters.

### Electricity During The Lunar Night

Electricity stands out as a key tool in the tool chest of a space able civilization both on the surface of the planet and beyond. Photovoltaic cells for the conversion of light energy into electrical energy allows us to generate electricity in space from the energy

resource of light from the star. Nevertheless, as usefull as electricity is, we are dependent on the *foodstuffs* resulting *from* photosynthesis. Plant growth is dependent on continual or diurnal irradiation. The following power system was designed with a mind to providing photosynthetic light to an environment that is subject to a 354 hour night (The Lunar Night), utilizing photovoltaic cells, as simply as possible. The purpose of growing plants was to make a lunar base self sufficient in life support systems.
Once life has established a foothold on the moon using photovoltaics to supply adequate photosynthetic light to support a population of Humans, a portion of those humans can consider the next logical light skills to be used in expanding the domain of life keeping in mind the best habitats represent the greatest challenges - habitable planets at other stars.
The authors personal R&D fascinations tend towards photon pressure devices, light sails, laser propulsion, laser resonator propulsion, Reverse Free Electron lasers, Reverse Cerenkov effects, The Compton effect, free space photovoltaics, the photoelectric effect, and ultimately the Light drive. The Polar Lunar Power Ring however has many other usefull and interesting applications.

## Origins of the Polar Lunar Power Ring

Many years ago I did a calculation to find the height of a tower, the top of which, when situated on the pole of the moon would remain constantly exposed to sunlight for the purpose of supplying electricity below to run electrolysis equipment to produce LH2/LOX fuel. The tower height was over 11 kilometers, The power tower had the disadvantage of being difficult to expand the electrical output and could be rendered useless by collectors assembled on the horizon. I examined placing Photovoltaic collectors on the horizon of the tower top, especially on those hills about 200 kilometers away and concluded a ring of collectors was better than the tower and easier to construct than a equatorial power ring. The horizon was about 6.5 degrees and is now proposed to be about 10 degrees due to uncertainties regarding the lunar terrain and the Lunar "winter". The power output is phenomenal, expendable and has the advantage of being continual. After reading Ref. 3 by J.D Burke regarding Polar power systems the ring concept was written up as a paper. Renewed interest in the polar regions of the Moon for its potentially cold trapped water, and renewed interest in Large scale Space Power projects as a solution to biosphere destruction , as well as the need to continue the exploration of space has led to this revised version of the polar power ring.

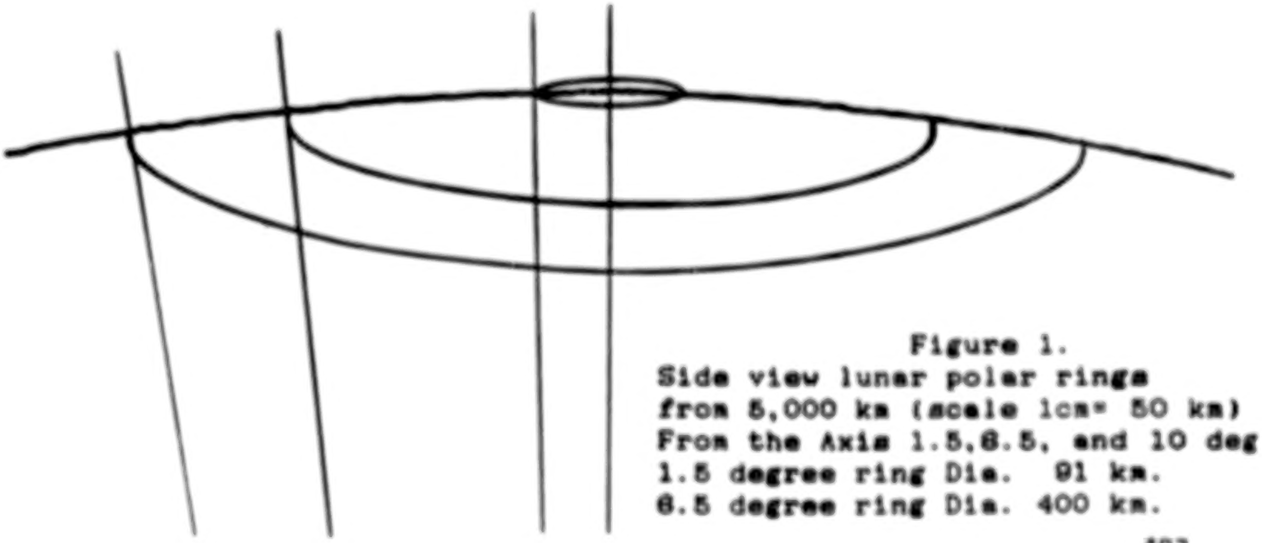

Figure 1.
Side view lunar polar rings
from 5,000 km (scale 1cm= 50 km)
From the Axis 1.5,6.5, and 10 deg
1.5 degree ring Dia. 91 km.
6.5 degree ring Dia. 400 km.

Terraforming analysis

The moon is a slowly rotating body under about 1.4 kilowatt/ square meter solar irradiation, the period of rotation is about 708 hours releative to the sun leaving 354 hours of time during which no solar irradiation is available to a given collector on the surface. Maximal axial tilt to star is about 6.5 degrees, lunar radius is about 1740 km equatorial. The Moon is oblate. if we calculate the distance from the axis at 6.5 degrees from the pole a rough figure of 390 kilometers is found, or a ring almost 800 km in diameter is described around the pole. The pole of the moon is not a smooth surface. Nevertheless from a terraforming theory point of view, a ring beyond 6.5 degrees axial would at any given time including lunar seasons, be under solar irradiation continuously except under conditions of eclipse. This is a situation in common with the equatorial regions and defines the Arctic Circle in Terran terms for survival purposes. Although it could be argued the Moon has 2 types of Arctic circles we seek a permanent location outside either, and this interesting situation will affect the shade cooling potential of the Power system. Having defined the rough minimum circumference of a ring under constant solar irradiation we then compare the boundaries against the topography of the polar surface as valleys outside of the ring allow the boundary to move towards the pole as do high ridges, promentories and crater walls, however these same high promentories outside the ring may force the power collection boundary outside towards the equator as they would eclipse the collectors, In this case the boundary moves to the equatorial face of the topographical feature to ensure adequate irradiation.

Collectors are preferably photovoltaic, amorphous silicon, and produced on the Moon(Landis ref #2)at the base in production facilities, and would be placed in amenable locations alongside pathways cleared by long distance rovers or Sun-Following Lunar bases(Landis Ref #1). Baseline collector size is 10 meters by 10 meters for 14 kw per collector at 10% conversion efficiency. The collectors are fixed for calculation purposes, therefore if we assume optimum collection efficiency to occur along a portion of the ring equal to 1 radius of the ring, at a collector spacing of 100 meters the output of the 400 km of ring would approximate 56 megawatts. This is a conservative figure. Use of foil reflectors and optimizing terrain advantage should bring the figure in at about 90 megawatts. a fully optimized ring to a height of 100 meters solid photovoltaics would put out 90 gigawatts continuous.

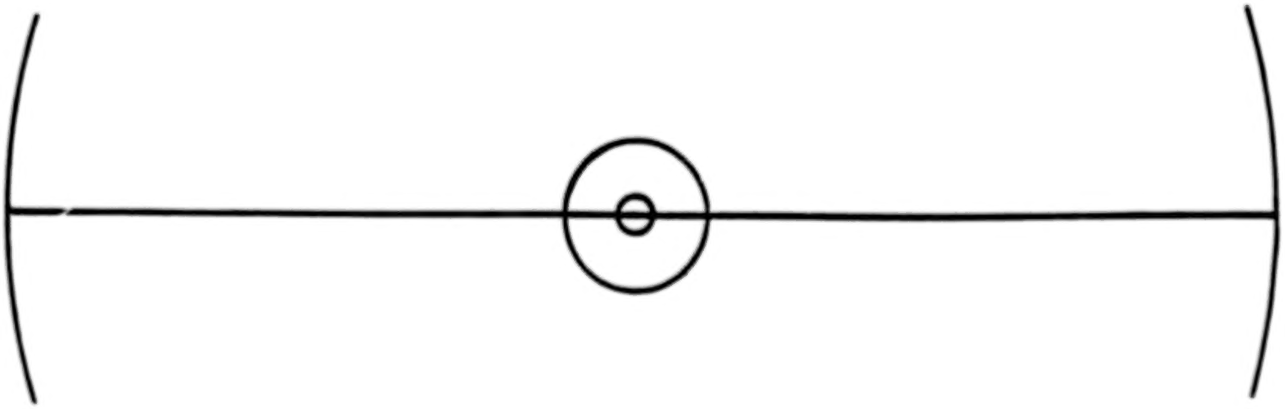

Figure 2. Relative Ring Sizes-Equatorial, 1.5 and 6.5 Degrees
Scale 1 cm = 200 km Line represents 1738 km Lunar diameter
Note - The actual power ring will be irregular due to topographical opportunities- ridges, crater walls, highland and mountains. Exploration challenges beckon. Will Lunar Prospector map out 1:50,000 topographical?

Regarding the initial construction of the Ring,

As the polar data from the Lunar Prospector spacecraft or Muses-A(Japan) becomes available, topographical features of extreme interest should become part of plans for initial bases and landing of development missions. In the case of the LPR 3 parameters come to mind: 1. Polar light mountains or ridges( Mountains of Perpetual Sunlight) are key features. 2. Clean landing areas near the light mountains ( or at a low enough lattitude to avoid lunar winter darkness if Light mountains do not exist) 3. Line of sight telerobotic communications with Earth. From there our search begins.
If one considers the concept of Landis- A Sun-following Moonbase(Ref. 1) and applies it to tracing the path for construction of the Polar Lunar Power Ring, construction of a Ring will inevitably result. It is my experience that explorers, if repeatedly travelling over the same path, will increasingly break trail and improve the trail as best as time and equipment permits, and at some point a smooth trail bed cleared of rocks with numerous resource caches to the side will exist, and eventually equipment caches will result as well. If the excess production of solar cells is made available the cells will be emplaced along the path and inevitably be linked together to supply stopover points at scenic areas, and as power transmission resources become available a ring will take form. I would expect power transmission cable sooner than extra solar cells, nevertheless I for one would enjoy the initial trail building

Polar Lunar Power Ring #1 initial and final Specifications

Scenario #1(no storage of power)- The initial Power system would be a rag-tag affair rigged up on the slopes of the most accessible mountain of perpetual sunlight or Polar Light Mountain(PLM). On the slopes of the PLM, photovoltaic collectors would face every sunward direction and be cable connected to a power conditioning unit to transmit the power back to the base. If the PLM is high enough to see sun during Lunar winter, the base will shift to the PLM from the Landing spot and a new landing spot or Port will be cleared and constructed. A manufacturing plant for the Lunar Production of Solar Cells(Landis and Perino ref. #2) would be assembled and operated using the output of the PLM cells. The next step it to identify other topographical features that are amenable to solar irradiation in excess of 14 days per lunar cycle and clear travel pathways to those features, gain access to the irradiated slopes, emplace newly produced collectors and run transmission cable back to PLM base. At some point these other collector points would be laterally connected and portions of a ring could take shape similar to a spiders web.
In this scenario we are stressing gaining access to continuous output over building a ring. Its key advantage is less reliance on storage, its 2 key disadvantages are a. the Polar light mountain must be proven to exist b. the aggressive development in polar regions is bound to be extremely hazardous. In such darkness the craft and crews would literally be working off an extension cord to the PLM power base. The extent of lunar winter may preclude this scenarion and force the following at a 6.5 degrees lattitude.

Scenario #2(reasonable amount of power storage) The Initial power system would be a trio of collectors at about 83 degrees north, with expansions to the south as terrain dictates, eventually outlining a circular power grid. A landing would be made in the plains north of Goldschmidt crater, just west of the meridian, with a good line of sight to Earth.

Full deployment of solar Cells and construction of a metal surfaced port facility would ensue to ensure dust free landing and takeoff of visiting craft. A road would be constructed to Gioja crater, to the north. From there the road would branch west to the nearside of Hermite crater and the East branch would travel to Nansen Crater passing just north of DeSitter, with a further branch there, with one road heading south south east to Schwarzschilds North Wall (an outstanding collector slope) and a westward branch headed to Cr. Plaskett. On the other road an extension to the west wall of Cr. Rozhdestvenskiy affords access to an excellent collector surface provided the surface is not shaded by the next destination, the northwest wall of cr. Plaskett. The road building should provide access to sufficient crater walls to provide continuous solar output as PVcells become available from the main base. The shape of the ring is dominated by Craters, the height of the north compared to the south walls and whether the East or West walls form high or extended ridges. Cr Nansen and Hermite are within 300 kilometers of Gioja.
Initial power output would be 90 Mw continuous excluding losses. As the Ring took shape output would increase to 90 Mw continuous at any point along the travel path continuous. Feeder lines could be run away from the ring in any direction to supply various forms of lunar enterprise, and eventually to the equator. This output represents 10x10 meter collectors spaced 100 meters apart along a 2000 km collector ring assuming about 600 km of collectors will be facing the sun. This is where proper utilization of terrain can increase power and fill up the troughs in output. It is actually better to have an assortment of steep slopes than have a smooth surface interms of collector density per given length of stretch.

Scenario #3- A Lunar Base is established far from the pole and as extra equipment and solar cells become available, investors pay for construction of a ring as a lunar power utility venture.

Use of Output- Electrical output may exceed other users demand and the excess output can be used to attempt the following:
Plasma refinement techniques, there may be large fragments of nickel iron meteorites that could be refined to extract the platinum group metals.
Plasma based construction techniques could be developed to construct metal chambers rather than fabricating from sheets of metal.
Certainly excess output could be applied to fabricating components of Power-Sats. If the power were used to run a large magnetic field, would we be able to deflect solar wind to concentrate and focus for collection of ions H and He, much like a Bussard scoop? This may evolve from Ion deflection radiation shields.
One form of output we do not want to see is misuse of the energy to run energy weapons. Additionally, this power system would hold some attraction to a Rogue AI. and may be a target of some suicidal hackers.
Linear accellerators(Linac) may be operated to enrich thorium and uranium for use in Mars missions, there may even be the possiblility of the production of Tritium and Helium-3 by Linac. A market may exist for modifying nuclear waste to make it more useful for deep space missions.
Antimatter production for research purposes would be possible when the ring reaches full upgrade. This would complement plans to build a fleet of vessels to explore the Jovian system and Asteroids. The Jovian moons have ample water resources for Martian developments in the future. Antimatter may be in demand.
A Laser-Sail facility could slow incoming craft thereby saving on fuel.
A laser resonator facility could test high energy lasers for transportation uses, and to study photons in general.

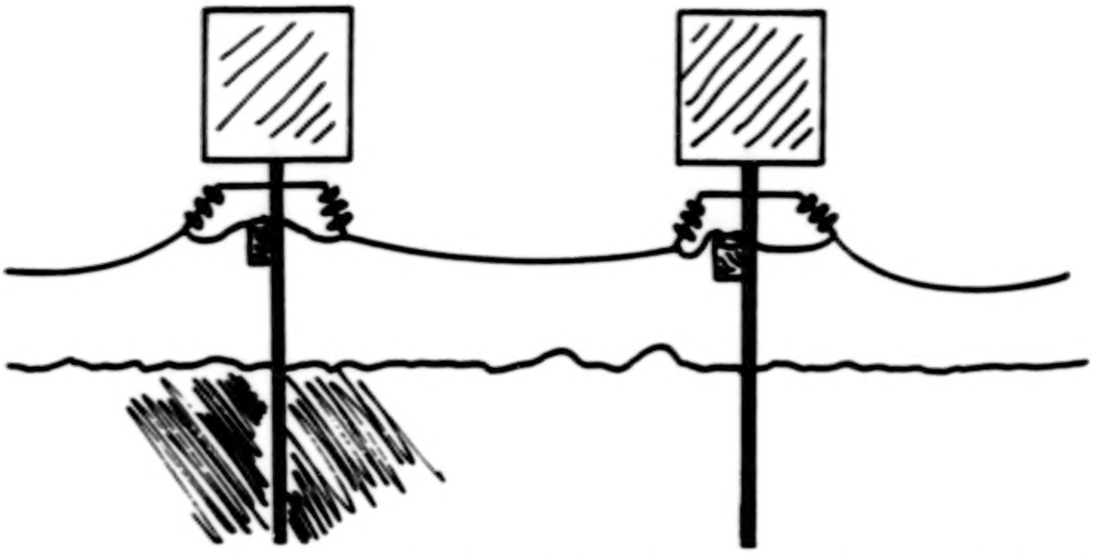

Figure 3. Solar Collectors on Glass composite poles simply tamped down holes augered into regolith. Tower spacing 20-100 meters, Scale 1cm = 5m

## RAIL LINE

Eventually a rail line would have to be constructed along the initial power ring. A rail line would allow the movement of components as large as portions of the 100 meter towers (for the 90 Gigawatt upgrade), power conditioning devices and spools of cable. A large gauge railway would allow radiation shielded human living quarters to travel to construction areas. Habitations would be quite massive, and thick-walled, rail alone would make them continually accessible. Ideally all or part of the line would be dual tracked to allow two way movement of personell and materials. Railways may seem primitive, however, they represent a mature technology with relatively easy guidance requirements. In addition it is reasonable to assume railways will eventually need to be constructed in order to conduct mining operations, and long term travel between bases as equatorial regions will dominate transportation. Rail transport is compatible with Lunar development as most lunar development will, in the long term likely be very large scale, serving Global needs. Lunar dust problems are avoided by using rail transport. The Iron for the rails could be cast from free iron via magnetic seperation from the lunar soil. This Iron would represent a resource cache previously refered to. A solar furnace could be used to melt the iron. It may be advisable to coat the Rails with Chromium refined from the lunar soil. The rail ties could be either Glass soil composites or cast composite Calcium metal. The calcium is assumed to be a by-product of Magnesium production.

LifeCredit Section- One of the best uses of electrical output during the lunar night is lighting Closed Environment Life Support Systems(CELSS). The Life Credit may assist in financing the LPR because the ring can provide sufficient energy to support photosynthetic light chambers.

Why is this usefull for Earth and Space Exploration?

There are fundamental rules regarding extinctions and Galactic Exploration:

1. an extinct life form(species) cannot be used to colonize a planet, repair, augment or support a CELSS. Its genetic life code is also lost. The individual diversity within the population of the species is lost and cannot be used for breeding desired characteristics. The rapid population increase that may be required is not possible as the reproductive methods

are non-existant. Even ideal conditions will not bring needed species back from the dead, and the humans at risk may not have time to wait for evolution to develop solutions to colonization problems, therefore conservation of existing life is essential and is equivalent or better than producing CELLS space technology. To expect biotechnology to produce wilderness or species on demand is perhaps an extreme act of faith, and would require an effort greater than conserving existing candidates from a (for now) fully functional biosphere now in its 17th circumnavigation of the Galaxy.

2. Evolution is the only offsetting factor against extinctions, as such it must be allowed to proceed on Earth, therefore wilderness must exist on Earth so that evolution can proceed unhindered and beyond the human intervention, just as it will also proceed under human direction.

3. It follows then that since some extinctions are a natural process in the long term, Humans can conserve the "naturaly deleted" life forms for later use, (In addition to species endangered by humans), for although the life form may no longer be suitable for current habitat on Earth, it may be usefull for the more recent and primitive barren habitats presenting in space. A Space Exploration insurance. IE-that which has outlived its usefullness in biosphere earth may be needed in new bioshperes, and should not be discarded as new applications and niches will result from space travel. We are the sentients, not the plants, our eyes should be open to the potential of life in the larger picture of the Galaxy.

4. Life originated on Earth, the best insurance against extinctions and to insure availability for use in space is a healthy habitat on the planet Earth, therefore to conserve habitat on earth is to conserve the biological resources that will assist in galactic exploration.

5. In the event other life is encountered most trade will initially be information. Our technology may be interesting for historical studies. Much more valuable the genetic information relating to biochemical functions, and life morphology from enzymes to the diversity of species. Especially so if the customers are DNA based. We may be destroying a treasure house of enzyme designs, worth the price of a Star Drive a dozen times over. For what in the Galaxy is rarer than life? Life truly is an interstellar class technology. As we travel further from our home star, we will quickly appreciate how valuable genetic data can be as we buy it to survive, colonize or sell back some diversity lost. Also- genetic information can be sent at the speed of light and would represent the beginnings of interstellar trade.(Ref.4 Feoktistov) Many reasons to conserve life.

The final argument for a large continuous power supply on the moon, and the LPR as worthy design, is simply this: Since humans must conduct agriculture to sustain themselves on the lunar surface and since this is a relatively inefficient use of energy, necessary nonetheless, large amounts of extra energy must be made available for growth of plants. Some plants may exist on earth that can readily adapt to the Lunar day cycle, but it is doubtful those species will produce oxygen and food in the dark. In the meantime the LPR can supply light, and the power to produce large Closed Environment Life Support Systems.

In conclusion the Polar Lunar Power Ring provides a permanent, long term, almost trouble free, solid state solution to the lunar night energy problem. It requires a initial lunar base capable of producing amorphous silicon Photovoltaic cells, power transmission cable and long distance rovers. The benefits include an eventual rail infrastructure, large scale plant growth capa lities, and plenty of power to conduct propulsion research. The cold traps on the Lunar pole can be enhanced to passive

cryogenics levels. The power output is inherently expandable and limited only by the output of solar energy conversion devices. Once constructed the electrical output can enable most of the lunar developments proposed. The ease and extent of potential space activity due to the capabilities of this power system may force a re-evaluation of Human goals that will place education, Conservation and the challenge of interstellar travel as 3 working surfaces of a single tool to carve a new future for Mankind. The dangers of the power system or the systems potential to alleviate environmental destruction are uncertain.
The forecast for lunar power systems is certain- sunny.

We were born to go
as far as we can fly.

-Hawkwind (In Search of Space)

Acknowledgements:

The author would like to thank Dr. Geoffrey A. Landis for his assistance in preparation of this paper by volunteering Ref 1. and 2. and providing comments which led to extensive revisions of the concepts as originally presented at the Lunar Bases and Space Activities in the 21st Century symposium April 7, 1988 Houston Texas. Also Dr. Robert L. Forward, for volunteering papers on daring propulsion concepts requiring power systems of this magnitude and Space Studies Institute for the tools to build with. A future version of the Polar Lunar Power Ring will include concepts by D. Criswell and J. Burke, and will be available from the Author in late 1990. Also of assistance was the Report of NASA Lunar Energy Enterprise Case Study Task Force July 89.

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Farside Cannons- Roger MacBride Allen
Flight of the Dragon Fly- Robert L. Forward
Ringworld and Ringworld Engineers- both by Larry Niven
Report of NASA Lunar Energy Enterprise Case Study Task Force, 7/89-TM101652
Potential Applications of Phase Conjugate Resonator Concepts- G.S. Galloway -pp 319 Space Manufacturing 6/ Faughnan and Maryniak, ed. Princeton 1987

N91-22179

# Applications of Thin Film Technology Toward a Low-Mass Solar Power Satellite

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## *Abstract*

Previous concepts for solar power satellites have used conventional-technology photovoltaics and microwave tubes. We propose using thin-film photovoltaics and an integrated solid-state phased-array to design an ultra-lightweight solar power satellite, resulting in a potential reduction in weight by a factor of ten to a hundred over conventional concepts for solar power satellites.

## *Introduction*

The concept of a Solar Power Satellite (SPS) to provide power for Earth was proposed in 1968 by Peter Glaser [1]. Glaser proposed to solve the energy crisis and provide abundant electrical power for Earth by putting large (1-10 Gigawatt) solar collectors into geosynchronous Earth orbit, and to transmit energy to the surface using a microwave beam. Solar power satellites based on this concept were extensively analyzed in the period around 1978-1981 [2-4].

The solar power satellite concepts examined in the late seventies had two significant difficulties: capital cost and weight.

In this paper we discuss the potential for two technologies currently under development to considerably reduce the mass-to-orbit required for such a satellite power system: thin-film photovoltaics and solid-state microwave electronics. It may be possible to design a solar power satellite to be constructed entirely by thin-film technology, consisting of thin (one to two micron) active components on a plastic substrate, with the microwave phased-array components integrated directly with the photovoltaics.

In essence, we propose discarding the "bridge-builder" mindset adopted by the initial designers of a SPS system, and adopt a thin-film, integrated-circuit mentality: why can't a solar power satellite be a single, integrated assembly deposited on a thin, lightweight substrate?

## *Thin film photovoltaics*

Thin-film solar cells consist of thin (thickness ~1-5μ) films of photovoltaic material deposited on a supporting substrate. In the 1980's a considerable research program has been devoted to development of thin-film photovoltaics for terrestrial power generation. Efficiencies over ten percent have been achieved on amorphous silicon and copper indium diselenide thin-films, and encouraging results achieved on other thin-film technologies such as CdTe and $CuInS_2$. Table 1 showsthe historical progress in efficiency of several of the thin-film materials over the last few years [5].

Because of the high optical absorption constant, for thin-film solar cells the active material may be as thin as one to two microns, and hence the materials inherently have the potential to be extremely light. However, very little current research is aimed at depositing thin-film cells on lightweight substrates, since most of the applications being currently considered are for terrestrial applications, where weight is not important.

Preliminary results show that thin-film solar cells appear to be inherently radiation tolerant, and may not require a glass cover for radiation protection [6]. They also are highly tolerant of small damage areas, such as due to micrometeoroid or debris impact.

A conservative projection would be to project use of a 5% efficient thin-film cell on a 25 micron thick Kapton substrate. This yields a photovoltaic blanket specific power of 1.7 kW/kg. An optimistic projection might be 15% thin-film cell on a 7 micron thick Kapton substrate, leading to a photovoltaic blanket specific power of 15 kW/kg. These numbers compare favorably to current technology spacecraft solar arrays, *e.g.*, 67 W/kg at the array level for the flight-tested SAFE array.

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Solar cell cost was highlighted as one of the single most critical improvements needed for a SPS in the NRC analysis. In addition to low mass, thin-film photovoltaics are also projected to have considerably lower costs. Materials cost is reduced due to the reduced amount of materials required; the cost of labor and assembly is reduced by the fact that large-area, integrated assemblies are directly produced on the substrate sheet.

For typical existing systems the photovoltaic blanket weight is only about a quarter of the total power system mass, with the array structure and power management and distribution (PMAD) accounting for the remaining three quarters. This provides a powerful incentive to integrate the PMAD elements directly to the solar array, and to design new array structures to take advantage of the ultralight blankets.

**Historical progress of thin-film solar cell efficiency. Experimentally achieved efficiencies (at AMO, in %) as of 1978, 1983, 1988, and projected values for future performance.**

| Material | 1978 | 1982 | 1988 | 1990's |
|---|---|---|---|---|
| $CdS/Cu_2S$ | 7.3 | 8.2 | 9. | 10 |
| $CuInSe_2$ | 5.3 | 8.5 | 11.2 | 12 |
| $CuGaSe_2$ | - | - | 4.6 | 12.5 |
| $CuInS_2$ | 2.9 | 2.9 | 6.1 | 12.5 |
| CdTe | 4.1 | 8.4 | 8.6* | 12.5 |
| a-Si | 4.4 | 8.1 | 9.0 | 11.5 |
| CASCADES | - | - | 12.5 | 17+ |

*(9.8% reported in May 1989)

*Microwave electronics*

Use of thin-film solar cells will reduce the satellite mass only if the mass required for the power management and the microwave beaming system can be reduced as well. This may also be achievable using solid state electronics. In the last ten years we have seen development of microwave integrated circuits, thin-film transistors, and thin-film microwave rectennas

Solid-state electronics has developed considerably, and it is now reasonable to consider that the microwave source for a SPS may be able to be manufactured from thin-film electronics.

Microwave rectifying antennas (rectennas) for receiving microwave power and converting it to DC power have been demonstrated using thin-film techniques on a thin-plastic substrate [7]. Using such a technology as a microwave source requires replacement of the GaAs diodes by appropriately phased microwave transistors.

Thin-film microwave electronics are a reasonable extrapolation of the union of two recent developments: solid-state microwave electronics, and thin-film transistors. Microwave integrated circuits are currently available which can operate in the gigaherz band proposed for SPS power transmission, and have been demonstrated by many different laboratories for operation at the tens and hundreds of gigaherz [8].

Thin-film transistors have been developed for other applications such as display screens, where many millions of devices can be integrated on a large-area sheet. Thin-film transistors are made from amorphous silicon or polycrystalline silicon, and could in principal be manufactured integrated onto the same substrate as the solar cells. Current technology only allows frequencies in the range of kiloherz to at most megaherz [9], but this will likely increase with further research. Development of gigaherz-speed thin-film transistors would allow the transmitter elements to be deposited on the thin substrate at the same time as, and possibly using the same materials as, the thin-film solar cells.

The concept for integration of the solar cell with the microwave oscillator and antenna is shown in schematic in figure 1. A slightly more complex version, where the solar cell metallization is used for the antenna in a "push-pull" configuration, is shown conceptually in figure 2.

Even without the development of thin-film microwave transistors, conventional microwave solid-state integrated electronics would be able to perform the function at a cost considerably below that of the microwave tube approach. Each element need operate only at a power level of a watt or less, well within the capability of microwave integrated circuits.

An alternate approach to operation of the solid-state microwave source at even higher frequencies is the use of superconducting electronics. Such technology is currently being researched; for example, a superconducting circuit operating at 33-37 GHz has been developed at NASA Lewis using thin-film YBaCuO at 77°K [10].

### *Micro-phased array distributed thin-film SPS*

In an highly-integrated SPS design, microwave oscillators and dipole antennas are be integrated directly on the plastic sheet with the solar cells, using phased-array techniques to steer the beam back along a pilot beam generated at the receiving antenna on the Earth. Rather than a smaller number of high-power microwave tubes, the integrated SPS will have billions of integrated transmitters, each operating at a power of no more than a watt. This integration would eliminate the power conditioning elements and the wiring used for power distribution.

The proposal consists of the following elements:

--Total integration. Microwave transmitters are integrated directly at the solar cell level. No wires or power management/distribution system is required.

--Thin-film technology. Lightweight photovoltaic films on a thin plastic substrate are used.

--Phased array technology. The antenna does not need to be physically "aimed" at the receiver.

The distributed thin-film SPS applies the integrated circuit approach to the satellite solar power concept.

Table 2 shows a comparison of the mass of an integrated thin-film solar power satellite compared to a baseline system. Using the "conservative" technology extrapolation the reduction in weight is by more than a factor of ten; assuming a more advanced technology a reduction in weight by over a factor of a hundred is achieved.

**Table 2:**
**Mass Comparison**

***Baseline SPS (1980):***

| | |
|---|---|
| 2.6 kg/kW | Transmission and control |
| 6.5 kg/kW | Silicon Solar Array |
| 0.6 kg/kW | Power Conditioning |
| **9.7 kg/kW** | |

***Thin-Film SPS (1990's):*** (5% efficient solar cell on 25μ Kapton)

| | |
|---|---|
| **0.7 kg/kW** | Solar array + integrated transmitter |

***Thin-Film SPS (2000+):*** (15% efficient solar cell on 7μ Kapton)

| | |
|---|---|
| **0.08 kg/kW** | Solar array + integrated transmitter |

It is important to design new, low mass structures in order to reduce the structural mass of the system proportionately to the photovoltaic and transmitter mass reductions. Many structural designs for such a system are possible. Two, the "bicycle wheel" configuration (shown in figure 3) and the "sphere" configuration (shown in figure 4), shown. The phased array microwave elements mean that the microwave antenna does not have to face directly toward the receiving station, as long as the antenna is not edge-on to the receiver. The "bicycle wheel" concept uses centrifugal force to place a thin circular membrane in tension. Since the size is large and the tension required small, a very low rotation rate, <<1 RPM, is sufficient to provide tension. Bracing cables to a central hub provide the requisite out-of-plane stiffness. If necessary, a counter-rotating flywheel can compensate for angular momentum.

Thin-film PV elements are light enough that high power to weight ratios may be maintained even if the satellite does not track the sun. In the sphere configuration, the solar cell/microwave transmitter elements cover the surface of an inflated sphere as in the Echo satellite. For a sphere radius of many hundreds of meters the surface/volume ratio is extremely low, and the gas pressure required to hold the form, and the associated leak rate, can be made small.

### *Applications*

The main application of a solar power satellite envisioned by Glaser and by most of the later advocates of satellite solar power was to provide baseline electrical power for terrestrial use. However, it is quite likely that some of the most important applications, and certainly some of the initial applications, will be in space. Existing power sources for use in space provide power at a considerably higher effective price than terrestrial power sources, and a remote power station would be able to serve several critical needs. Some of these are:

(1) Beamed power for lunar base night operation
(2) Inter-orbital ion-engine propelled transport spacecraft
(3) Power for Earth-orbital stations [11]
(4) Support for Mars missions and solar system exploration and exploitation.

### *Difficulties*

So far we have only discussed the potential advantages, and not the problems. The concepts outlined above have been schematic, not detailed engineering designs of how such a system could be built, and many problem areas remain to be addressed. We will only briefly identify some of the issues here, without attempting to detail all of the possible approaches.

A phased array system requires a pilot beam from the target to be directed to the microwave source. The pilot beam need not be at the same frequency as the output beam. A pilot beam at, or near, the same frequency as the power beam has the advantage of automatically correcting for atmospheric effects. A disadvantage of a pilot beam at the output frequency is the difficulty of distinguishing pilot beam from output (*e.g.*, by polarization difference). Failure to adequately isolate the pilot beam from the output beam would result in undesirable self-stimulated oscillations of the transmitter. One way to eliminate this problem would be to use a pulsed laser as the pilot beam. Each laser pulse would be used to set the phase signal.

The issues involved with phase-conjugation of the reference beam, including the problem of providing a reference clock signal and distributing the phase signal have not been addressed. This could be done either with analog processing or with digital circuitry. The difficulty of this problem is decreased if it is assumed that significant amounts of integrated processing capability is available at low cost.

If the elements are equipped with local oscillators, then the phase signal is only required to keep the local oscillators in correct phase. If the system does not have local oscillators, or if the local oscillators have poor frequency stabilty, a continuous phase signal is needed.

Off-angle losses have not been discussed. The antenna elements will be tuned to radiate at best efficiency at a given angle. Also, polarization issues have not been discussed. Finally the question of a backplane for the microwave antennae has also not been addressed. The shorter the wavelength used, the less difficulty this provides.

In general, all of the problem areas are amenable to engineering solutions. The important question is whether resolution of the problem areas would unacceptably increase the complexity, weight, or cost of the system.

### Conclusions

Thin-film photovoltaics and microwave solid-state devices have the potential to create a revolutionary improvement in solar power satellite design, with possible improvements in power to weight ratio of a factor of ten to a hundred. Thin-film photovoltaics alone could cause a significant performance improvement, however, to take full advantage of the technologies being developed, we have suggested design of a fully integrated photovoltaic/microwave system, where the phased array microwave elements are deposited integrally to the solar cells, eliminating all the power management and distribution.

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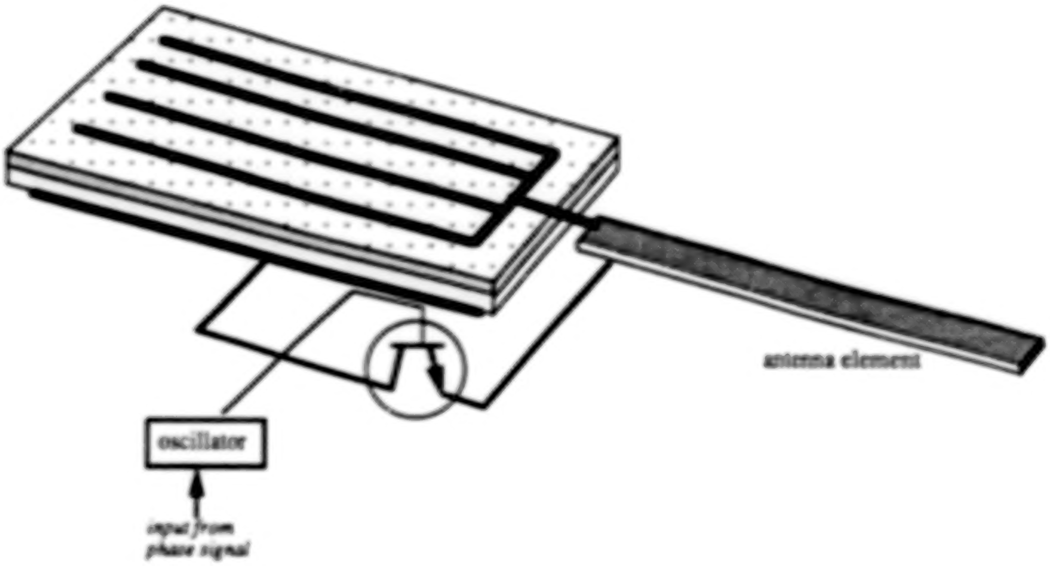
antenna element
oscillator
input from
phase signal

Figure 1. Solar cell with integrated microwave antenna element (conceptual diagram). The local oscillator receives a phase signal to make the output signal coherent as part of a phased array beam.

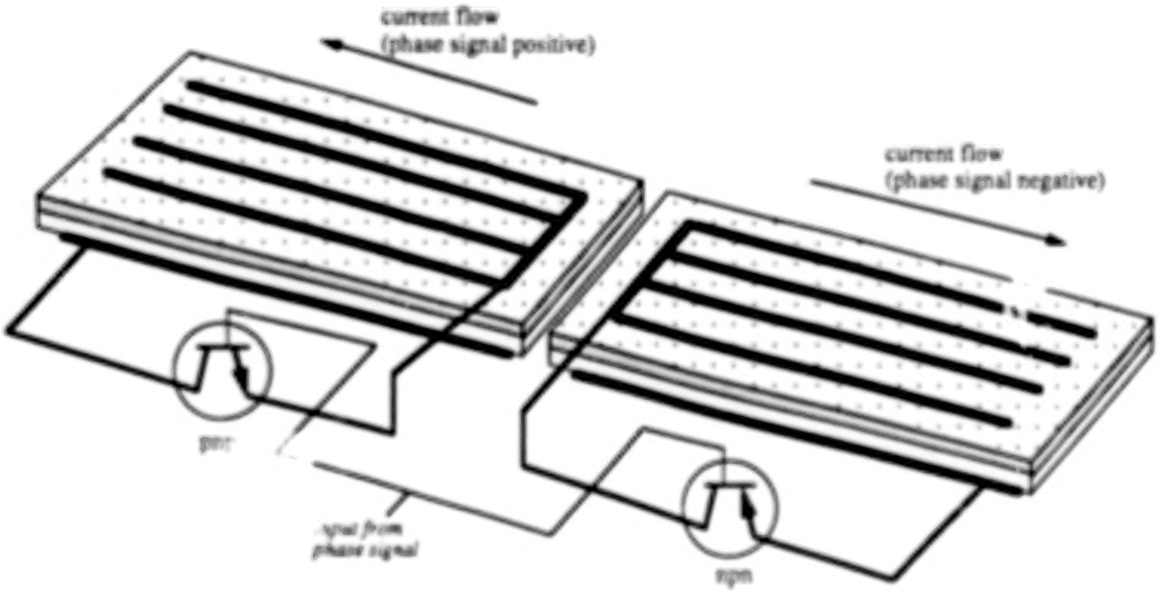
current flow
(phase signal positive)
current flow
(phase signal negative)
pnp
input from
phase signal
npn

Figure 2. Schematic diagram for integrated solar cell/transmitter conmination in a push-pull configuration. The contact metallization of the solar cells serves as the antenna element for the integrated microwave transmitter. Complementary pnp and npn transistors receive the same phase signal

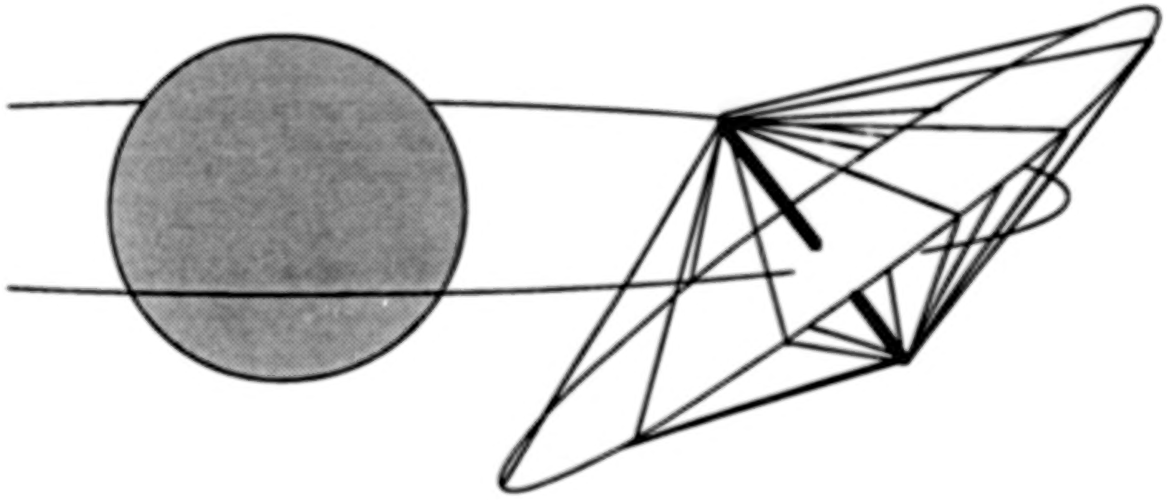

Figure 3: "Bicycle-Wheel" SPS Configuration

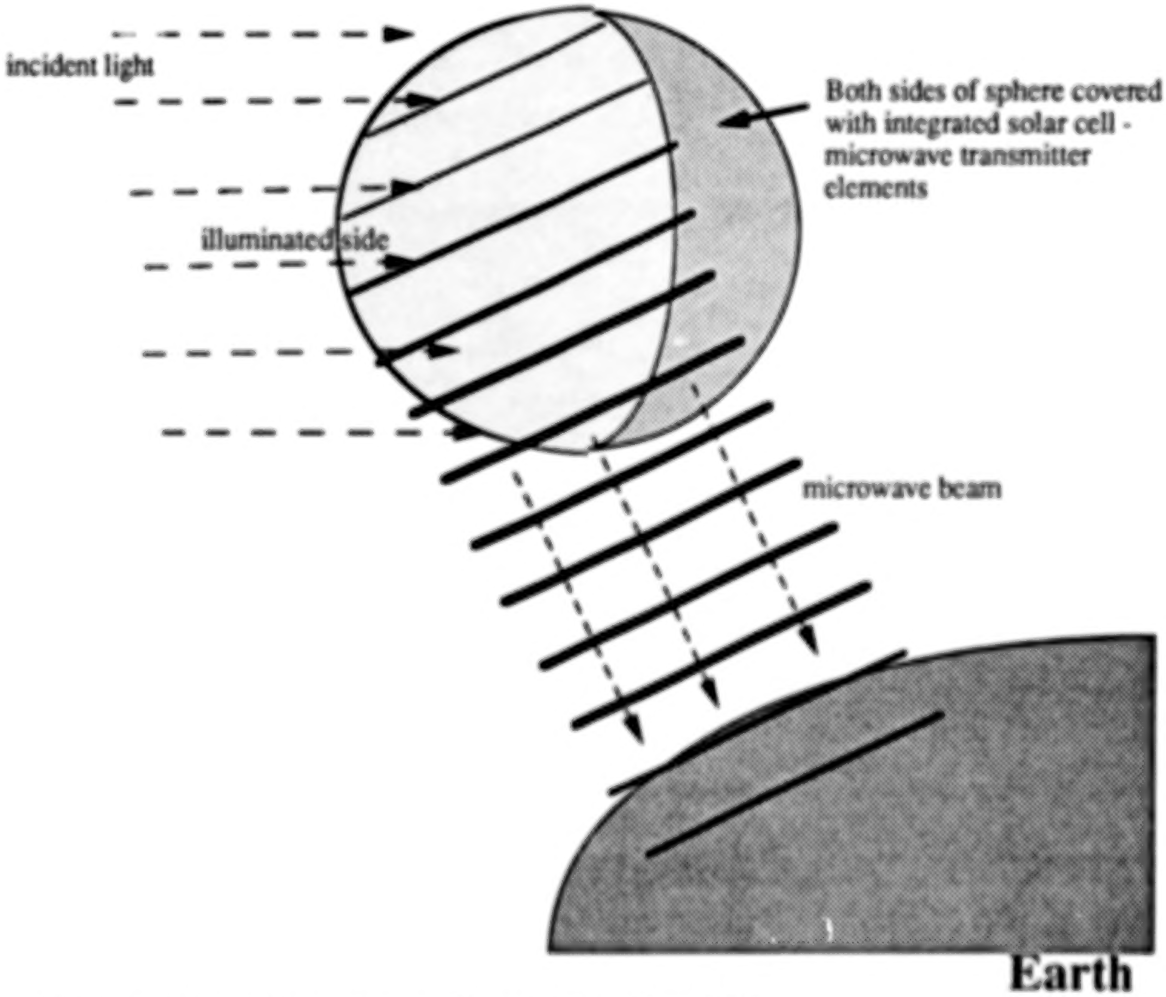
incident light
illuminated side
Both sides of sphere covered with integrated solar cell - microwave transmitter elements
microwave beam
Earth

Figure 4: "Sphere" configuration for SPS.

N91-22180

# PLANETARY MATERIALS AND RESOURCE UTILIZATION: AN INTERDISCIPLINARY ENGINEERING DESIGN COURSE AT MICHIGAN TECHNOLOGICAL UNIVERSITY

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## ABSTRACT

A new course was developed and instituted in the spring quarter of 1989 dealing with topics related to space resource utilization and related engineering. The course development required a concerted, coordinated effort, because a similar course which might be used as a guide could not be identified anywhere and the interdisciplinary perspective that was required was not concentrated anywhere on our university campus. A dominant role in the course was played by 13 visiting speakers from NASA, USGS, private companies and universities who each gave 2 or 3 lectures. Ten faculty in six different departments provided introductory and connective lectures in the course. Students in the class worked on interdisciplinary design projects which culminated in papers and oral presentations. Each of the six design groups consisted of several engineers with different disciplinary roots. The entire course lecture sequence, about 50 hours in all, was videotaped. We have edited this resource for distribution to others interested in this topic area. In this paper, we will discuss our experiences in developing the course, including the course syllabus and speaker list.

## INTRODUCTION

The original motivation for the course was to introduce faculty, graduate students, undergraduates and our own community to the possibilities and engineering challenges involved with lunar bases and space manufacturing. We perceived that President Bush's recent statements about the future of planetary exploration would, if implemented, foster interesting career opportunities for engineers, and as one of the country's largest engineering schools, we felt that these opportunities should be emphasized. As an additional motivation, many of the faculty with space-related interests saw the course as a means to develop interdisciplinary engineering efforts with colleagues who had similar research interests. After attending an April 1988 NASA Symposium on Lunar Bases and Space Activities in the 21st Century, we decided that the interdisciplinary scope required to effectively solve lunar base and space manufacturing/engineering problems was not currently being realized in our engineering curricula.

## INITIAL EFFORTS

It was decided that a new course was necessary to address space related engineering problems, but it was not clear what form was appropriate and what content should be included. Existing courses in several different departments contained some relevant material. However, an appropriate combination of these existing courses was never assembled because an established degree program focusing on space engineering problems was not available. Several factors led us to develop a single interdisciplinary course that would allow us to stimulate students and faculty to become involved with a new interdisciplinary curriculum. No individual faculty member at Michigan Tech had the breadth to organize a course dealing with a variety of space engineering topics. We also felt that after developing a single course, we would have a better appreciation of what individual topics could best be expanded into separate courses. Finally, the effort required to develop a sequence of courses would have placed a tremendous burden on even a core group of interested faculty.

Our activities resulted in a course for engineering students which gave them sufficient background on extraterrestrial materials and processing conditions so that specific design problems could be examined. The course had a strong orientation toward planetary materials utilization since the core group of faculty had backgrounds in materials science, mineral processing, and geology. We decided that the new course should be accessible to advanced undergraduates, who had basic engineering and science backgrounds, and to graduate students at different levels who would provide more academic maturity to the student population. We also hoped that a broad group of faculty would participate in the course.

We were unsuccessful in identifying appropriate courses on other campuses that could serve as a useful guide. Courses dealing with lunar and planetary geology exist on many campuses but focus on purely scientific topics. Engineering courses were available dealing with topics such as microgravity and materials given by Jean Koster at the University of Colorado that were rather specialized. What we wanted was a course that emphasized a strong interdisciplinary engineering design element related to space resource utilization.

Several procedural problems with regard to the "mechanics" of the class also arose. It was not clear, because of our desire for an interdisciplinary format, what department the course should reside in. We were concerned that if an upper division course was assigned to a particular department, students in other departments would be reluctant to take it. University procedures for courses not specifically assigned to an academic department were very cloudy and we were unsure how students would react to becoming involved in such courses. This presented us with an important issue faced by university faculty interested in interdisciplinary research and educational topics and who want to involve students in an upper division class outside of their normal curriculum. Our solution was to use "wild-card numbers" for classes designated as "Advanced Topics" in various departments.

We realized that visiting speakers could play a crucial role in the course given the limited number of internal faculty with space-related experience. It was also felt that external speakers could stimulate further research projects by interacting with faculty and graduate students. We had difficulty devising a comprehensive list since each of the core group of faculty had ideas for appropriate external scientists and engineers. It also became clear that we needed financial resources to support the travel of the external speakers and an individual to organize the course; the latter was necessary since it would be difficult for any single faculty member to have enough time to devote to all the details of the course organization.

We convinced university administrators to provide funds on the basis that such a course would help initiate a creative new engineering program at Michigan Tech which did not seem to be available at other universities. In addition, it was argued that the course would help foster a greater interaction between university faculty and various external groups already examining such problems. Since we planned to videotape the lectures of the outside speakers, we also pointed out that the course would benefit the university by having these lectures available as an up-linked, televideo course and as a non-profit educational package. The budget was divided into three major categories: video-production costs, travel expenses for the external speakers and support for a course coordinator. Jim Paces was selected as the coordinator since he was a recent Ph.D. graduate from the Geology program at Michigan Tech and had been a NASA Graduate Student Trainee at Johnson Space Center. The latter gave him a number of contacts we used in developing our external speakers list.

## ORGANIZATIONAL DETAILS PRIOR TO THE CLASS

The initial task of the course coordinator was to organize a course outline (Table 1) and to develop a list of external speakers (Table 2). At first, the course syllabus changed constantly. Although we were interested in a very broad focus, it became obvious that we should concentrate on topics that the internal faculty had expertise in. As a result, the course content focused on self-sufficiency in future space operations by examining in-situ resource utilization and microgravity processing of materials. Topics covered in the course could be placed, therefore, into three broad categories: non-terrestrial geological resources (lunar mineralogy, petrology, surface conditions, remote sensing), utilization of non-terrestrial resources (geotechnical and mining, lunar power systems, lunar materials processing systems), and utilization of non-terrestrial conditions (low gravity and high vacuum materials processing in space).

We were fortunate to have the help of Dr. Wendell Mendell of NASA-JSC in developing the external speaker list. After examining an early copy of the course outline, Dr. Mendell provided a list of potential speakers in each of the topic areas we had selected. Most of the "targeted" outside speakers accepted our invitation and their stimulating lectures were an integral part of the success of our course. Each speaker usually gave two different talks: a more-technically oriented lecture for the class participants, and a more-generalized presentation open to the public. The involvement of Dr. Mendell and other NASA personnel greatly contributed to this portion of the course.

Another responsibility of the coordinator was to build interest in the new course within the faculty and students of Michigan Tech. This was important because internal faculty could summarize fundamentals, introduce the concepts discussed by external speakers, and bridge any gaps between presentations by

## Table 1

### Planetary Materials and Resource Utilization
Spring Term 1989
Syllabus

Week 1

| Speaker | Topic |
|---|---|
| W. I. Rose, GE | Organization, Introduction and Purpose |
| **Paul Spudis** | Geological History of the Moon |
| **Paul Spudis** | Exploring the Moon in the 21st Century |
| **Paul Spudis** | A Lunar Base in the Crater Mare Smythii |
| Week 2 | |
| **John Alred** | Engineering a Lunar Outpost |
| **John Alred** | NASA Office of Exploration Future Studies |
| J. B. Paces, GE | Lunar "Ore" Deposits: Mineralogy |
| S. T. Bagley, BIO | Biological Extraction Processes |
| Week 3 | |
| W. I. Rose, GE | Magmatic Processes and Rock Types |
| W. W. Predebon, ME | Surface Processes: Impacting |
| S. K. Kawatra, MY | Surface Processes: Radiation |
| **Larry Taylor** | Lunar Mineralogy and Petrology |
| **Larry Taylor** | Origin and Evolution of the Lunar Soil |
| **Larry Taylor** | Thin Sections of Lunar Samples: A Demonstration of Mineralogy and Petrology of Rocks and Soils |
| Week 4 | |
| **J. Zimbelman** | Mars as a Planet |
| **J. Zimbelman** | Exploration of Mars: Past, Present and Future |
| W. I. Rose, GE | Remote Sensing - Polar Orbiter Mission |
| Lab Session | Lunar Thin Section/Remote Sensing Lab |
| Week 5 | |
| B. D. Alkire, CE | Geotechnical Engineering |
| **David Carrier** | The Strength of Lunar Soil |
| **David Carrier** | Mining on the Moon |
| **Ron Sovie** | Power Systems Options for Lunar/Space Base Applications |
| **Ron Sovie** | Power Systems for Production, Construction, Life Support and Operations in Space |
| Week 6 | |
| **Dave McKay** | Processing Raw Materials |
| **Dave McKay** | Emerging Technology for Utilization of Lunar Resources |
| S. K. Kawatra, MY | Concentration/Separation |
| B. J. Pletka, MY | Ceramics Engineering |
| Week 7 | |
| **James Blacic** | Properties, Production and Application of Lunar Glasses |
| **James Blacic** | Melt Tunneling: An Example of Unique Lunar Engineering Applications |
| J. E. Pilling, MY | Cellular Structures |

| | |
|---|---|
| **Week 8** | |
| R. W. Kolkka, MA | G Jitter |
| A. Agrawal, ME | Two Phase Fluid Dynamics |
| **R. Boudreault** | Materials Science and Manufacturing in Microgravity |
| **R. Boudreault** | Can You Really Use Space to Help Diabetics? |
| **Week 9** | |
| **Jean Koster** | Low G Science: A Challenge in Fluid Mechanics and Transport Physics |
| **Jean Koster** | Behavior of Fluids in Space |
| A. Hellawell, MY | Aspects of Materials Processing |
| **John Perepezko** | Drop Tube Experiments & Low-Gravity Simulation |
| **John Perepezko** | Containerless Processing in Space |
| **Week 10** | |
| **Martin Glicksman** | Solidification Experiments in Low-G Environments |
| **Martin Glicksman** | University Involvement in Space Programs |
| **Wendell Mendell** | Human Exploration and the Need for Space Resources |
| **Wendell Mendell** | Justifications, Policies, and Key Technological Problems for 21st Century Space Colonization |
| Class Project Presentations | |

the visiting speakers. We also wanted participation from a number of different disciplines. Many faculty and students were skeptical initially about the course since they were unfamiliar with NASA's long-range plans and/or confused about the indecisive stance of the government and the populace towards the societal importance of space issues. A publicity campaign was mounted in which articles in campus and local newspapers, bulletins, and flyers described the course content. This campaign was devised to encourage student participation from the widest possible spectrum of engineering and scientific backgrounds. Our only selection criterion was that the student have upper level (senior or graduate student) status. The most effective aspects of the publicity was through word-of-mouth. These efforts of the course coordinator were essential in broadening participation and giving us a critical mass.

## THE CLASS ITSELF

Special consideration had to be given to classroom logistics since the entire course was to be videotaped. Class lectures were held in a video-classroom complete with video-production equipment. Unfortunately, viewing conditions were compromised for the audience; the video-format forced the class to view slides and overhead projections on TV monitors rather than having these visual aids projected onto a screen for greater clarity. The small size of the room (30 seat capacity) also required any overflow audience to view the lecture in an adjoining room with a live video-feed. In addition, the opportunity to question/clarify points as they were raised during the class was precluded by the continuity required in the videotape. These compromises were necessary, however, if high quality videotape of the visiting lectures were to be obtained for use as a resource in future courses. A large lecture hall was employed for the evening presentations to allow for a larger community audience although the quality of the videotape was degraded. Often, more than 100 attended the evening lectures.

## Table 2

## Planetary Materials and Resource Utilization, Spring 1989
## Invited External Speakers

**John W. Alred**. Exploration Studies Office, Code 1Z2, Vangard Building, NASA Johnson Space Center, Houston, TX 77058: Lunar Base Study Systems; Bootstrapping; General Space Systems Engineering.

**Richard Boudreault**. Canadian Astronautics Limited, 1050 Morrison Drive, Ottawa, Ontario K2H 8K7: Microgravity Processing.

**W. David Carrier**. Bromwell and Carrier, Inc., P. O. Box 5467, Lakeland Florida 33807: Geotechnical; Mining; Civil Engineering.

**James Blacic**. Geophysics Group, Division of Earth and Space Sciences, MS C335, Los Alamos National Laboratory, Los Alamos, New Mexico 87545: Lunar Glasses.

**Martin E. Glicksman**. Materials Engineering Department, Rensselaer Polytechnic Institute, Troy, New York 12181: Solidification and Microstructures; University Involvement in Space Programs.

**Jean Koster**. Associate Director, Center for Low Gravity Fluid Mechanics and Transport Phenomena, Dept. Aerospace Engineering Sciences, Engineering Center, Campus Box 429, University of Colorado at Boulder, Boulder, CO 80309-0429: Microgravity Fluid Dynamics and Materials Science.

**David McKay**. Planetary Science Branch, Mail Code SN2, NASA Johnson Space Center, Houston, TX 77058: Space Resources Utilization; Lunar Oxygen; Lunar Materials Processing.

**Wendell W. Mendell**. Mail Code SN21, NASA Johnson Space Center, Houston, TX 77058: Space Resources Utilizations; Lunar Base Concepts.

**John Perepezko**. Metallurgical and Mineral Engineering, University of Wisconsin, Madison, WI 53706: Containerless processing; Simulation of Low Gravity Conditions; Drop Tube Experiments.

**Ronald Sovie**. NASA-Lewis Research Center, Mail Code 301-5, 21000 Brookpark Road, Cleveland, OH 44135: Lunar Power Sources.

**Paul Spudis**. Branch of Astrogeology, U. S. Geological Survey, 2255 N. Gemini Drive, Flagstaff, AZ 86001: Planetary Geology and Petrology; Lunar Origin Theories.

**Lawrence A. Taylor**. Dept. of Geological Sciences, University of Tennessee, Knoxville, Tn 37996-1410: Mineralogy, Geochemistry and Petrology of Lunar Materials.

**James R. Zimbelman**. Center for Earth and Planetary Studies, National Air and Space Museum, Smithsonian Institute, Washington, DC 20560: Planetary Science; Viking Results; Mars Geology.

Twenty-two students enrolled in the course; half were undergraduates and half were graduates from five different departments. Because of the nature of the course, no specific textbook was utilized although the outside lecturers supplied us with pertinent reading material and/or references. The students were encouraged to use the symposium volume "Lunar Bases and Space Activities of the 21st Century" (ref. 1) as background reading. These papers also served as models for the design projects (Table 3) which formed an essential part of the class requirements. The students were divided into six groups composed of four or five individuals. We selected the groups so that a diverse set of disciplines was present in each group, and an advanced graduate student (usually a doctoral candidate) led each design group; one or more of the internal faculty acted as advisors to each group. Each group was responsible for writing a paper using the format in the symposium volume as a guide and giving an oral presentation to their colleagues and faculty. The presentations and papers formed the basis of the assigned grades.

Students were asked to evaluate the class. They were very positive about the visiting lecturers (including the opportunity to interact with them), the topics that were covered, and the design emphasis. They were most critical of the uneven coverage, gaps, and overlaps that resulted from having a number of different speakers. The other problem was that because of the interdisciplinary nature of the participants in the course, some students (and faculty) were lectured on material they were already quite familiar with while others did not have an adequate background for certain topics. Although we worked from a logical course outline, we could not control the individual content of the lectures from both internal and external speakers. This sometimes led to obscure and frustrating transitions between speakers. With the videotape resource, the internal faculty are now in a much improved position. We are capable of designing and preparing the necessary introductory lectures with emphasis on terminology, fundamentals, and context, as well as constructing the bridges necessary to link the lectures of the external speakers. The design group experiences were particularly positive and overshadowed all of the minor problems.

## SUMMARY

Although it required a concerted effort on the part of the core faculty and the coordinator, we believe that the objectives we had envisaged for the course were met effectively. The course will be offered in almost the same format this summer (1990) at an accelerated pace. We expect to add new, more specialized courses to the curriculum and to eventually establish a degree program in space-related engineering. Editing of the technical presentations of the external speakers is completed and these lectures are ready for distribution. Individual lectures are available and there are also "short course" groupings of lectures on topics such as: "Geology and Mineralogy of the Moon" and "Scientific Use of Microgravity". A condensed 10 hour short course version of these tapes is also planned.

## REFERENCES

1. Lunar Bases and Space Activities of the 21st Century (ed. W. W. Mendell), Lunar and Planetary Institute (1985).

## Table 3

### Planetary Materials and Resource Utilization, Spring 1989
### Class Design Projects

Purpose: Form small working groups in order to investigate a given design problem related to lunar base development. Use an integrated, multidisciplinary approach taking advantage of the individual backgrounds of each group member.

1) Define specific aspects of the problem
2) Determine appropriate requirements
3) Research background and principles
4) Propose solution

Goal: Prepare a concise paper modeled after those presented in Lunar Bases and Space Activities in the 21st Century, 1984, Lunar and Planetary Institute

Topics:

I. REGOLITH BRICK PRODUCTION

Most appropriate raw materials; brick design/geometry; production techniques; energy requirements.

II. SOLAR POWER SYSTEMS FOR MATERIALS PROCESSING

Direct (thermal) vs. passive (photovoltaic); furnace design; transport and assembly on lunar surface; site/geometry considerations.

III. LUNAR $O_2$: COMMINUTION

Bedrock (mining, fracturing, blasting) vs. regolith (particle size distribution, screening); maximum particle size requirements; dry milling and handling techniques.

IV. LUNAR $O_2$: SEPARATION AND CONCENTRATION OF ILMENITE

Distribution and form of oxide particles; dry handling; purity requirements.

V. LUNAR $O_2$: EXTRACTION OF $O_2$

Mineral chemistry; reduction reactions; hydrogen source; by-products; cryostorage.

VI. DEALING WITH DUST

Particle distribution; cohesion; filtration; abrasion.

N91-22181

# ANTIMATTER APPLIED FOR EARTH PROTECTION FROM ASTEROID COLLISION

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## ABSTRACT

An earth protection system against asteroids and meteorites in colliding orbit is proposed. The system consists of detection and deorbiting systems. The analyses are given for the resolution of microwave optics, the detectability of radar, the orbital plan of intercepting operation, and the antimatter mass required for total or partially blasting the asteroid. Antimatter of 1kg is required for deorbiting asteroid of 200 m in diameter. An experimental simulation of antimatter cooling and storage is planned. The facility under construction is introduced.

## INTRODUCTION

The space activities of mankind have been supported by

1) curiosity in sense to find new laws of universe and in exploration to find new world,

and will be so in future. Another motivation of space activities especially in future is

2) desire to preserve the human race, i.e. instructive move for survival.

The second motivation may take the form of Solar Power Satellite (SPS) and human resource exploitation, i.e. helium 3. These are counter action against the energy crisis coming in future. The energy crisis is related to the expansion of human activities. On the other hand, there is another type of crisis, the natural disaster caused by asteroid collision with the earth.

Asteroids approaching to the earth have so high relative velocities about 30 km/s that the kinetic energy is extremely large even if it is much smaller than the earth. Asteroids collision with the earth result in not only the craters formation and tidal waves but also the earth environment modification. The Hiroshima atomic bomb has an energy of $10^{15}$J, which corresponds to the estimated kinetic energy of a meteorite, 10 m in diameter and 5 in specific gravity. Celestial body in 10 m class crashes on the earth once every several hundreds years. It is reported that a collision with a meteorite in the Cretaceous period changed the climate and exterminated the dinosaur in the mass. The collision corresponds to 5 billion Hiroshima bombs. The asteroid collision equivalent to the dinosaur extinction happens once per 100 million years[1]. Asteroids very frequently near-miss the earth [2] as shown in Fig.1. Figure 1 is not a complete list since the asteroids in Fig.1 were accidentally observed by voluntary observers. The 1989 FC passing by the earth on March 22 in 1989, had several times as large kinetic energy as the asteroid which formed the Arizona's famous 1.2km-meteocrater.

Antimatter annihilation propulsion for interstellar and deep space missions has been recently studied because of high energy density of antimatter-matter reactions. Most of the studies emphasize the mission analyses and the conceptual designs of antimatter engines. On the other hand, we have been studying the storage of the antimatter and considering the application of the earth protection system against asteroid collisions.

The asteroids of 10-100 m size, which had impinged in the ocean and caused global weather impact, may not be recorded in the history books.

The earth protection system presented here not only actively detects the celestial bodies approaching to the earth but also modifies their orbit. The objectives of this paper are:

1) to estimate distance to detect asteroids from the earth and remaining time before collision,
2) to estimate requirements for radar system characteristics using Very Long Base Line Interferometry (VLBI),
3) to estimate the amount of antihydrogen for the orbital modification of meteorites,
4) to examine antimatter storage,
5) to design the antimatter factory and base.

## REMAINING TIME BEFORE ENCOUNTER

In order to make the analysis simple, three dimensional effects such as the inclination of the asteroid orbit are ignored. Meteorites are either comets or minor planets. Based on orbital data of comets[3], the eccentricities are found around 0.8 and the distance of perihelion ranges from 0.13 to 0.98 a.u.. Figure 2 shows the relation between the remaining time and the distance. The asteroid is assumed to move from the aphelion toward the orbit of the earth in the calculation. While the orbital radius distributes from 2 to 3 a.u. in the case of meteorites which are categorized as minor planets. It is concluded from these results that the distance to have to detect the asteroid is 1 a.u. and the remaining time before encounter is from 2 to 3 months.

## RADAR SYSTEM BY VLBI

The radar system is required to detect and track an object approaching to the earth as soon as possible. Suppose the asteroid of 100 m in diameter is detected at 1 a.u. distance from the earth. A great number of celestial bodies are observed by the photographic method with large telescopes. Asteroids approaching to the earth are discovered and tracked by the radar. The precision for the tracking radar requires 0.1 nrad in angular resolution. Such a resolution can be achieved by VLBI in radio astronomy such as VSOP (VLBI Space Observatory Program) in Institute of Space and Astronautical Science (ISAS). For the purpose of determination of transmission power of microwave, wave length, diameter of antennas and baseline distance, it is assumed:

1: Antenna for microwave transmission are the same size as receptive one.
2: Receptive sensitivity is $10^{-20}$ W.
3: Microwave is scattered at the asteroid's surface uniformly and isotropically.
4: Reflectivity of asteroids is 0.1.

The first assumption is made only for convenience. The second precision corresponds to an observation of a radio galaxy with 1 mJy ($1\text{Jy}=10^{-26}\text{Wm}^{-2}\text{Hz}^{-1}$) by a radio telescope which is 100 m in diameter and 100 kHz in band width[4]. The third assumption is very natural since the surface unevenness is larger than the wave length. The reflectivity of Apollo objects is assumed to be 0.1.

Now we derive the relation between the transmitted and the received power. Microwave beam diverges with transmitting distance. The theoretical minimum for beam collimation is given by [5]

$$A = \frac{4}{\pi} L^2 \left(\frac{\lambda}{d}\right)^2 \qquad (1)$$

A : cross sectional area of microwave beam
L : distance between asteroid and the earth
d : diameter of transmit and receptive antenna
$\lambda$ : wavelength of microwave.

The power received by the receptive antenna is given by the equation,

$$P_r = \frac{\frac{\pi}{4} D^2}{A} \cdot \frac{\delta \Omega}{4\pi} P_t$$

$$= \frac{\pi^2 D^2 d^4}{4^3 L^4 \lambda^2} P_t \qquad (2)$$

$P_r$ : received microwave power
$P_t$ : transmitted microwave power
$\delta\Omega$ : solid angle of receptive antenna measured from asteroid
$D$ : asteroid diameter

The angular resolution of VLBI can be estimated to be the order of $\lambda/L$. The power required for each satellite is about 10 kW if the millimeter wave is transmitted at 10 Hz repetition with the pulse width of 10 $\mu$s. The required characteristics of the radar system are summarized in Table 1.

Table 1: Specifications of the Radar System

| | |
|---|---|
| diameter of asteroid | : 100 m |
| diameter of antennas | : 25 m |
| received power | : $10^{-20}$ W |
| wave length | : 0.1 mm |
| peak power of microwave | : 100 MW |
| number of transmission satellites | : 10 |
| baseline distance | : 1000 km |

As exhibited in Table 1, these satellites are only 2.5 times larger than that of the VSOP in size. The surface accuracy of the receptive antenna, however, will be required at least 2 order of magnitude higher than that in VSOP since the parabolic surface of the VSOP is controlled to maintain within the small displacement of 0.1mm. It is indicated that additional difficulties are found by the VLBI of millimeter range at present[6].

## ANTIMATTER REQUIRED FOR MODIFICATION OF ASTEROID ORBIT

For small asteroids, the interceptors loaded with the antihydrogen can destroy them completely. If the asteroid is too large to be entirely exploded, it is necessary to penetrate into the celestial body and blast off the surface materials effectively. It is estimated in the case of orbital change using explosion that the energy utilization efficiency is less than 1% i.e. the ratio of the exploded mass to the remaining mass of the asteroid.

The antimatter-matter annihilation generates shock wave and produces high energy plasma at the center of the explosion. If the plasma dissipates its energy to surrounding materials efficiently, lava with high energy will be blasted off. The reaction against the asteroid produces the thrust. The energy E generated by the annihilation is related :

$$E = \frac{4}{3}\pi r^3 \rho \left( \frac{\Delta V^2_{\mathrm{orbit}}}{2} + \frac{U^2}{2} \right) \qquad (3)$$

$\rho$ : asteroid density
r : diameter of lava region ( depth of penetration )
M : asteroid mass
$\Delta V_{melt}$: effective velocity corresponding to melting energy
U : mean velocity of lava

The first term of the right hand side of Eq.(3) represents the internal energy and the second one does the kinetic energy of the lava. It is assumed that a half of the lava blasts and contributes to the orbital modification, and the rest half merely heats up surroundings. The velocity change V and the efficiency $\eta$ are calculated:

$$V = \frac{EU}{2M\left(\Delta V^2_{melt} + U^2\right)} \qquad (4)$$

$$\eta = \frac{EU^2}{8M\left(\Delta V^2_{melt} + U^2\right)^2} \qquad (5)$$

The radius r is a control parameter of the explosion (or thrust generation). Choosing r so as to maximize the efficiency:

$$V_{opt} = \frac{E}{4M\Delta V_{melt}} \qquad (6)$$

$$r_{opt} = \left(\frac{3\,E}{4\,\rho\,\Delta V^2_{melt}}\right)^{1/3} \qquad (7)$$

are obtained as the optimum velocity change and penetrating depth. For example, the optimum depth is estimated to be 240 m from Eq.(7) for the antihydrogen of 1 kg.

Figure 3 shows the relation between the mass of the antihydrogen and the delta-V calculated from Eq.(4). A value of 1m/s is the minimum delta-V required for the orbital change if the orbit is modified at 1.3 a.u. distance from the sun. Generally, the minimum delta-V is a function of the orbital elements, the direction of the thrust and the distance from the earth. The closest distance between the earth and the asteroid is plotted in Fig.4 with thrusting directions as parameters. The optimum direction is either parallel or anti-parallel to the orbiting direction of the asteroid. The delta-V at $\vartheta = \pi/3$ is ten times as high as that at $\vartheta = 0$ for given distance. Figure 5 shows the closest distance as a function of delta-V on deorbiting position as parameter. The delta-V required for $1.55\text{x}10^8$km is ten times as large as that for $2\text{x}10^8$km. As the farther distance the orbit is modified, the smaller the delta-V is required. This means the importance of the early detection and the orbital modification as soon as possible.

## ANTIHYDROGEN STORAGE

At present, antiprotons are generated by the method of a collision between a heavy metal target and a proton beam which is accelerated up to several tens of GeV or more. Reference 7 reports that $10^{n}$ antiprotons (~pg) are obtained per hour in Fermilab. The productive amount of the antiprotons has been increased at the rate of 10 times per 3.5 years ever since the discovery by Segre and Chamberlain so that the antiproton will be available industrially in 2020's if it monotonically increases. It is necessary that the antimatter is stored as solid antihydrogen at cryogenic temperature since the antimatter required for the orbital modification amounts to the order of kg or more as seen in Fig.3.

The storage processes are shown in Fig.6. At first, the produced antiparticles are cooled by stochastic and electron coolings[7] because antiprotons are tremendously hot just after they are generated by an accelerator. They are decelerated as slow as several keV and are turned into the antihydrogen by three body recombination with cold positrons. The unrecombined particles are cycled in the antiproton and positron rings being collimated with accelerator and electrostatic lens. The resultant antihydrogen beam is decelerated and trapped by means of laser cooling. A vacuum ultraviolet CW laser for the hydrogen cooling have not been accomplished yet, but will be put to practical use in near future with stable multi-ionized ion sources recently accomplished. Solid antihydrogen is produced from the trapped antihydrogen, and is stored electrostatically.

Experimental demonstration is in progress with respect to the recombination and the deceleration of the antihydrogen. The antihydrogen is simulated by ordinary matter argon in the experiment, since antiparticle can be regarded as particle with opposite charge without annihilation. Except for the differences in the mass and the energy level for the laser cooling, the argon in a metastable state has the advantage of being incorporated with laser diode, which has energy level related to near infrared range. The photograph of the experimental apparatus is shown in Fig.7. It consists of a plasma source which simulates a low energy antiproton beam, a recombination chamber, a cooling and trapping chamber.

## ANTIMATTER FACTORY AND INTERCEPTOR BASE

Necessary conditions for establishing the antimatter factory and the interceptor base comprises[1];

1: Sufficient solar power can be easily obtained,
2: Energy to launch the interceptor is small,
3: The earth's safety is assured at an accident.

The construction of the factory farther than the Mars orbit from the sun is not beneficial since the SPS (Solar Power Satellite) collects solar energy to produce antimatter. Lagrange points of L4 and L5 between the sun and the earth have the advantage of the minimum launching energy since they are the points of the gravitational equilibrium. It is also convenient from following standpoints to construct the plant on the back side of the SPS. First, the plant is cooled down as cryogenically as the space back ground temperature of 3 K because of isolation from the solar energy flux. Second, the high vacuum environment keeps the loss rate of the stored antimatter low because of low background density, a few particles per $cm^3$. Even if the disaster by the annihilation occurs in the plant, the irradiation from the antimatter factory remains as low as several times of natural level at the earth with 150 million km (1 a.u.) distance between the earth and the plant[1].

Next we estimate the antimatter fuel to be changed in the interceptor. Suppose the interceptor encounters the asteroid at 200 million km from the sun in 30 days after launch. The necessary delta-V is about 30 km/s when the interceptor is launched in the same direction as the earth evolution, and about 90 km/s in the opposite direction. As for the latter mission, it is impossible for chemical rockets because of large payload mass ratio of $10^9$. However, antimatter engine enables such a mission since the specific impulse can be chosen just like electric propulsion and the thrust density is as high as that of the chemical rocket. Forward[8] indicates that the payload mass ratio of the antimatter rocket do not exceed 5.

The mass of the vehicle, $m_v$ is assumed as 1 ton including an apparatus for the antimatter storage. Energy utilization efficiency $\varepsilon$ by the antimatter-matter reactions is assumed as 0.32. The necessary antimatter is given by[8]

$$m_a = \frac{0.39}{\varepsilon} \frac{\Delta V^2}{C^2} m_v \qquad (8)$$

$m_v$ : mass of vehicle included an apparatus for antimatter storage
$m_a$ : mass of antimatter propellant
$\varepsilon$ : energy utilization efficiency by annihilation
$\Delta V$ : mission delta-V
$c$ : speed of light

Substituting $\Delta V$=90km/s into Eq.(8), the required amount of the antimatter is 0.1g at most, which is negligible compared with that for the orbital modification of the asteroid. The mass of the reaction fluid is 4 ton. Consequently, the launching from the antimatter base located on the Lagrange points is possible. The earth protection system is schematically shown in Fig.8.

## CONCLUSION

First, necessity of the earth protection system against the meteorite collisions is investigated. The designed system consists of the VLBI radar tracking system, the antimatter plant and the interceptor to modify asteroid orbits. The radar tracking an asteroid by means of VLBI is feasible considering the state of arts of required technology. Some issues of millimeter wave remains open. An experimental simulation for the antimatter storage is introduced. It is desirable to construct the antimatter plant and the interceptor base combining the SPS at the Lagrange point. Destruction or orbital change of asteroids is concluded to be impossible without use of the annihilation energy.

## REFERENCES

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3. "Table of Scientific data" in Japanese, Maruzen Corp. Tokyo (1987)

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5. Wetherill,G.W., "Apollo Objects" Scientific American, March, 38-49 (1979)

6. Flash,L.L, et.al., "Electrothermal Propulsion of Spacecraft with Millimeter and Submillimeter Electromgnetic energy" J.Propulsion, Vol.4, No.4, 334-340 (1988)

7. Hirabayashi,H., "VLBI observation on large scale using satellites" in Japanese, Japanese Scientific Monthly, 1087-1093 (1989)

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9. Forward,R.L., "Antiproton Annihilation Propulsion" AFRPL TR-85-034, 129-131 (1985)

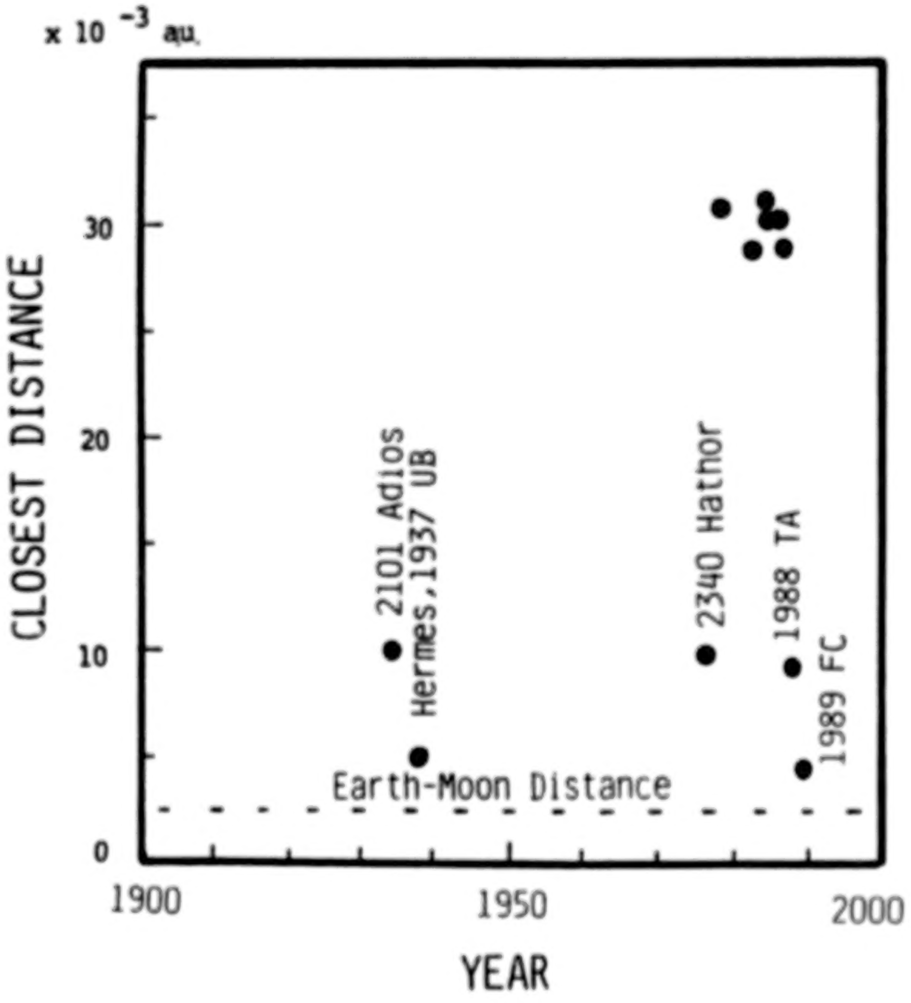

Figure 1. Near missed asteroids in 20th century

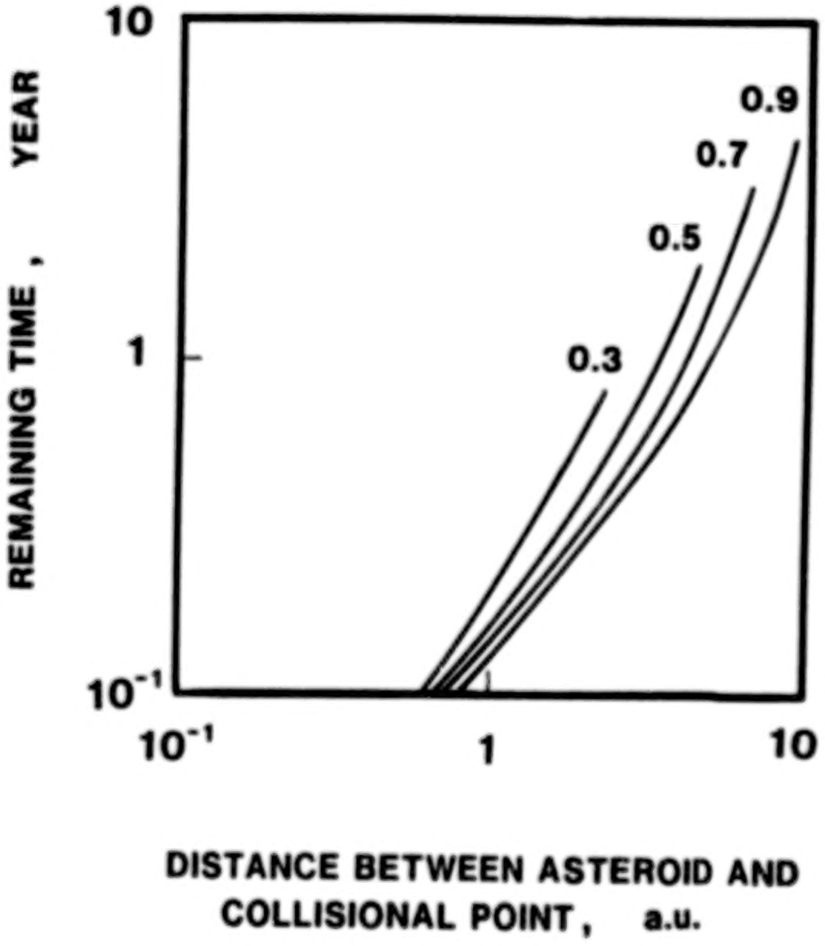

Figure 2. Remaining time before collision and perihelions. The asteroid orbit with eccentricity of 0.8 is assumed.

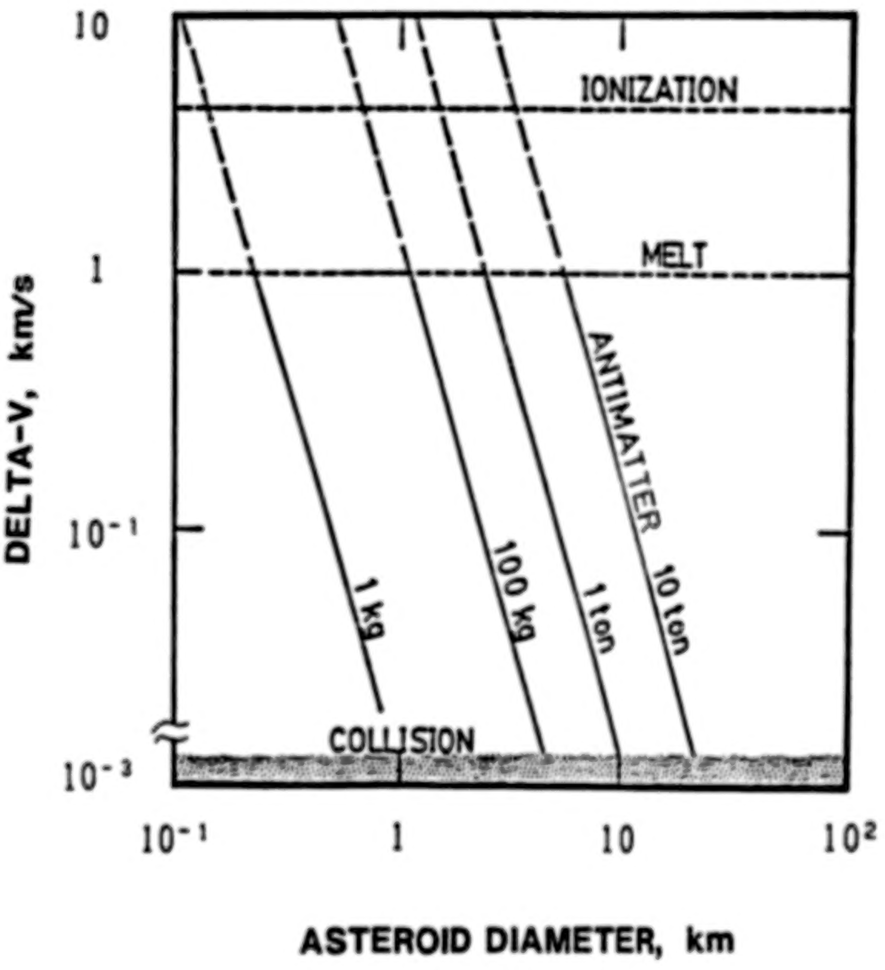

Figure 3. Delta-V and amount of antimatter. Asteroid orbit with eccentricity of 0.9 and periherion of $5 \times 10^7$ km is modified at $2 \times 10^8$ km distance from the sun.

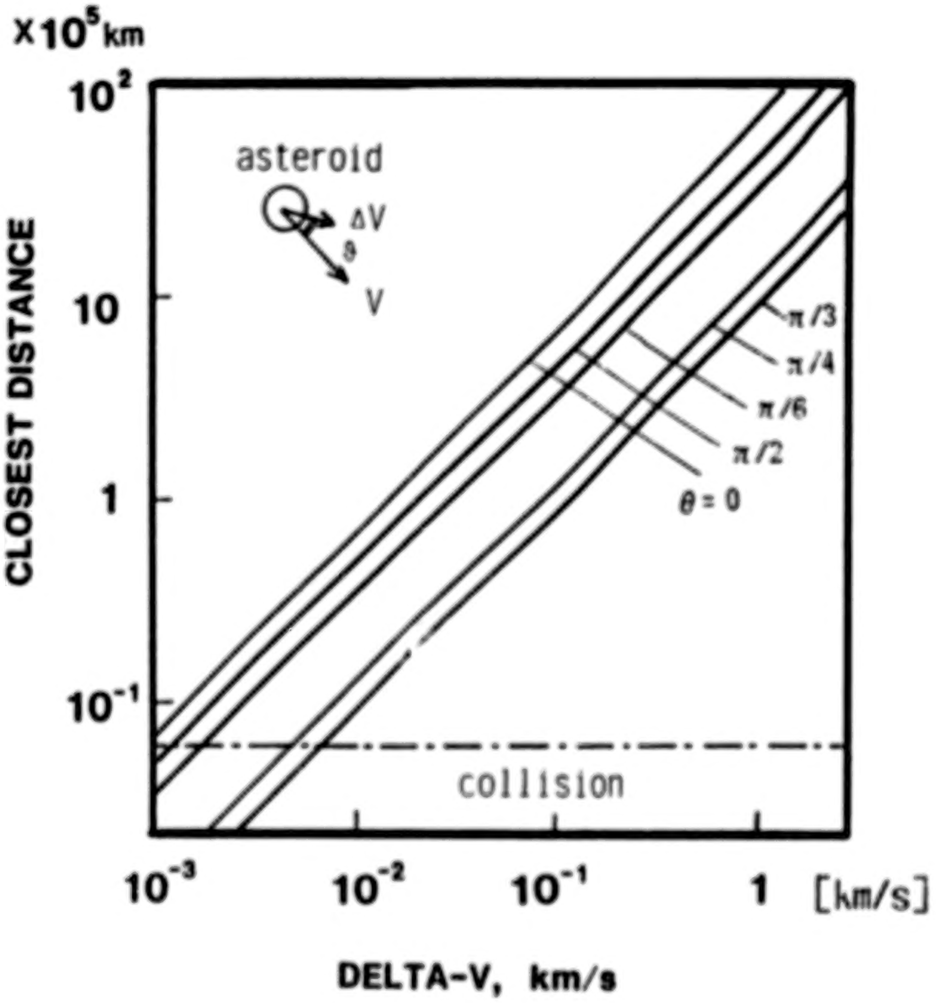

Figure 4. Closest distance from the earth and ΔV. Thrusting an asteroid is the parameter. Eccentricity of 0.9 and periherion of $5 \times 10^7$ km, and orbital modification at $2 \times 10^8$ km distance from the sun are assumed.

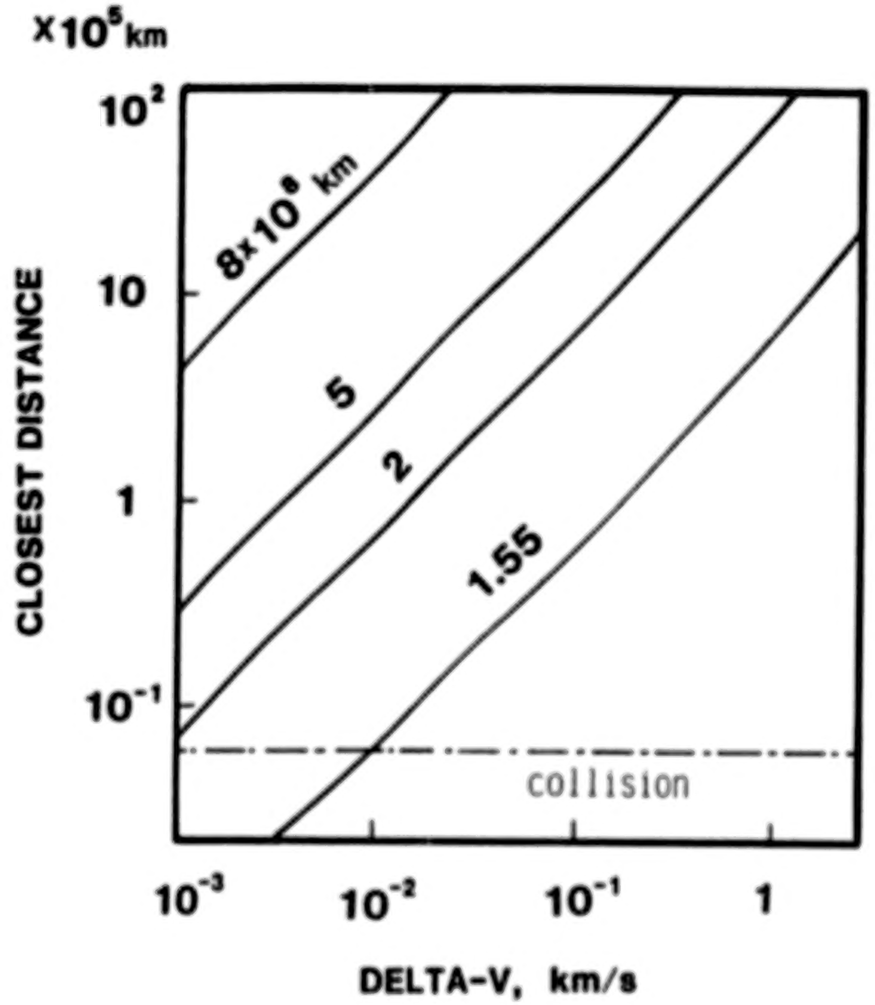

Figure 5. Closest distance from the earth and Delta-V. Position from the sun is the parameter. Asteroid orbit with eccentricity of 0.9 and perihelion of $5 \times 10^7$ km, the delta-V parallel to the proceeding direction are assumed.

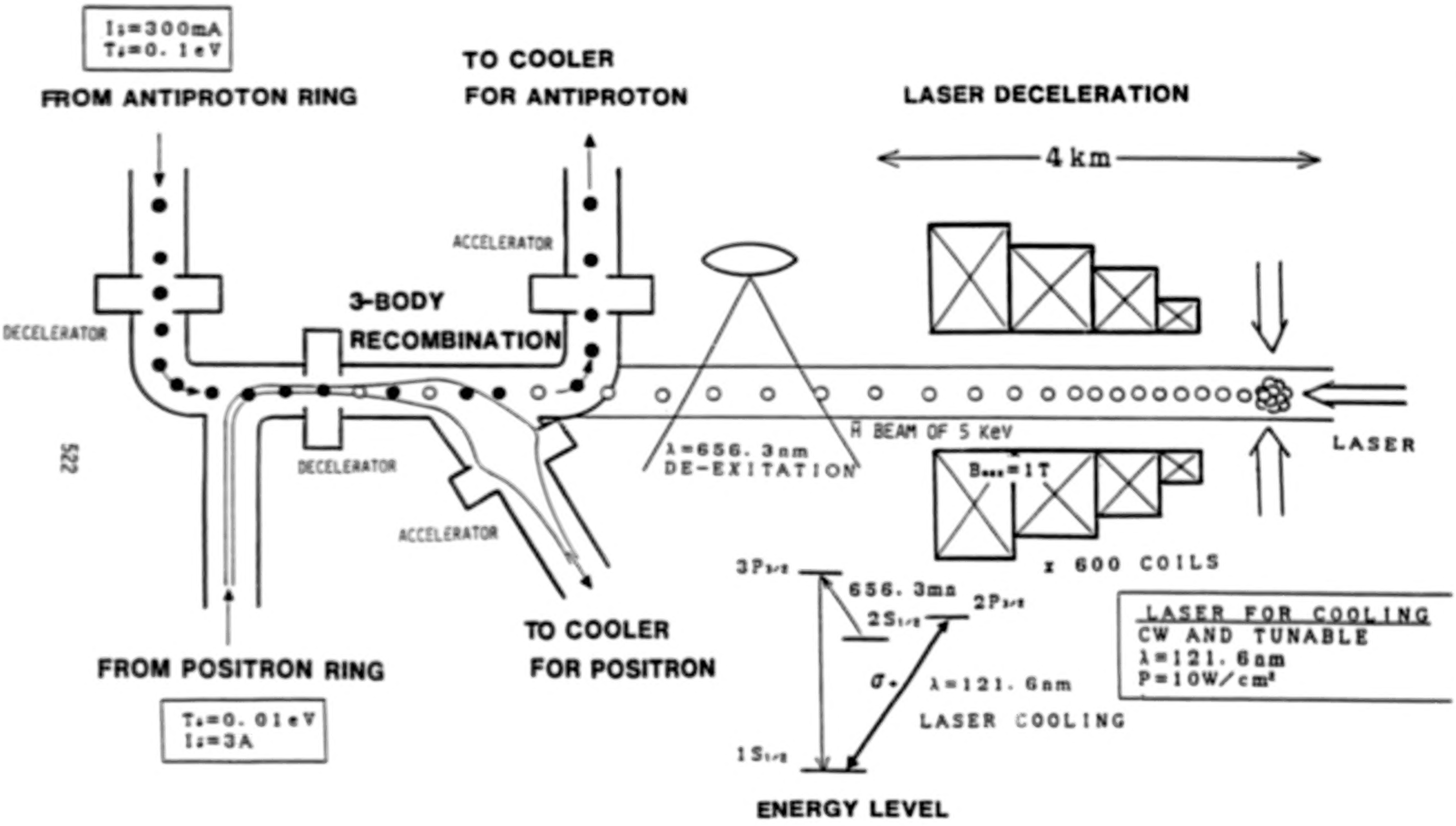
I₁=300mA
T₁=0. 1eV
FROM ANTIPROTON RING
TO COOLER
FOR ANTIPROTON
LASER DECELERATION
4 km
ACCELERATOR
DECELERATOR
3-BODY
RECOMBINATION
DECELERATOR
ACCELERATOR
λ=656. 3nm
DE-EXITATION
H̄ BEAM OF 5 KeV
Bmax=1T
LASER
x 600 COILS
3P3/2
656. 3mn
2S1/2
2P3/2
σ+
λ=121. 6nm
LASER COOLING
1S1/2
ENERGY LEVEL
LASER FOR COOLING
CW AND TUNABLE
λ=121. 6nm
P=10W/cm²
FROM POSITRON RING
TO COOLER
FOR POSITRON
T₂=0. 01eV
I₂=3A

Figure 6. Antihydrogen breeder

ORIGINAL PAGE
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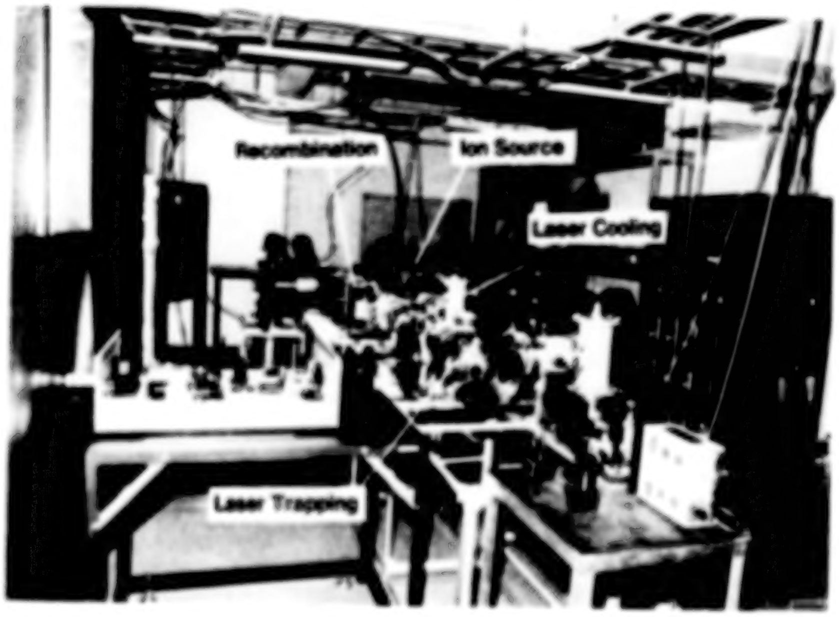
Recombination
Ion Source
Laser Cooling
Laser Trapping

Figure 7. Experimental apparatus

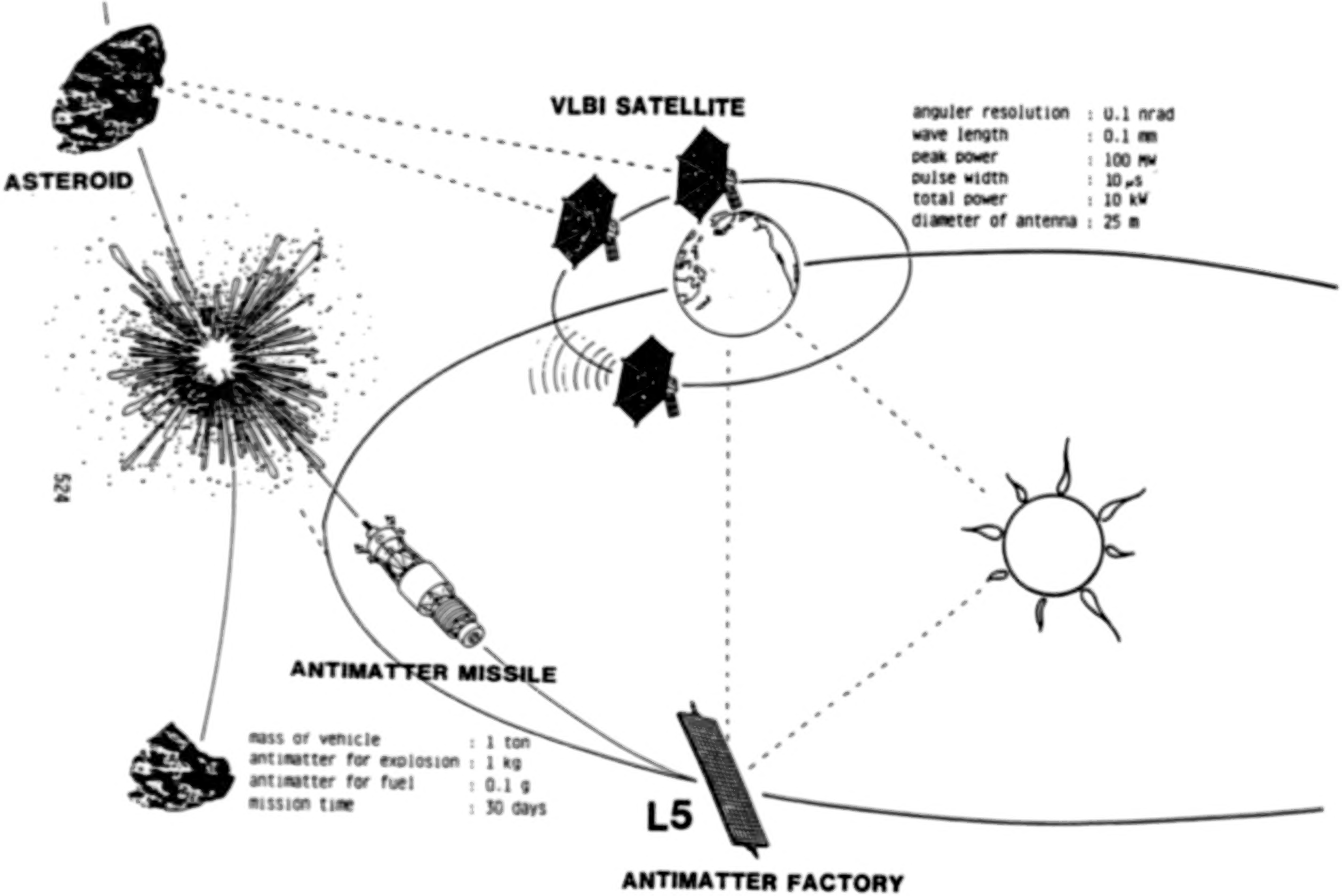
ASTEROID
VLBI SATELLITE
anguler resolution : 0.1 nrad
wave length : 0.1 mm
peak power : 100 MW
pulse width : 10 μs
total power : 10 kW
diameter of antenna : 25 m
ANTIMATTER MISSILE
mass of vehicle : 1 ton
antimatter for explosion : 1 kg
antimatter for fuel : 0.1 g
mission time : 30 days
L5
ANTIMATTER FACTORY

Figure 8. Earth protection system

*Journal of The British Interplanetary Society, Vol. 42, pp 567-569, 1989*

# SPACE-BASED SOLAR SHIELD TO OFFSET GREENHOUSE EFFECT

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**A thin glass shield built from lunar materials and located near the first Lagrange point of the Earth-Sun system could offset the greenhouse effects caused by the $CO_2$ buildup in the Earth's atmosphere.**

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## 1. INTRODUCTION

Terraforming the near planets of the solar system is a long, complex and ambitious undertaking lying well into Man's future. There are two major problems to be addressed viz global temperatures and atmospheric composition. A suggested method for the control of planetary temperatures is the use of space-based shields to modify the incident solar flux. Terraforming shields for planets such as Venus or Mars would, of necessity, be large, complex structures requiring vast amounts of lunar or asteroidal material and a well established space manufacturing and long-range transportation system. One possible stepping-stone to understanding and mastering the technologies and physical processes involved would be the construction of a shield to offset the greenhouse effect on our own planet, Earth. Such a project would be much smaller in size and scale and not require interplanetary capabilities.

Concern has been expressed worldwide for the changing composition of the Earth's atmosphere. In the twentieth century, increased atmospheric concentration of molecules such as $CO_2$, which absorb infrared radiation, has intensified the trapping of thermal radiation in the atmosphere now referred to as the greenhouse effect. As the concentration of these gases rises, the resulting temperature increases and other climatic effects become more serious. Because of the complexity of the many processes involved, there is considerable uncertainty regarding the rate of build-up of the greenhouse gases and the magnitude of the resulting climatic changes [1]. The time required for removal of these gases from the atmosphere by natural processes is also uncertain: current estimates are several centuries.

The uncertainties in the scale and duration of the climatic changes resulting from the greenhouse effects has led to calls for more research into these problems as well as restrictions on the generation of greenhouse gases. The most important restriction would be on the burning of hydrocarbons for power generation and transportation. Since the build-up of $CO_2$ in the atmosphere is cumulative and as there are no accepted technical solutions to the greenhouse effect, it may prove crucial to restrict the generation of these gases as soon as possible, perhaps in the 1990's. The existence of a possible technical solution could thus have a major short-term impact in influencing short term consumption restrictions, even if the solution could not be implemented until the next century.

A conceptually simple method for offsetting the greenhouse radiation trapping effects would be to decrease the solar heating by the use of a space-based solar shield [2]. Approximately 2% of the solar radiation reaching Earth must be blocked to offset the predicted greenhouse trapping expected in the next century [3]. The shield postulated would be 2000 km in diameter and located $1.5 \times 10^6$ km from Earth, near the first Lagrange point between the Earth and the Sun. A shield 10µ thick would weigh approximately $10^{11}$ kg and may cost from one to ten trillion dollars. It would be fabricated from lunar materials launched by a mass driver.

## 2. SHIELD ORBIT

The space shield must be placed in an orbit where it remains positioned between the Earth and the Sun. This point will be near the classical first Lagrange (L1) point. The location is determined by two requirements. Firstly the angular velocity of the shield around the Sun and of the Earth must be the same so that the shield remains in a line between the Earth and the Sun. Secondly, there must be an acceleration balance on the shield between the centripital acceleration from the orbit around the Sun, the gravitational accelerations of the Earth and the Sun and the proton acceleration on the shield. A photon thrust of zero would locate the shield exactly at the L1 point. Combining these requirements gives,

$$(\delta/R)^3 = (1/3)\left[\frac{m}{M} + \left(\frac{a}{a_o}\right)\left(\frac{\delta}{R}\right)^2\right](1 - \frac{\delta}{R}) + O\left(\frac{\delta}{R}\right)^3$$

$$a_o = GM/R^2$$

where m, M and G are the masses of the Earth and Sun and the gravitational constant. R and δ are the distances from the Earth to the Sun and from Earth to the shield. The photon acceleration of the shield is a. For zero photon thurst, one obtains the L1 point

$$\delta_o = 1.50 \times 10^6 \text{ km}$$

When there is a photon acceleration on the shield, the balance point becomes

$$\delta = \delta_o [1 + (1/3)(a/a_g)]$$

$$a_g \equiv Gm/\delta^3 = 0.0177 \text{ cm/sec}^2$$

where $a_E$ is the Earth's gravitational acceleration at the shield distance.

The shield orbit will be semi-stable as any small radial perturbation towards or away from the Sun will cause the shield to be pulled out of position. However, perturbations perpendicular to the Earth-Sun axis will be stable. Station-keeping at the L1 point requires constant adjustment to the orbit. This is why there will be no dust and natural satellites at the L1 point, as commonly found at the L5 or Trojan orbits. The accelerations required to hold this orbit are very small and well within the capabilities of the shield.

The shield is balanced only at the Earth-Sun axis. All other sections of the shield are drawn to the axis by a radial acceleration $\alpha_r$

$$\alpha_r = -G\left(\frac{m}{\delta^3} + \frac{M}{R^3}\right)r$$

where $r$ is the radial distance from the Earth-Sun axis. To be in an orbit around the Earth-Sun axis, the centripital acceleration must balance the radial acceleration

$$\Omega^2 r = -\alpha_r$$

$$\Omega = \sqrt{G}\,(\frac{m}{\delta^3} + \frac{M}{R^3})^{1/2} = 2.0\ cycles/year$$

where $\Omega$ is the angular velocity of a section of the shield. Since $\Omega$ is independent of $r$, the shield can rotate as a solid body and have each section be in orbit around the Earth-Sun axis. In this condition there would be no stresses in the shield. A faster rotation rate than is required for gravitational balance will create a radial stress in the shield, which may be desirable to help maintain the shield as a flat disc.

The disc rotation will, unfortunatly, act as a gyroscope which keeps the disc orientated with its axis pointed in one direction. Since the disc axis must always point toward the Sun, a torque must be applied to the disc by a control system to cause the disc to precess at one cycle per year. It is not clear if this control system is simpler than using solar sails at the perimeter of the disc to supply a radial tension to balance the radial gravitational acceleration.

## 3. PHOTON THRUST OF SHIELD

When a photon is absorbed or emitted, its momentum is transferred to the shield. If the photon is reflected, the momentum transferred is twice the photon momentum. The solar pressure from incident radiation on the shield is then

$$P = (F/c)\,[\alpha\,(1 + \frac{\epsilon_S - \epsilon_E}{\epsilon_S + \epsilon_E}) + 2r]$$

where $\alpha$ and $r$ are the shield absorptivity and reflectivity for sunlight, $\epsilon_S$ and $\epsilon_E$ are the infrared emissivities of the shield on the Sun and Earth sides. For a shield of thickness t and density $\rho$; the acceleration, a, of the shield is then

$$a = P/\rho t$$

Shield acceleration may be controlled by the optical design. If the shield is opaque, then the Sun side should have a low reflectivity (high absorptivity) for the solar spectrum. The photon thrust from radiated infrared energy can be used to offset the thrust from absorption. The infrared emissivity should be minimised on the Sun side and maximised on the Earth side. An ideal opaque shield would scatter the Earth bound solar energy into diffuse infrared energy.

The shield may also be transparent and simply scatter the visible photons away from the Earth. The required scattering angle, $\Theta$, is 8.5mr or one-half degree. A glass shield may act as a prism to deflect Sunlight away from the planet in accordance with Snell's law. The light beams are deflected as they enter and exit the thin glass shield. If both surfaces of the glass are parallel the deflections cancel and the light beams continue in their original direction. If the two surfaces are at an angle $\beta$ with respect to each other, the glass then acts as a prism. For the near-normal incidence configuration used to minimize shield reflection, the deflection angle, $\Theta$, is equal to:

$$\Theta = \beta\,(n_g - n_{vac}) = 0.5\,\beta$$

where

n = index of refraction = 1.0 for vacuum

= 1.5 for most glasses

The deflection angle does not depend to first order on the angle of incidence of the light on the shield so the orientation of the shield would not have to be closely controlled.

The shield may have a pattern of shallow parallel grooves on one side and be flat on the other. If the distance from peak to valley is 200 microns, then the required change in thickness is 3.4 microns. If the average shield thickness is 10 microns, this may be an acceptable change in thickness. If a thinner shield is desired the parallel grooves must then be closer together.

A transparent shield would reflect some of the light at both the entering and exiting surfaces. The amount reflected could be minimized by applying complex layered coatings to the glass but the relatively small benefit incurred may not justify the cost. The reflectivity would be 0.04 for an uncoated surface.

The photon acceleration for a 10μ transparent shield made of glass with a density of 2.5 is

$$a = 0.0030\text{cm/sec}^2$$

The displaced orbit position is

$$\delta = \delta_o\,(1.0565) = 1.58 \times 10^6\ \text{km}$$

## 4. SHIELD SIZE AND EFFECTIVE BLOCKAGE

The effective blockage by the shield is given in Fig. 1 as a function of the shield diameter. For a shield diameter less than 1200 km, the whole projection of the shield area on to the Sun's surface would lie within the Sun's disc when viewed from any point on the Earth's surface. The solar blockage would, therefore, be uniform over the Earth's surface. For a shield diameter of 2000 km, the projected shield disc falls partially off the Sun's disc only for Earth locations on the outer 6% rim of the projected Earth disc, i.e. of the standard circle map showing one side of the Earth. Except for arctic regions, these will be the regions near sunrise and sunset. Thus, the shading averaged over the entire day will be almost constant across the Earth with the arctic regions receiving slightly less shading. This condition should avoid some of the potential political problems associated with having some sections of the Earth shielded more than others.

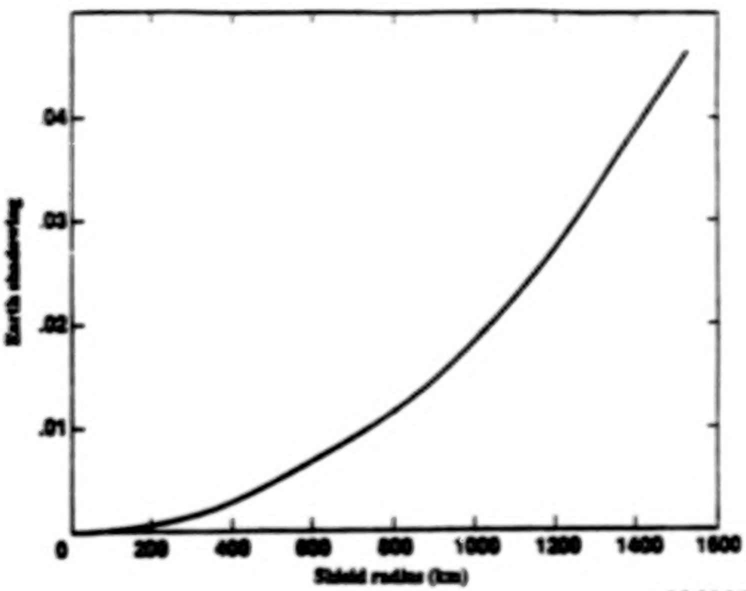

Fig. 1 Effective shadowing of the Earth as a function of the solar shield radius.

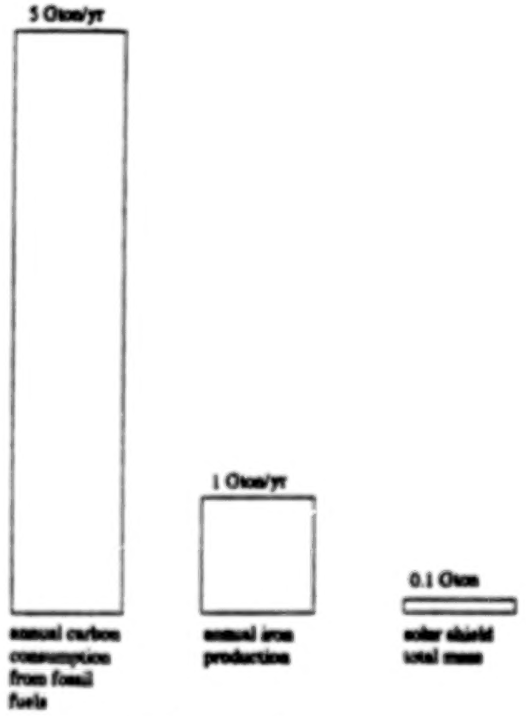

Fig. 2 Comparison of materials consumption

## 5. SHIELD DESIGN

The shield probably would be made from lunar materials. Lunar soil can be formed into glass for either a transparent or opaque shield [4]. The properties of glasses made from various lunar soils must first be determined. Most glasses will be sufficiently transparent in 10μ thicknesses. For an opaque shield, an appropriate coating material would have to be found for the glass substrate. Iron may be obtained from lunar soil with only moderate effort but is probably too dense to use as the base material for an opaque shield. Any material used must be capable of maintaining its properties for centuries in the presence of radiation from solar wind, solar UV radiation and cosmic rays.

The shield material would be processed into glass ingots then drawn into thin sheets. Glass fabrication techniques must be investigated to determine if 10μ is a viable thickness. Commercial plastic wrap is 13μ thick and aluminum foil is typically 13 to 25μ. Designs for advanced solar sails have been proposed using 2μ sail materials [5]. The glass may be launched to the shield location by a mass driver. A number of studies [6] have indicated that mass drivers are feasible and economical for launching unmanned payloads from the lunar surface. If the glass sheet is sufficiently flexible it may be formed into sheet on the lunar surface and launched in rolls.

There are many other design problems such as the fabrication of glass sheet, the impact of lunar perturbations and the required transfer orbits from the Moon to the shield which should be investigated. The shield structural support, the positional control system, the infrastructure to build and maintain the shield and the control of electrostatic charge are some of the major undefined systems.

The scale and costs of this project would be enormous but the economic impact of the greenhouse effect may be much larger. The scale of this project is not beyond the scale of man's activities on Earth (Fig. 2) but it is unclear whether such a massive project could be accomplished in space.

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* This work was performed under the auspices of the US Department of Energy by Lawrence Livermore National Laboratory under contract No. W-7405-Eng-48.

# Romans to Mars

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"The path made by a Leader is tread on sand;
his track is seen for others to follow
only as one footstep follows another.
For if he stands still, the trail is erased;
its footprints washed away by the changing tide. "
--- Domitius Lucullan, 195 AD

We are standing on a crumbling stone abutment that overlooks the port of Ostia on the mouth of the river Tiber, gateway to Imperial Rome. On late summer afternoons like this one, the breeze blows onshore and carries with it the pungent aromas of the Mediterranean and the shriek of gulls that wheel and dip in wide circles over the harbor. The harbor is crowded with vessels of every kind, from huge war galleys with multiple banks of oars that stroke the water in confident, well-coordinated, wingbeat-like sweeps, to lighters and tenders scurrying among the larger ships like beetles. And there are sailing vessels, too -- mostly merchantmen tied up at docks or riding at anchor near the harbor mouth, waiting for evening when the wind will shift from onshore to the offshore breeze that will carry them out to sea.

A few paces from us stands a very distinguished looking figure staring out to sea, his arms folded behind him at parade rest. His cape is scarlet, trimmed with ermine. He wears the silver breastplate of a proconsul. His retinue is huddled some distance behind him muttering to themselves and casting worried glances in his direction. Their master is scowling; his jaw is set with hard lines around his mouth. Lucius Marcellus Varsovian is not a happy man. He has driven his chariot hard all the way back from the capitol after being handed one of the few defeats in his career. To compound his frustration, insult has been added to injury -- the westbound courier already cleared the harbor earlier in the day, and so he is unable to obtain passage home on a military galley. The first leg of the long voyage back to Spain will have to be on a merchant vessel, a sailing ship. He is not accustomed to having to wait for the wind to change, and he is furious.

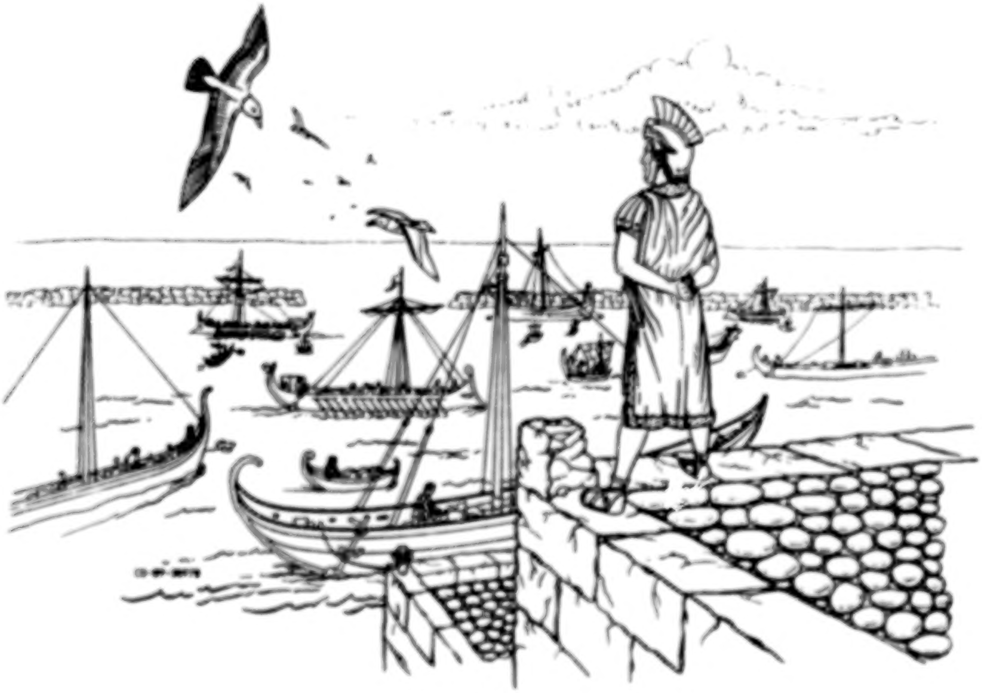

FIGURE 1.

The senate failed to back him again. Too bad -- his proposal was bold and imaginative. It could have resulted in a fresh infusion of riches for the Empire. Possibly, it could have restored Rome's declining fortunes and brought a new sense of purpose, ending the petty squabbling now going on. If China could be reached by going west across the ocean, then the wealth of the Orient could flow to Rome, not in a trickle on the backs of a few pack animals, but by the shipload. How could they be so shortsighted? All he had asked for was some men and a few ships. . . . .

At this point you might be tempted to characterize Lucius Marcellus as a visionary, a man ahead of his time. That would be a mistake. The ancients ( table I ) knew the world was round ever since Aristotle; from the calculations of Eratosthanes and Hipparchus, they had a pretty good idea of its size. By the second century A.D. they were making geometrically accurate maps by using astronomical observations to locate position.

And Lucius Marcellus Varsovian is not a dreamer, interested in discovery or commerce. He wants to take his legions to China and plunder their cities!

The riches of the Orient have tantalized the Romans for a long time. Their knowledge of China is more tangible than just fables because, in the third century, there is regular contact and trade. In Rome's heyday, the emperor Marcus Aurelius had maintained emissaries at the Han court in Peking. Their reports told of large cities, linked by a network of good roads, heavily populated, but not heavily fortified. The richest cities were furthest east, on a wide coastal plain that extended eastward to an ocean. The reports also indicated that the Chinese empire was more a loose confederation of fiefdoms than an empire. Although every warlord had an army, there was no national army nor the political cohesiveness to sustain one. It had not been necessary because they were so well isolated.

Distance and geography kept the two empires apart. The known route to China, traveled by the caravans, is a tortuous overland journey which permits a limited exchange of communication, trade goods, and culture, but so far has prevented the more direct form of cultural intercourse that Lucius Marcellus Varsovian is contemplating. Taking armies on a long march over the caravan route would be out of the question. Varsovian knew that all too well. As a young centurion in Atticus' disasterous Afghan campaign, he was one of the few who had made it back across the Khyber Pass alive.

## THE ANCIENTS

| | |
|---|---|
| **Pythagoras** | (550 BC) First to offer scientific arguments that the world was round. |
| **Aristotle** | (350 BC) Proved convincingly, and to everyone's satisfaction, that the world was not only round but also the true center of the universe. |
| **Eratosthanes** | (240 BC) First accurate estimate of earth's size, based on solar observations. |
| **Hipparchus** | (120 BC) Discovered earth's precession; published first accurate astronomical tables predicting eclipses, positions of stars and planets throughout the year. First to systematically use trigonometric principles, he refined, based on astronomical observations, earlier estimates of the sizes, distances to sun, earth, and moon. |
| **Claudius Ptolomy** | (140 AD) Last and most authoritative of the classical Greek astronomers. Compiled and codified the body of Greek thought (including Hipparchus) on astronomy, earth sciences in thirteen volumes ("The Syntaxis"). Published first dimensionally accurate (conical projection) map of the known world; but underestimated size of earth by 30%. |

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TABLE 1.

## PTOLEMY'S MAP OF THE WORLD

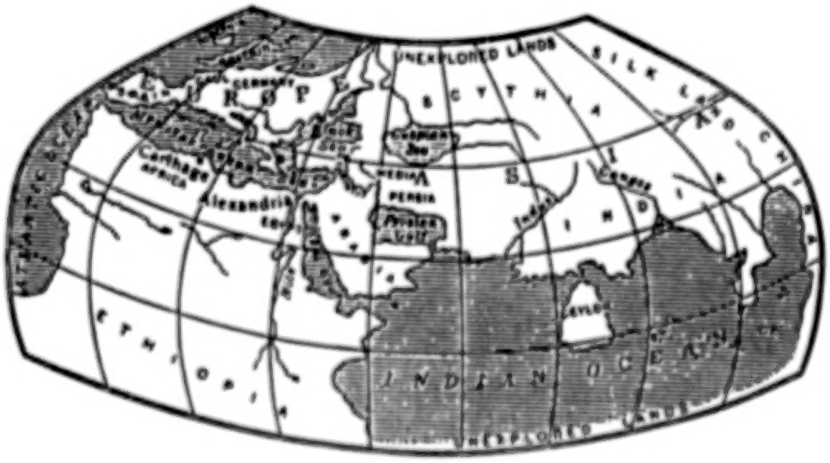
UNEXPLORED LANDS
SILK LANDS
SCYTHIA
EUROPE
GERMANY
Carthage
AFRICA
Alexandria
Egypt
Black Sea
Caspian Sea
MEDIA
PERSIA
Persian Gulf
ARABIA
ASIA
Ganges
Indus
INDIA
CHINA
CEYLON
INDIAN OCEAN
ETHIOPIA
ATLANTIC OCEAN
UNEXPLORED LANDS

FIGURE 2.

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But a sea route would change everything. Lucius Marcellus was mainly a land soldier, but not entirely unappreciative of sea power. He understood the surprise value of an amphibious assault, having used this tactic successfully to crush the Berber rebellion in Mauretania. Ferrying his troops along the coast just out of sight of land until nightfall, he had come ashore at dawn and driven swiftly inland before they could rally their tribes, cutting off their main encampment and capturing their chief, who was subsequently drawn and quartered.

Wounded in that campaign, he was sent to Alexandria to recuperate. It was on one of his frequent visits to the Great Library there that he had encountered the astronomer Claudius Ptolomy's Map of the World ( fig. 2 ), the first conical projection based on astronomical observations and the most accurate map of its time. Intrigued by the map, he studied Ptolomy's Syntaxis,which explained how the map had been made, how astronomy could tell you the size of the world, and where you were located on it. The map showed the easternmost part of China, where the richest cities were, to be located furthest away from the west coast of Spain, where he had been born. But, if that map were wrapped around a globe according to the method explained in the book ( fig. 3 ), the east coast of China and the west coast of Spain were actually facing each other, separated only by an undetermined stretch of ocean.

According to the calculations and depending on the accuracy of the astronomical measurements, the distance across that stretch of ocean was somewhere between 1500 and 2000 leagues. Lucius Marcellus couldn't fully understand all the explanations which led to this result, but he was quick to grasp its military significance -- if the ocean could be crossed, the richest part of China might be directly accessible to his armies.

Would it be possible to cross the Great Ocean? The 1500 to 2000 leagues of open sea was certainly a formidable distance. But it was not an insurmountable distance. Roughly equal to the Empire's dimensions from western Spain to eastern Persia, it was in fact less than the sea distance routinely navigated from Asia Minor to Britain. What if he could muster his troops at the port of Gades ( now Cadiz ) on the west coast of Spain, load them into ships, and head directly west? The seas would be calm in summertime. They could follow the setting sun, or the lodestone. An accurate landfall wouldn't be needed; it would be hard to miss the China coast.

Compared to the perils of an overland march, a sea voyage would be short and uneventful. After a few weeks cooped up in their ships, his troops would be spoiling for a fight, eager to attack. A seaborne invasion would not be expected. From an eastern beachhead, his invasion force could easily sweep across the wide coastal plain unopposed; the cities would be easy prey for his seasoned legions and their siege engines. Even if the Chinese emperor were able to rally his minions and prepare a counterattack, it would take time -- time to allow him an orderly retreat back to his ships, laden with the spoils of war.

He could return to Rome in triumph, perhaps become Emperor. Lucius Marcellus had a rough understanding of the relationships between military strength and economic growth. By the third century, Rome had already absorbed the western world; there was nothing else nearby left to conquer. The army was not engaged in conquest any more but was, instead, relegated to maintaining order on the frontier, collecting taxes and putting down rebellions. That was no challenge. On the other hand, the fabled cities of the Orient would provide a worthy target for his legions. Why waste well-disciplined troops skirmishing with barbarians when their skills could be used so much more profitably against civilized societies? Why burn down some squalid frontier village when, to the east, there were magnificent cities waiting to be sacked? What satisfaction was there in ravishing unwashed savages in animal skins when, to the east, there were palaces to be looted -- with voluptuous princesses, succulent concubines draped in silk and jewels, their bodies bathed in perfumes and spices . . .

Before his armies could embark, however, he would have to know more about where they were going. Detailed information was needed. Exactly how far was it to the China coast? Where were the best places to land an army? Where could they land unopposed, or better yet, undetected? Before invading by sea, the coastline would have to be positively located and explored. His calculations indicated it should lie 1500 to 2000 leagues west of Spain, but that was only an estimate. Even though he believed an invasion was feasible, he couldn't commit hundreds of ships and thousands of men to a one-way voyage into the sunset without tangible proof that the Great Western Ocean could be crossed, and that China indeed lay on the other side.

The first mission of the campaign would therefore have to be a voyage of exploration -- or, in terms more familiar to Lucius Marcellus, a reconnaissance.

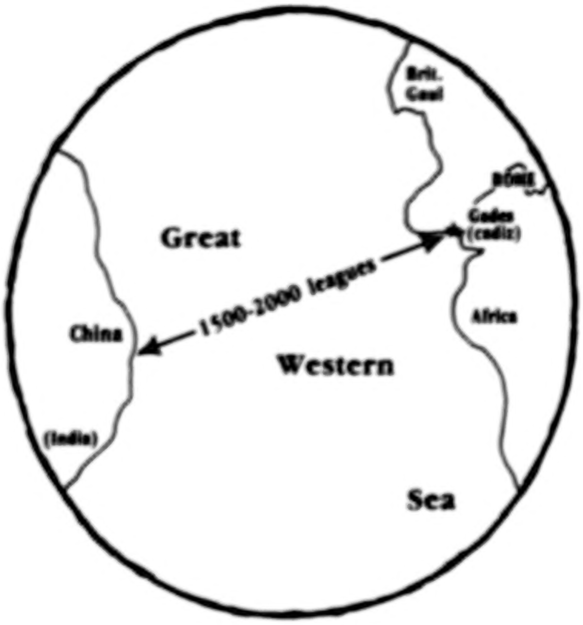
Brit.
Gaul
ROME
Gades
(cadiz)
Great
1500-2000 leagues
China
Africa
Western
(India)
Sea

FIGURE 3.

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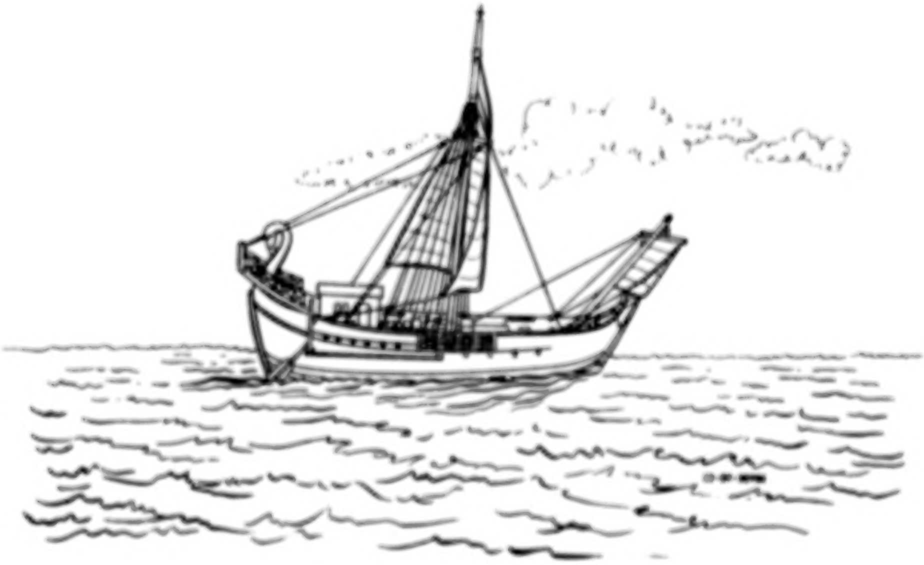


FIGURE 4.

To cross the Great Ocean, he would need a ship with extraordinary range. What kind of ship could go that distance? A sailing vessel would seem to be the logical choice, since it has the most economical form of propulsion. By Varsovian's time, the sturdy little merchantmen that carried Rome's trade to the four corners of the Empire were routinely sailed beyond the Mediterranean up and down the west coasts of Europe and Africa. Beamy and bluff-bowed, their trademark was a single, loose-footed, square mainsail, often augmented by a spiritsail carried well forward for added stability when the ship ran downwind in rough seas. Their Greek and Phonecian design heritage reflected sailing conditions on the Mediterranean -- which includes generally pleasant, but often unpredictable weather. They were unable to hold a course more than a few points away from the wind, but their shallow draft enabled them to be sailed right up to the shore. Most were light enough to be dragged onto the beach by their crew. Hugging the coastline and making forward progress as long as the wind was behind them, they could steer for shore whenever the wind turned against them, and wait there until better conditions prevailed.

That strategy, however effective along the well-settled Mediterranean, would not work offshore. The ungainly little Roman vessels were adequate for coastal navigation, but, unable to make headway against the wind, they would not be suited for travel on uncharted waters. If the prevailing winds were easterly, they would never make it to China. If westerly, they would never make it back.

A more reliable form of propulsion would be required. Oars, with the built-in reliability of hundreds of rowers straining their backs in unison, would be the propulsion system of choice. There was no larger, faster, or more reliable vessel ever propelled by oars than the Roman galley.

But the reliablility of those straining backs comes at a price. The men who pull the oars must be fed and watered. This severely limits the amount of time a galley can stay at sea. For short voyages it is not be a problem. For longer voyages, however, large amounts of food and water must be carried on board. Space is limited on any ship, but a galley is more restricted because such a large fraction of the available space is taken up by its crew. It is the amount of supplies that can be fitted into the remaining space, together with the rate at which they are consumed, which determines how how many days at sea the ship can operate.

Compared with the nonstop distances commonly traveled by military vessels, Varsovian's requirement was unprecedented. A trireme, with its slender hull crammed with rowers for high performance, could achieve perhaps three days at 11 knots. The quinquireme, a much larger warship, could last about a week, but only at a sustained speed of about seven knots. That would be enough to cross the Mediterranean from Italy to North Africa, but not enough for a voyage beyond the Pillars of Hercules.

To row across the ocean, Varsovian would need a ship that maximized the range he could travel before his onboard supplies were exhausted. The solution was to find a galley with moderate crew size and extra cargo capacity, and a cruising speed that took a of minimum effort to sustain. Fortunately, his experience suggested a compromise -- the common troop galley ( fig. 5 ) which had served him so well in the Mauretanian campaign. A medium size vessel of about 70 tons displacement, there were hundreds of them in service throughout the empire, used to ferry the army to wherever there was trouble. Designed to carry a cohort of 100 fully armed troops and their officers, the ship was propelled by another 100 men pulling on the oars. It also carried a lugsail rig for periods of favorable wind. A good compromise -- the sails provided economy, the oars provided reliability. With moderate effort, a galley of this design could be rowed continuously at 4 to 5 knots, enough to cover 30 to 40 leagues per day.

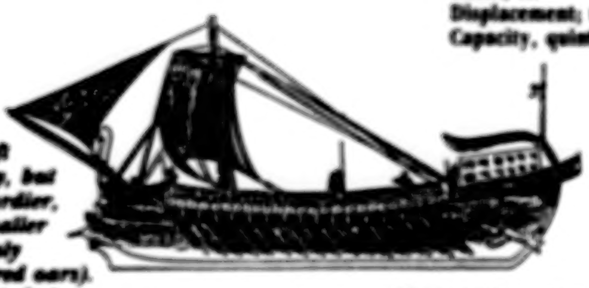
This galley was not as fast as a warship, but it could stay at sea for a much longer period. With its wider hull and smaller crew, it normally carried enough food and fresh water for voyages of about 10 days. Varsovian could modify this vessel by removing the troop accomodations and putting in more supplies, essentially replacing the 100 fully armed troops with provisions for his rowers. Based on the weight and volume margins allowed by this modification, he could lay in enough extra provisions for an estimated 54 days of travel, a little less than two months at sea.

Varsovian calculated that if the ship could average 37 leagues per day ( assuming assistance from favorable winds no more than half the time ), 54 days of continuous travel would cover 2160 leagues. That would be enough to cross the Great Ocean, if Ptolomy was right.

However, crossing the Great Ocean nonstop would still not be enough range to accomplish the mission. If he got there -- if he really found the coast -- he wouldn't be able to count on a friendly port or fresh provisions. He

**ROMAN TROOP GALLEY**

Length overall, feet: CXL
Beam, ft: XXI
Displacement; tons: CXX
Capacity, quintals: MCM



*Not a swift war galley, but wider, sturdier, with a smaller cohort (only one hundred oars). Not as fast, but able to stay at sea much longer.*

**SPEED**
nominal cruise 4-5 kt.
maximum sustained 7 kt.
dash 8 kt.

**PROPULSION:** sail or oars depending on wind direction (assume half time each mode).

**ENDURANCE**
continuous cruise 54 days

**COHORT,** officers: XII
oarsmen: XCVI

**RANGE**
nominal one way 2160 leagues (1 leagues = 3 mi.)
round trip 1080 " "

FIGURE 5.

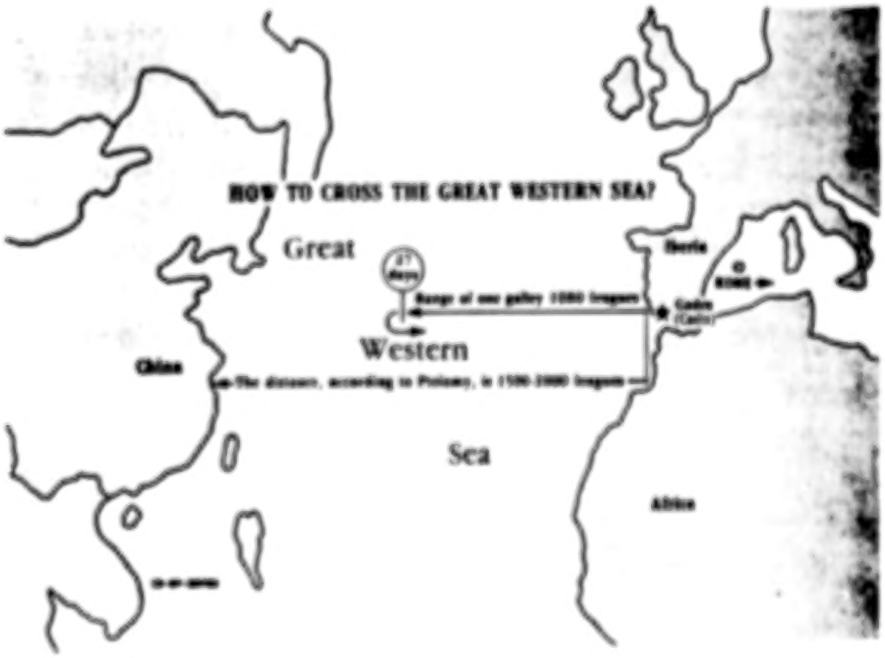

HOW TO CROSS THE GREAT WESTERN SEA?
Great
Western
Sea
27 days
Range of one galley 1080 leagues
The distance, according to Ptolemy, is 1500-2000 leagues
China
Iberia
Rome
Gades (Cadiz)
Africa

FIGURE 6.

might just have to turn around in empty ocean and head home. The 2000 plus leagues of range that he had managed to squeeze out of his troop galleys so far was only a one-way range. What he actually needed was 2000 + leagues of round trip range, more commonly known as the "distance to point of no return" where half the supplies are exhausted. Rowing westwards from Gades, his ship would reach its point of no return ( fig. 6 ) only 27 days into the voyage, a little over 1000 leagues. If he was willing to gamble with the expedition and keep going, a one-way voyage might possibly land them on the China coast, and, with a little bit of luck, they might find a secluded harbor where they could foray ashore for food and water. But the risks jeapordized the success of his mission. What if they never sighted land? That would be dissappointing but nonetheless valuable information. And how much worse would it be to make a successful landfall, only to be butchered on shore by the local cavalry while trying to hustle a few supplies. . . .

The only way his reconaissance mission could be successful was to ensure that they returned home with the information. ( And knowing you can return generally enhances morale! )

It would have to be a two-way voyage: westward across that distance to China, or at least as far as China should be, then eastward across that distance back to Gades. Varsovian had to find a way to stretch his range to twice the 2000 or so leagues that he had so far obtained, from ships that were already at their limits. It was a problem which would have caused a lesser man to give up.

But Varsovian managed to solve this problem also. He did it by organizing the mission in stages, augmenting the expedition with additional vessels that would replenish the other ships at carefully timed intervals.

A fleet of 32 galleys would be required. They would all leave the port of Gades together on the Ides of June and row westward ( fig. 7 ). Eighteen days later, however, after one third of the food and water had been exhausted, the fleet would be split into two groups. The ships would pair off with one another in midocean, and, within each pair, supplies would be transferred from one ship to the other. The ship receiving supplies would be fully reloaded, and would continue westward ( fig. 8 ). The donor vessel, with just enough inventory left for a return trip, would turn east and head home. The westbound ships would gain an additional 54 days of operating

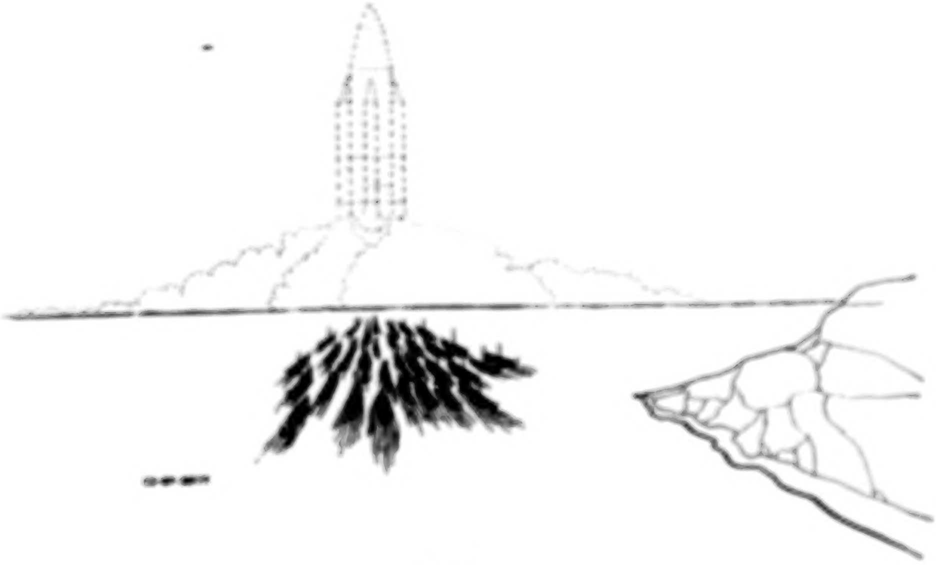


FIGURE 7.

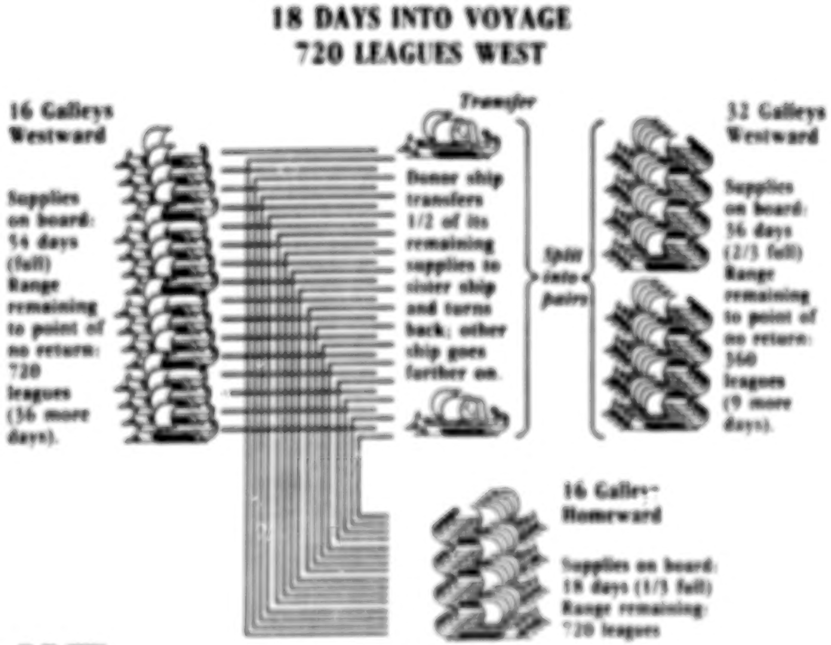

18 DAYS INTO VOYAGE
720 LEAGUES WEST
16 Galleys Westward
Supplies on board: 54 days (full) Range remaining to point of no return: 720 leagues (36 more days).
Transfer
Donor ship transfers 1/2 of its remaining supplies to sister ship and turns back; other ship goes further on.
Split into pairs
32 Galleys Westward
Supplies on board: 36 days (2/3 full) Range remaining to point of no return: 360 leagues (9 more days).
16 Galleys Homeward
Supplies on board: 18 days (1/3 full) Range remaining: 720 leagues

FIGURE 8.

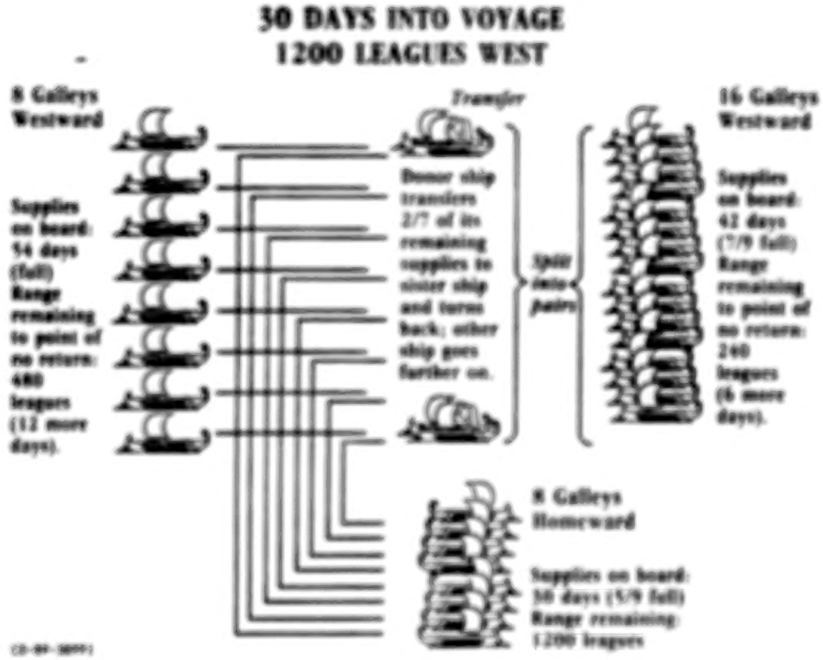

30 DAYS INTO VOYAGE
1200 LEAGUES WEST
8 Galleys Westward
Supplies on board: 54 days (full) Range remaining to point of no return: 480 leagues (12 more days).
Transfer
Donor ship transfers 2/7 of its remaining supplies to sister ship and turns back; other ship goes further on.
Split into pairs
16 Galleys Westward
Supplies on board: 42 days (7/9 full) Range remaining to point of no return: 240 leagues (6 more days).
8 Galleys Homeward
Supplies on board: 30 days (5/9 full) Range remaining: 1200 leagues

FIGURE 9.

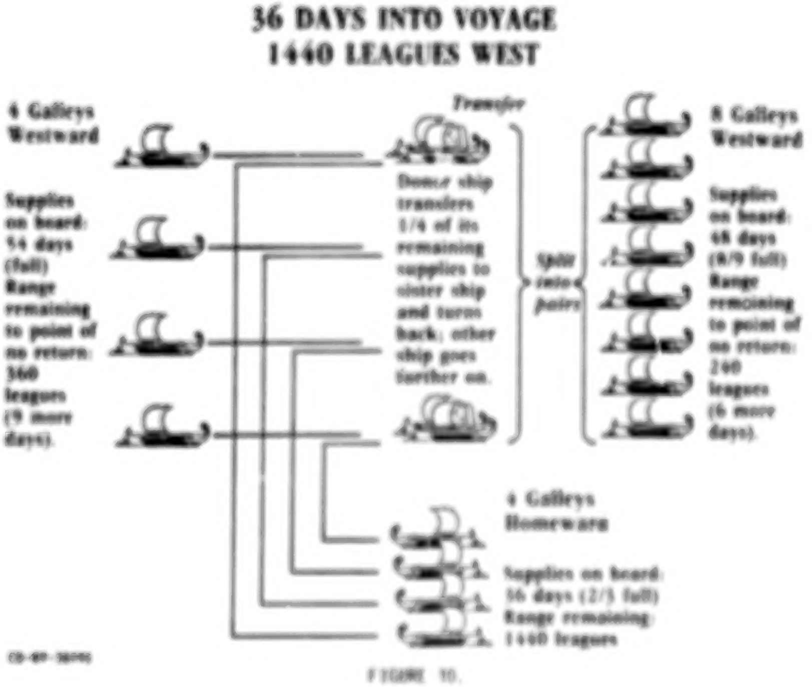

36 DAYS INTO VOYAGE
1440 LEAGUES WEST
4 Galleys Westward
Supplies on board: 54 days (full) Range remaining to point of no return: 360 leagues (9 more days).
Transfer
Donor ship transfers 1/4 of its remaining supplies to sister ship and turns back; other ship goes further on.
Split into pairs
8 Galleys Westward
Supplies on board: 48 days (8/9 full) Range remaining to point of no return: 240 leagues (6 more days).
4 Galleys Homeward
Supplies on board: 36 days (2/3 full) Range remaining: 1440 leagues

FIGURE 10.

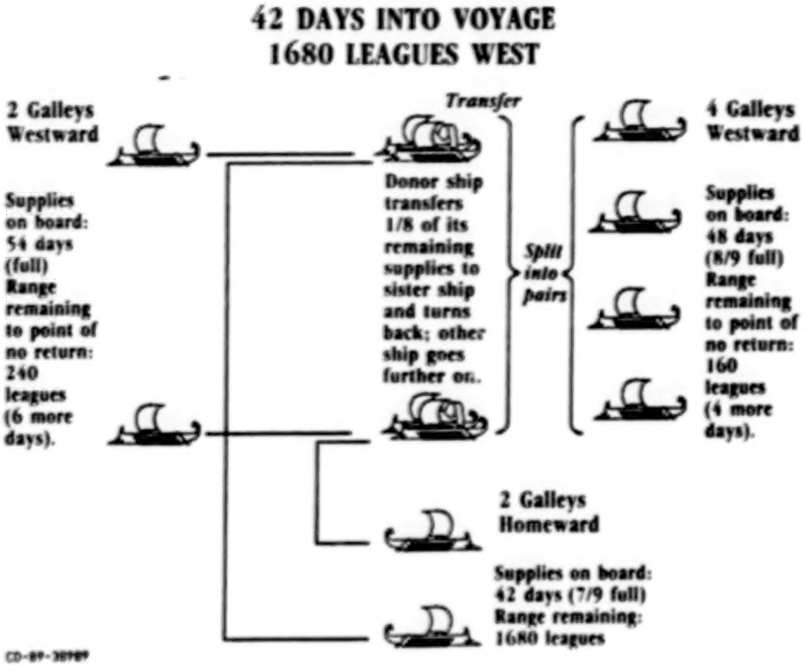

42 DAYS INTO VOYAGE
1680 LEAGUES WEST
2 Galleys Westward
Supplies on board: 54 days (full) Range remaining to point of no return: 240 leagues (6 more days).
Transfer
Donor ship transfers 1/8 of its remaining supplies to sister ship and turns back; other ship goes further on.
Split into pairs
4 Galleys Westward
Supplies on board: 48 days (8/9 full) Range remaining to point of no return: 160 leagues (4 more days).
2 Galleys Homeward
Supplies on board: 42 days (7/9 full) Range remaining: 1680 leagues

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FIGURE 11.

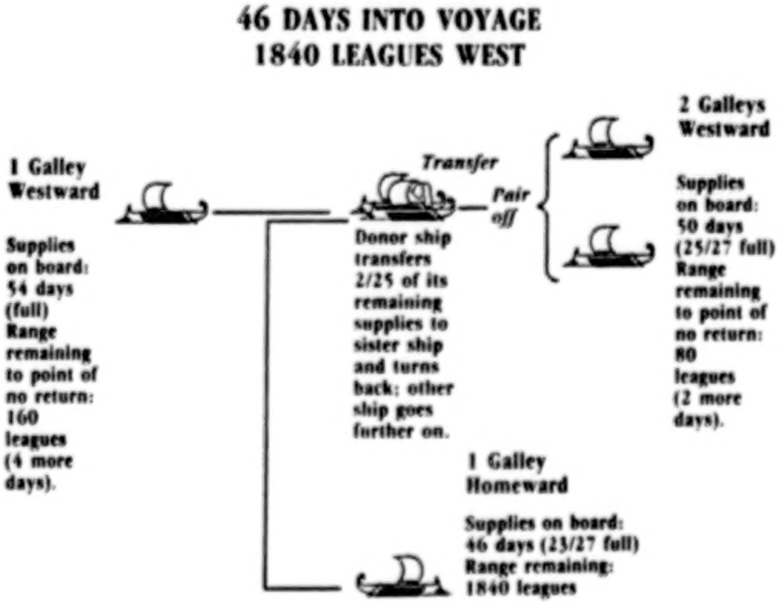
46 DAYS INTO VOYAGE
1840 LEAGUES WEST
1 Galley Westward
Supplies on board: 54 days (full) Range remaining to point of no return: 160 leagues (4 more days).
Transfer
Pair off
Donor ship transfers 2/25 of its remaining supplies to sister ship and turns back; other ship goes further on.
2 Galleys Westward
Supplies on board: 50 days (25/27 full) Range remaining to point of no return: 80 leagues (2 more days).
1 Galley Homeward
Supplies on board: 46 days (23/27 full) Range remaining: 1840 leagues

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FIGURE 12.

C-7

time beyond the 18 already used, thus stretching their round trip range an extra 360 leagues beyond the original point of no return.

Twelve days later, or 30 days into the voyage, this maneuver would take place again. Of the sixteen ships that had continued to row westward, eight of them would relinquish a fraction of their supplies to their sisters and turn home, leaving eight ships to continue the voyage ( fig. 9 ). Again, the westbound ships would be fully replenished; the eastbound ships would head back with exactly enough food and water for the return trip, since it was, in fact, the same amount they had consumed on the outbound leg. The expedition would gain an additional 240 leagues of range.

Six days later, 1440 leagues west of Gades, the fleet would divide itself once more ( fig. 10 ). Four galleys rowing westward, four galleys rowing back. Another range gain -- 120 extra leagues.

After another six days, or 42 days elapsed, if land had not been sighted yet, the remaining ships could pair off again, leaving two galleys to venture onward ( fig. 11 ). The expedition would have covered 1680 leagues of ocean at this point, a round trip distance further than any individual galley could have gone, and close to the estimated distance from Spain to China. They would have another four days to push westward before dividing up the fleet again.

If land was not yet sighted after 46 days from home port, 1840 leagues beyond the Pillars of Hercules, there was still an additional four-day margin. The expedition could split itself up one more time ( fig. 12 ), stock up the last galley, and send it west for another 160 leagues. The final round trip range that resulted would be 2000 leagues. As always, every vessel would have just enough supplies left to make a safe passage home.

Varsovian's reconnaissance would be the first known use of staging to boost the range. Figure 13 summarizes the mission stages, their separation points, and the fractional gain in operating time and range.

This approach had major strengths. It allowed Varsovian to navigate a round trip distance which would have otherwise been impossible, obtaining the endurance he needed from ships whose individual capabilities were limited. Not only did his plan extend the range to almost double that of any individual ship, it guaranteed that every ship in the fleet could return. With portions of his fleet dropping out and returning home as the various mission stages were expended, news of the expedition's progress could be reported home at regular intervals. At each staging point he could choose which ships should continue, thereby ensuring that only the soundest ships and strongest crews continue the voyage. Failed or weakened elements could be removed to the rear; these would not have to make the return voyage alone.

Best of all, the plan allowed for contingencies. There was plenty of margin for error. If his range estimate was wrong, if the China coastline proved inhospitable, or if he was unable to make landfall for any reason -- he could still complete the mission. The plan not only extended his range, it did so in a way which maximized the probability of success while minimizing the risks to his ships and crew.

Unfortunately, Varsovian's plan could not anticipate the most difficult phase of the mission where the risks were greatest: the presentation of his proposal to the Emperor and assembled senate ( fig. 14 ). The review began encouragingly enough; many senators supported his plan. In principle no one was opposed to a China campaign -- everyone agreed that, if a sea-borne invasion was to be considered, a reconnaissance mission would be the next logical step. As for feasibility of the voyage, no one doubted it; the analytical results Lucius Marcellus presented were far too convincing. The Emperor had listened to his plan carefully, had liked it, and had endorsed it. Varsovian's proposal would provide a practical demonstration of something that already appeared to be scientifically sound. His plan was reasonable, the risks were modest. Most important, the mission could be accomplished without any new technology development.

But the mission was expensive. In fact, the costs were enormous. Instead of a simple scouting foray, this expedition ( summarized by stages in table II ) had the dimensions of a full-scale military campaign -- 32 troop ships and 4000 men, just to see if China was on the other side of the ocean!

The cost of modifying 32 troop galleys alone was no small amount of money. Of course, shipyards from Venicia to Tarantum would be busy for months. Because of the amount of business the expedition represented, Varsovian obtained a great deal of political support from the shipbuilders who had once furnished the fleet that carried Julius Caesar to Britain. They festooned the outer halls of the senate chamber with banners that proclaimed --

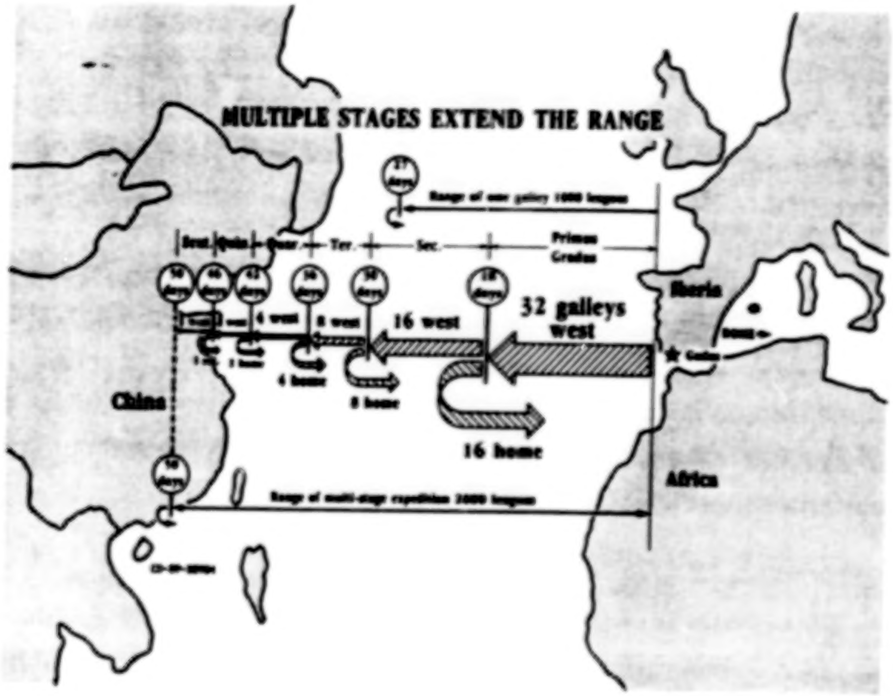
MULTIPLE STAGES EXTEND THE RANGE
Range of one galley 1000 leagues
Sext.
Quin.
Quar.
Ter.
Sec.
Primus Gradus
32 galleys west
16 west
8 west
4 west
16 home
8 home
4 home
China
Iberia
Africa
Gades
Range of multi-stage expedition 2000 leagues

FIGURE 13.

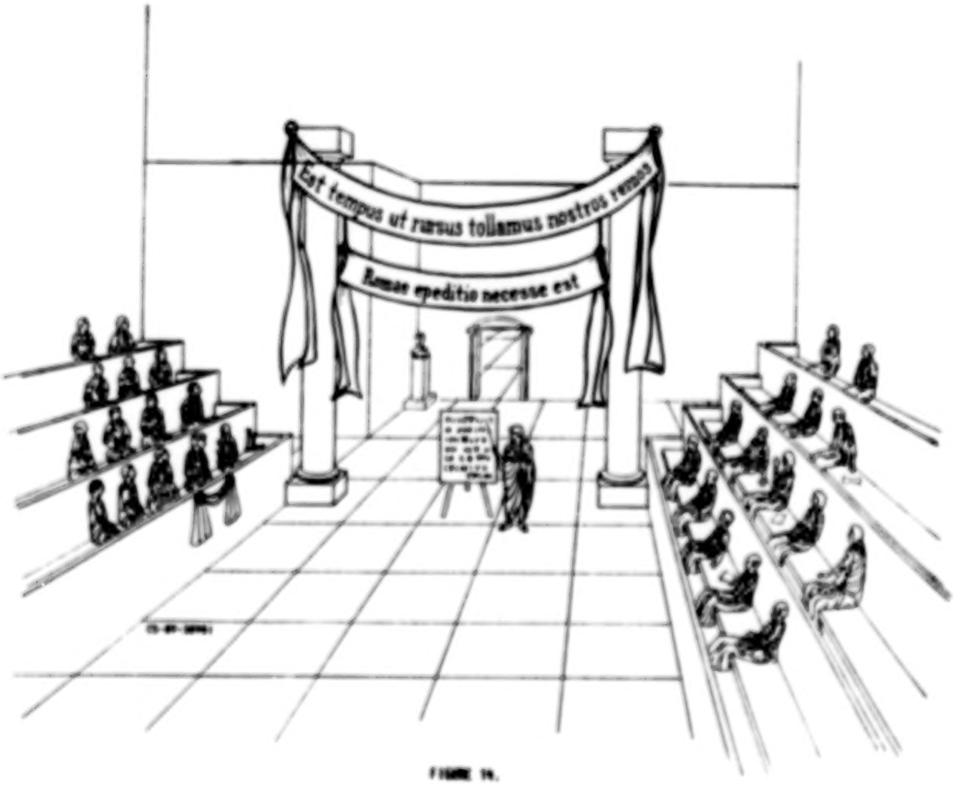
Est tempus ut rursus tollamus nostros remos
Romae epeditio necesse est

FIGURE 14.

"Rome needs the Reconnaissance" and "It's time we raised our oars again."

But when the total costs of the mission were presented ( table III ), the opposition gave way to a clamor. The senators could not understand going to the expense of outfitting 32 ships for a voyage that would actually be completed by 1, or, at the most, 4 ships.

"Isn't your proposal just a little bit gold-plated?" asked Flattus Flavius, the ranking senator on the floor.

"Yes, it is gold-plated," replied Lucius Marcellus, "That is the only way we can do it when we really need to have gold, but can't afford it . . . "

With 32 galleys under stroke, the operating costs were outrageous. Consider the anticipated charges for provisioning the ships -- food, wine, fresh water and casking, not to mention wages for the crew ( they would all have to be volunteers ).

The opposition was vocal and the criticisms were hard to answer. How could these expenditures be justified? Varsovian's proposal violated the basic rule that governs every enterprise where public monies are involved: Where expenditure is great, great risk is not tolerated; where risk is great, great expenditure is not tolerated.

"It is nothing but a publicity stunt," some said. "Take the army on a boat ride to China?" With riots at home and rebellions abroad, there was no way to justify committing all those troops to such a speculative expedition. After all, Varsovian couldn't guarantee success. Could he show a tangible benefit? Maybe after the campaign was finished, but certainly not within the next fiscal year. . .

"What will they do when that last galley finally gets there? Conquer all of China with one cohort?" snorted Caius Crassus. "If you're going to take all those ships to begin with, why not just keep going and invade the place while you're at it? It wouldn't be any more expensive than the fiasco you have proposed!"

For several hours the debate raged on. No decision was reached. But they agreed to appoint a committee to study the plan further and subject it to a cost/benefit tradeoff analysis to see if the mission could be reoptimized

## EXPEDITION TO THE ORIENT ACROSS THE WESTERN SEA

| GRADUS | | Primus | Sec. | Tertius | Quartus | Quint. | Sextus | TOTAL |
|---|---|---|---|---|---|---|---|---|
| Number of Galleys Each Stage | | XVI | VIII | IV | II | I | I | XXXII |
| Number of Days at Sea | | XXXVI | LX | LXXII | LXXXIV | XCII | C | |
| OPERATING EXPENSES | | (mille sestertium) | | | | | | |
| WAGES | oarsmen | $\overline{IV}$DCCCXC | $\overline{III}$CMXC | MMCD | MCD | DCC | DCCXIX | $\overline{XIV}$CLXXX |
| | officers | MDCCCLX | MDLX | CMXX | DXL | CCXC | CCCXX | $\overline{V}$CDL |
| CONSUMEABLES | | | | | | | | |
| meat | | MCXI | CMXII | DLVI | CCCXX | CLXXXI | CXCV | $\overline{III}$CCLXXV |
| bread | | DLXVI | CDLXIX | CCCLXXXI | CLXIV | XC | CIX | MDCLXXX |
| goat butter | | LXXIV | LXIV | XXXV | XXVI | X | XII | CCXXI |
| black olives | | XLII | XXXIII | XXII | XIV | XII | XIV | CXXXVIII |
| grapes | | CXL | CXXII | LXIII | XLIX | XIX | XIX | CDXII |
| zucchini | | LXXXIV | LXXIX | XLVII | XXI | XII | XX | CCLXIII |
| pasta | | CCXXII | CLXXXI | CXI | LXVIII | XLVI | XLV | DCLXXIII |
| water (incl. cashing) | | CCXL | CCXIV | CXXIII | LXIX | XXIX | XXXI | DCCVI |
| wine | | CCCXII | CCLVI | CX | XCIX | LII | LVI | DCCCLXXXV |

TABLE II.

## IMPERIAL TREASURY EXPENDITURE

## EXPEDITION TO THE ORIENT ACROSS THE WESTERN SEA

**FIXED COSTS**

Conversion of 32 troopships to extended range configuration $\overline{LXXVIII}$CXLIV

**OPERATING EXPENSE**

| WAGES | |
|---|---|
| oarsmen | $\overline{XIV}$CLXXX |
| officers | $\overline{V}$CDL |

| CONSUMEABLES | | TOTALS | (mille sestertium) |
|---|---|---|---|
| meat | $\overline{III}$CCLXXV | Fixed Cost | $\overline{LXXVIII}$CXLIV |
| bread | MDCLXXX | Operating Expense | $\overline{XXVIII}$DCCCIII |
| goat butter | CCXXI | material | $\overline{XLVII}$CCCXXV |
| black olives | CXXXVIII | burden | $\overline{XII}$DCCCXXXI |
| grapes | CDXII | labor | $\overline{LXXXVIII}$DCCXXXIV |
| zucchini | CCLXIII | overhead | $\overline{XXXV}$CXCIV |
| pasta | DCLXXIII | G + A | $\overline{LXXXII}$CCXCIX |
| water (incl. cashing) | DCCVI | cost of money | $\overline{VIII}$DLXXIV |
| wine | DCCCLXXXV | fee | $\overline{LVII}$CMXLIII |
| | | Grand Total | $\overline{CDXLIV}$CCXXVII |

TABLE III.

for a reduced range of performance parameters and budget constraints. It was at that point that Varsovian turned on his heel and marched out of the senate chamber in disgust.

In the end, they voted to table the issue until a more decisive mandate could be established.

Which brings us back to that brooding figure standing on the pier. Bitter, disillusioned and cynical, he stares out to the sea dancing on the horizon past the breakwall. His eyes pierce the afternoon sunlight, but they are blinded by disappointment. How can the Empire continue if it is not bold enough to mount even this modest expedition? When men and nations no longer dare to dream, what can the future hold?

But as he stands there staring out to sea, he fails to see a most marvelous thing taking place right in front of him. A graceful Arab dhow ( fig. 15 ), with her lateen rig and deep keel, her sharp prow and delicate forefoot biting cleanly into the waves, is threading its way out of the harbor close-hauled, beating upwind toward the breakwall opening and the open water beyond.

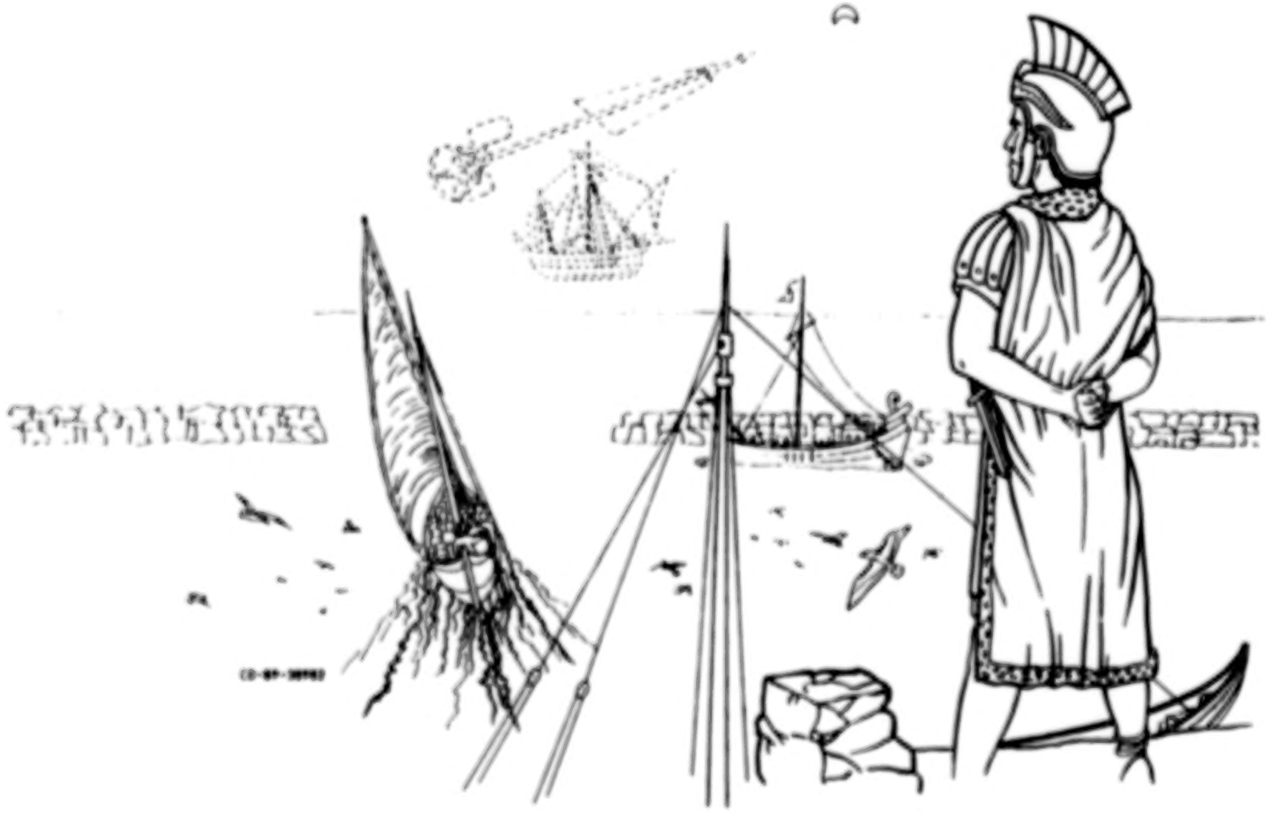

FIGURE 15.

## V. Panel Discussion Groups

- Non-nuclear Propulsion
- Tethers
- Power
- Space Resources
- Speculative Propulsion
- Robotics
- Nuclear Propulsion
- Interstellar Travel
- Space Advocacy

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**Non - Nuclear Propulsion**
**Summary of Group Discussion (4/4/90)**
**Vision 21 Symposium - Cleveland 1990**
**Moderators: Mark Klem**
**Stuart Fordyce**
**Note-taker: Mike Binder**

In order to focus the discussion, it was decided that the group should use the Human Exploration Initiative (mission to Mars) as a framework. It was further decided that mission requirements rather than technological innovation should drive propulsion system research.

Specific technologies for consideration included Laser propulsion, Magnetic Sails, and Chemical Propulsion.

Early in the discussion, the group reached general agreement that cost was a major driver in future space missions. It was further agreed that launch to Low Earth Orbit (LEO) or Geosynchronous Orbit (GEO) represents the greatest single expense in most missions. The discussion was therefore focused on creating a robust and inexpensive launch capability.

The first option considered in detail was laser propulsion. Ground based lasers could be used to propel launch vehicles into orbit; the launch vehicles themselves would be relatively simple and would carry only a small amount of propellant. Although the large ground-based lasers would be expensive to construct, this launch system could be very inexpensive to operate. Estimates of expense claim that small payloads could be launched for about $10 per kilogram. The size of the payload to be launched is constrained by the size of the laser. The cost of the laser is likewise dependent on its size and power requirements. For the size of laser that the laser advocates envisioned in the next 10 to 20 years, the payloads would have to be small (20 kg. was used as an example but the real limit was not well defined in this discussion).

The assumed size of the laser launched payload, about 20 kg., represents a major concern with this system. Very few major components for a space station or space vehicle are that small. Even though the cost to launch these payloads is small, collecting and assembling these materials into useful systems will require a robust, and perhaps expensive space infrastructure. Bigger payloads can be launched with bigger laser systems (or a number of ground lasers focused by a series of satellites). This solution will be considerably more expensive, however, such that the cost trade-off with chemical propulsion became a concern.

Other issues addressed were the effect of the atmosphere on the laser beam and the effect of lasers on the environment. By launching payloads from high elevation sites and by using the proper wavelength, the atmospheric attenuation of the beam can be minimized. Turbulence and moisture effects are still under consideration. The impact of the beam on the environment must also be studied.

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This lead to the group's second option for consideration, chemical propulsion. For large payloads, chemical rockets on Heavy Launch Vehicles (HLVs) will probably be cheaper to build and operate than laser launch systems. The trade-off will depend on the size and frequency of the launches. Another consideration here is development time and cost. There is considerably more experience in chemical propulsion than in laser launch systems. The SSX system from Lockheed, et. al., for example, may cost only a billion dollars to complete development and could launch payloads for about $45 per kilogram. Most members of the group agreed that chemically propelled launch vehicles are here to stay, at least in the near future.

For launch, the best option may be to use HLVs for large modules and other heavy payloads, but use laser launch systems for small payloads, fuel, and perhaps personnel.

The remainder of the session was spent discussing space propulsion options and other mission drivers. The requirement for a safe and stable form of space propulsion was raised. Beamed energy and electric propulsion were suggested. Some of the mission drivers discussed were:

- Human Exploration Initiative (Lunar, Mars, and beyond)
- Solar-power generation satellites
- Commercial communication satellites
- Mission to Planet Earth (environmental observation satellites)
- Colonization of Space
- Tourism
- Fast commercial air transportation
- Disposal of nuclear waste (into the sun)

Ultimately, the group felt that this discussion should continue (needed more time for discussion).

# Tether Workshop Summary

Moderator: Joe Kolecki, NASA Lewis

It was generally agreed that ideas for applications of tethers were plentiful, and that the immediate goal is to, first, get the important tether technologies demonstrated in space to open the way for more ambitious uses, and, second, push near-term tether applications into use as "showpiece" demonstrations for what tethers can do. In the long term, one participant suggested that "the tether will be to travel in the solar system what the wheel is to travel on land." But near-term and mid-term applications were emphasized as being the road to application.

### *Issues and Questions*

1. How do tethers fit into the nation's exploration programs?

--multipurpose applications associated with manned and unmanned activities

--near term, mid term, and far-term applications were touched upon.

2. What are the necessary flight demonstrations, and how are they to be funded? (How is interest created and maintained?)

3. What are the most "realistic" near-term activities?

### *Discussion*

Several issues and near- and mid- term applications were discussed. Use of tethers for momentum transfer in Low Earth Orbit (LEO) was discussed, with the potential for a tether extended downward from higher altitudes to serve as a "way station" to reduce the energy required to reach orbit (see figure).

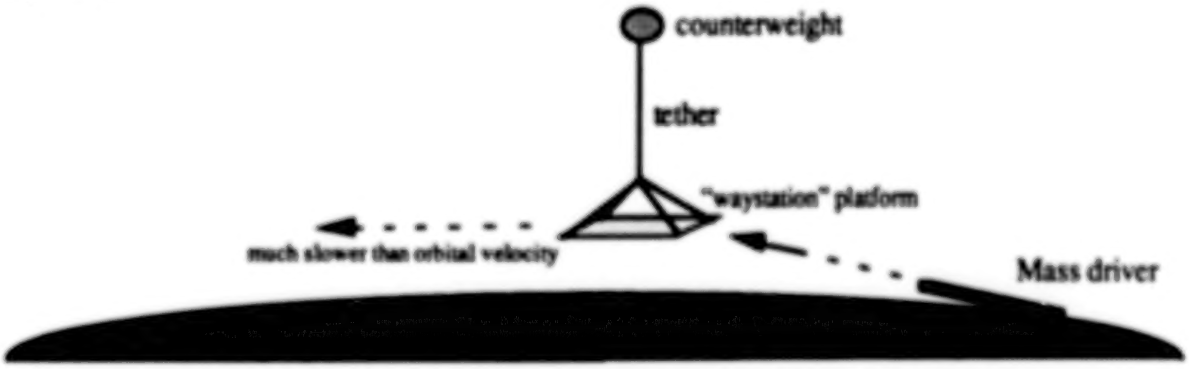

Figure 1. A tether "way station" extending down from higher orbit lowers the required energy to reach LEO. Payloads (e.g., water) might be launched to the way station by a low-cost method such as a mass-driver or gas cannon

### *Technologies needed:*

For tether applications to really take off, the following technologies are needed and must be demonstrated in space:

(1) Deployers, both large and small. These will be demonstrated in the 1991-1992 time frame on TSS-1 and TSS-2 (large tether deployer) and on the Delta-II experiment (Small Expendable Deployment System).

(2) Conducting and non-conducting tethers. To be demonstrated in above tests, but continuous upgrading of technology is needed.

(3) Intelligent control of tether dynamics. One participant mentioned that this may ultimately be done using an "intelligent tether," with the equivalent of "muscles" and "tendons".

(4) Debris effect mitigation. "What happens if the tether gets hit by debris or a micrometeroid?" seems to be the most common objection raised to large-scale tether applications; this issue must be addressed for applications to be common. This should not be a terrifically difficult problem; one solution mentioned (Penzo) was a tether consisting of many parallel strands distanced apart by spacers which can also distribute the load; only if many strands were hit in the same interval between spacers would a tether be likely to break.

(5) Ultimately, tether materials much stronger than Kevlar are going to be needed.

***Applications:* Near Term:**

(1) Many agreed that the most profitable first application could be disposal of waste (primarily waste water) from space station Freedom. The shuttle may not have sufficient capacity to fly down all the waste generated by the station! Waste could be lowered down the gravity well on a tether and released to burn up in the atmosphere; the process of lowering the waste on a tether recovers much of the energy originally used to bring the water up in the first place.

(2) Emergency crew return from the space station, working in a similar fashion.

(3) Milligee laboratory. A tether extending from the space station could easily reach milligees for experiments.

(4) Space station center of mass control. This could be used to center the experimental area in the lowest gee environment.

(5) Variable-gee free-flyer facility. Gravity variable from 0-1G could allow useful experiments.

(6) Refueling of vehicles. Liquids don't easily pour in zero gee; a tether could be a convenient method of producing enough effective gravity for liquid transfer.

(7) Unmanned exploration of planets, moons, asteroids, comets?

***Applications:* Farther Term:**

(7) Artificial gravity for manned interplanetery travel.

(8) Tether transport for applications across the solar system!

***Moderator's Closing Recommendation:***

Flight experiments are required to demonstrate the feasibility of a variety of tether concepts in space (beyond TSS). These experimenst should be funded so that tethers may be established as a viable and important space technology in the next millenium

*--Summary by Geoffrey A. Landis*

# POWER WORKSHOP SUMMARY

Sheila Bailey, Moderator
Barbara Kenny, Recorder

The major concerns addressed by the group were: solar intensity as a limiting factor for photovoltaic systems; pointing accuracy as a limitation for applications of solar dynamic systems; future development of low cost, efficient storage systems such as regenerative fuel cells; transmission of power between source and loads; energy management in terms of utilizing thermal radiation; public policy as a limitation for use of nuclear power systems.

Photovoltaic systems are suitable for Mars applications although the insolation is approximately half that of Earth. Advances in the efficiences of solar cells will extend their applicability to further out in the solar system. Thin-film solar cells on lightweight substrates may offer extremely high power to weight ratios. Some of the concerns with photovoltaic systems include radiation degradration of solar cells and dust covering the cells in planetary applications.

Solar dynamic systems offer potential mass and surface area benefits if thermal advances are realized. Pointing accuracy is perceived to be a significant problem in orbit and contamination on orbit and on planetary surfaces may reduce efficiency.

Solar based power generation require storage capability particularly for lunar applications. Future development of highly efficient storage devices is necessary. Fuel cells appear promising. Demonstration of regenerative fuel cells with cryogenic storage is desirable.

Nuclear power systems have the advantage of not requiring storage, however, public perception of nuclear safety is very negative. A deeper understand of risk versus benefits is required. The advantages of a non-solar dependent power source must be weighed with the disadvantage of required shielding, particularly for manned missions. Massive public education would be required to gain support for space based nuclear power. For future high power requirements there was a general concensus that nuclear power would be necessary. Fusion may provide an attractive future alternative because of reduced shielding requirements, but the technology to utilize fusion is still to be developed. The jury is still out on cold fusion. Theoretical means exist to utilize charged particle fission products to generate electricity directly. Environmentalists are weighing risks and benefits due to global warming and acid rain. The public may be concerned about flying a nuclear reactor but a cold reactor may be safer than radioisotope thermoelectric generator containing plutonium 238. Engineers must be cognizant of their terminology; "walk-away safety" and "passively safe" do not inspire confidence.

Energy management is crucial to development of large power systems to insure efficient utilization of all forms of energy. Consideration should be given to direct use of thermal power without electric conversion. All conversions should be performed at the highest possible temperature to increase radiation efficiency and reduce radiator area. This requires development of high temperature materials and components. Typically for terrestrial applications heat dissipation has not been difficult. However cogeneration has increased overall power plant efficiency by effective energy management. Space based systems will require new thermal management techniques.

Power must be transported from the source to the user in the most efficient way possible. Beamed power and superconducting transmission may provide new methods for increased efficiency. For beamed power, development of higher frequency transmission and reception would reduce system mass. Future development of high temperature superconducting materials may enhance efficiency.

Photovoltaics is a proven technology for present space efforts. Nuclear fission may provide an alternative in about ten years and fusion in about forty years. Anti-matter power sources may be developed for space power needs in the next century. The next stage of space commercialization will require reducing energy costs close to terrestrial levels.

## SPACE RESOURCES/EXP.

The discussion group was attended by 15 persons representing universities, the aerospace industry, and NASA Lewis. The breadth of the topic to be discussed was revealed when a quick survey of the participants indicated about ten distinguishable items which the participants were prepared to discuss. The group agreed to classify the topics into the two categories of space resources and resource utilization.

### SPACE RESOURCES

Space resources were classified as either material resources or energy resources. Material resources received most of the attention of the discussions.

Earth visiting asteroids having a characteristic size of 1 Km or greater are suspected of numbering about 1,000 to 2,000; those 0.1 Km in size may number as many as 400,000. While the content of these asteroids may vary widely, they are known to contain Water, molecular hydrogen, nitrogen, carbon and oxygen in relatively accessible forms; in addition to structurally useful minerals. These resources are in contrast to their lunar counterparts which are weak or devoid of the biologically important elements mentioned above or, as in the case of oxygen, have been found in forms that would be comparatively more difficult to extract. The astronomical community has only recently become aware of the existence of these asteroids through orbiting infrared astronomy that was undertaken for other (deep space) purposes and for which the Earth-visiting asteroids were a source of infrared background noise. Apparently, the reflectivity of these bodies to solar irradiation is inadequate for ready ground-based detection.

The group recommended that this under-considered resource deserves considerably expanded attention. In particular it was recommended that a survey and cataloging of these asteroids be undertaken, even though this would require either a redirection or new capability for on-orbit infrared astronomy. Secondly, the near-Earth asteroids should be strongly considered as a first source of extra-terrestrial material resources, the pursuit of which electric propulsion is very well suited.

A second important class of space resources consists of Earth-manufactured hardware. This resource includes the STS external tank, used propulsion systems, other containment structures, power systems, etc. Nearly any scenario for human exploration includes the extraction of liquid or gaseous materials which would require high integrity containment. Similarly, habitable volumes will be required. The STS external tanks could be adapted for these purposes. Integrated systems designed for high reliability, such as propulsion systems, are the result of costly manufacturing processes on Earth and, for a comparatively small incremental cost, could be upgraded for extended re-use. As a minimum, used propulsion systems could serve as a source of subsystems or component parts. Practically any mass launched from Earth is made of known quantities of high quality materials which should be recycled. The group recommended that the technologies required for the recycling of these systems be developed, and in its anticipation that a system for tracking and cataloging Earth launched hardware be initiated.

Lunar resources, identified during the Apollo missions, are well known, and were not itemized during the group discussions. The group did note some comparisons between resources on the moon and those available from the Earth visiting asteroids. Lunar resources exist at the bottom of significant gravity well, and may therefore be comparatively difficult to utilize away from the lunar surface. Little biomass is available on the moon subjecting any potential exploitation activities to the limitations of importing all life support materials. The most important lunar resource may turn out to be the supply of $He_3$, which would provide an extremely valuable source of nuclear fusion power for Earth or space based operations.

The discussion of the resources available on Mars were limited to the observations that transportation to Mars for colonists may be less of a technological obstacle than the provision of the means to exploit the indigenous resources necessary for extended life support requirements. Again unlike the Moon, all of the resources necessary for life support may well be available on Mars.

The group noted (with some dismay) that time did not allow discussion of energy resources in the solar system including solar energy, gravitational potential, and the kinetic energy of small bodies such as asteroids.

## RESOURCE UTILIZATION

The discussion of resource utilization began with the observation that while we have considerable knowledge of what resources are available, and perhaps as much knowledge of what end use we might envision for these resources, comparatively little is known about effective means for converting resources into useful goods. Because refinement and manufacturing technologies must in this context minimize mass and power consumption, repackaging Earth-based techniques may be at best wasteful or at worst impossible. New technologies must be developed with a fresh perspective that takes full advantage of reduced-gravity phenomena (eg. surface tension dominated fluid systems, suppressed buoyant fluid motion and sedimentation, etc.) and the availability of an essentially infinite supply of high-quality vacuum.

One manufacturing technique was discussed wherein the Earth-based process of metal casting could be replaced by methods of gradual applications of molten material to a levitated seed core using a focussed spray that might be controlled with, for example, electromagnetic fields. Specialized manufacturing techniques need to be considered for the purpose of recycling Earth-launched hardware.

## SUMMARY

Time for summary discussions were quite limited, and began with the observation that the planning for space resource utilization is unlikely to be productive in the absence of long term objectives of human exploration of the solar system. Three classes of long term objectives were identified, namely: scientific exploration and discovery, industry and manufacture for the benefit of Earth, and colonization or settlement. Only with clear plans can a rational plan for utilizing space resources be conceived, and the technology developed to support its execution.

## SPECULATIVE PROPULSION
Tuesday Session, room 3102 Group

Moderator: Mike LaPointe
Note Taker: Marc Millis
Attendance: 11 total

This session touched on a variety of propulsion concepts, the majority of which were variations of existing ideas, and some discussion on unexplored ideas such as "Space Coupling Propulsion." During the course of discussing ideas, a recurring theme surfaced: How do we advance new ideas beyond the conceptual stage into physical tests and eventually into prototype application? These issues evolved into a brief list of recommendations. The ideas and recommendations are summarized below.

### IDEAS:

**Echoing Photon Light Sail Propulsion** (A quantifiable concept):
By using a pair of ideal reflectors (superconducting plates), one mounted on a vehicle, the other anchored to a reaction mass (Moon), photons could be made to echo between the reflectors thus providing multiple photon momentum transfer through multiple photon collisions. The photon source (microwave array) would also be stationed on the reaction mass.

**Space Coupling Propulsion** (A topic for future speculation):
This category was discussed in terms of the unexplored possibilities to satisfy the common objections to the notion of generating propulsive forces by interacting with the structures of space. The common objections include; maintaining conservation of momentum without a reaction mass, unknown nature of the "structure" of space, and need for Grand Unification Theories.

**Velocity Reference Frame for Space Travel** (A new idea?):
Use the Cosmic Microwave Background as a navigation reference for deep space travel (measure velocity relative to mean rest frame of galaxies).

**Inflatable Solar Concentrators** (Existing concept):
Various applications and ideas for demonstrating this concept were discussed. The variety of applications include:

a. Light sails.
b. Beamed energy propulsion.
c. Solar thermal propulsion (Specific impulse up to 1000 sec).
d. Solar thermal power (Specific mass = 0.01 kg/kW).
e. Materials processing (Expl: Lunar resources via solar heat).
f. Climate modification (Focusing extra sunlight onto select areas of Earth).

**Microwave Sail / LINAC Combination** (Variation on an existing idea):
Use solar concentrator to reflect and focus beamed microwaves to power Linear Induction Accelerator, thus providing higher thrust during the initial portions of the journey. The sail becomes the dominant propulsion device as the vehicle reaches its maximum velocity.

**Nuclear Propulsion** (existing concepts):

Discussions on this topic focused on the relative safety and potential benefit of this technology in contrast to popular fears of nuclear energy.

**Earth to Orbit Accelerators** (Existing concepts):

Discussed the variety of existing concepts for providing cargo launch via ground based accelerators:

a. Rail Guns.
b. Mass Drivers.
c. Ram Accelerators.
d. Rotational sling shots.

## RECOMMENDATIONS:

1. **Support Speculative Research**:

Devote higher percentage of the NASA budget, perhaps a constant 6% fraction, to evolve the many high risk and high payoff concepts into physical demonstrations.

a. Several concepts in parallel.
b. Be willing to accept failures.
c. Utilize universities for economical small scale work.

2. **Advance Inflatable Solar Concentrator Technology**:

Because of the simplicity of inflatable solar concentrators and the multitude of potential applications (listed in IDEAS section), increase emphasis and support for developing inflatable solar concentrators.

Space contains a solar energy density of 1.4 GW per square km in Earth orbit that can easily be collected by solar concentrators (specific mass of 0.01 kg/kW), that have demonstrated concentration ratios of 12,000:1. The applications of concentrated thermal power are numerous and include answering the space power needs of the Space Exploration Initiative.

Deployment of even large solar concentrators (up to 100's of meters in diameter) could be demonstrated on earth by use of buoyant inflation gasses as an economical precursor to an in-space demonstration.

3. **Promote Space Nuclear Power and Propulsion**:

NASA should take the lead in promoting the relative safety and potential benefits (Mars transits more feasible) of nuclear power and propulsion (Uranium based), promoting a responsible image and improving public awareness.

## SPECULATIVE PROPULSION
Tuesday Session, Room 3109 Group

Moderator: Joseph A. Hemminger
Note Taker: Joseph A. Hemminger
Attendance: 18 total

The session began with individual introductions and an initial discussion regarding the definition of "Speculative Propulsion". The remainder of the session involved discussions regarding four "advanced" or speculative space propulsion concepts and identified several issues concerning public awareness and acceptance of developing and utilizing these propulsion options.

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**NUCLEAR FISSION PROPULSION:** This space propulsion class is the one with the most extensive technical history. The discussion centered on historical highlights of two major programs and the public issues that resulted in the programs being terminated.

(1) PROJECT ORION. The essential concept, which in summary involves exploding small nuclear charges behind a "pusher plate" on a spacecraft, was a significant effort in the early history of NASA. A variety of propulsion schemes (as many as 12) were explored as the technical details of this concept evolved. It had the potential for propelling large, manned spacecraft on planetary missions and was originally conceived as a joint US/USSR venture.

(2) NERVA. The nuclear-thermal-reactor-powered Nerva engines had the most significant hardware and ground-test history.

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**NUCLEAR FUSION PROPULSION:** The big question with this space propulsion concept is "when?". Three general fusion areas were discussed:

(1) "HOT" FUSION. Significant progress is being made toward energy "break-even" in ground-based hardware concepts. Due to weight constraints, space propulsion requires different concepts, several of which were touched on. One concept involved impinging beams, the reaction of which was used to heat ordinary hydrogen in a long tube and expelled for propulsion.

(2) "WARM" FUSION. This discussion focused on a muon-catalyzed deuterium-tritium fusion. Rates of approximately 170 fusions/muon have been demonstrated in colliding-beam systems. A Russian concept is a hybrid involving a fission blanket surrounding the fusion reactor. The big technical challenge with this concept is the handling of the significant neutron production.

(3) "COLD" FUSION. The jury is still out on this concept. Tritium production has been widely verified but the mechanism causing the production and the source of the heat produced has not been identified. Much of the current work is being performed by chemists (approximately 80%). More work by physicists is needed to overcome skepticism and help characterize the physics of the

process.

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ANTI-MATTER PROPULSION: Significant progress is being made in the production and storage of anti-matter (especially anti-protons) at accelerator facilities around the world. Up to 10^15 anti-protons per year are being produced. The technology exists for fabricating production-oriented facilities at a relatively reasonable cost. Studies have shown that the utilization of anti-matter for high-thrust space propulsion has potential for significant Isp increases. Actual application appears to be quite a few years away. The potential of anti-matter utilization in two other technology areas may help accelerate its application to space propulsion:

(1) MEDICAL. PETScan (Positron-Emmission-Topography) is one possibility. It has been estimated that 10^12 anti-protons could be used to cleanly eradicate 1000 cancerous tumors. This application has potential to grow into a $50B/year business.

(2) MATERIALS. Various material processing and diagnostic procedures are potential applications.

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GRAVITY-WAVE PROPULSION: This is the "farthest-out" concept discussed. Its applicability is potentially dependent on the discovery of "negative matter", the existence of which has not been verified to date.

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PUBLIC ISSUES REGARDING ADVANCED SPACE PROPULSION AND MISSIONS. During the propulsion discussions, several issues were identified and suggestions made addressing their resolution:

(1) NUCLEAR APPREHENSION. Arms race, nuclear reactor (especially waste disposal), and environmental concerns have negatively influenced public support of any space propulsion technology with nuclear implications. Technologists should help provide accurate information to public education efforts in this area.

(2) SCIENCE EDUCATION. Public understanding of scientific phenomena is, in general, weak (e.g., in the nuclear arena, as noted above). Technologists should actively support science education efforts, e.g., getting involved in science education in schools, supporting Science Fair activities and summer student employment programs, etc.

(3) GLOBAL TECHNICAL SUPPORT. Future (especially planetary) space missions are too expensive, and involve such significant technical and social complexity, that they probably won't happen in the near term without international involvement. Technologists should support activities and interaction that will encourage global cooperation in space exploration.

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---JAH (4-11-90)

# SPECULATIVE PROPULSION
## Wednesday Session

Moderator: Marc Millis
Note Taker: Kevin Breisacher
Attendance: 10 total

This session began with a review of the previous day's Speculative Propulsion session (the room 3102 group), and evolved into a discussion about the barriers to advancing speculative ideas into applications. Although many of the barriers are on a scale beyond the scope of any individual to correct, several small scale solutions were discussed.

People within the session shared perspectives on the kinds of barriers that they have encountered in their professions and in their countries. The most notable barrier is the prevalent cultural stigma against pursuing speculative ideas amidst the focus of wanting immediate solutions. It is difficult to spend time on speculating new ideas, as well as to spend resources on evolving established concepts into space flight applications. The prevalent cultural focus is to only refine the well established flight-proven methods.

Another shared barrier is inadequate open access to the right information and people. It is difficult to sort out the valuable information from the enormous bulk of papers and reports in the open literature. Because of its fledgling nature, speculative work is seldom published, hence it is difficult to establish contacts and collaboration with people who share similar technical interests.

Rather than dwelling on the barriers, the group discussed various possible solutions. The group focused on solutions that can be started on a small scale rather than trying to change an entire culture. The most notable solution in that regard is to improve communication. This includes having more "Vision-21" type symposiums, and possibly creating a network of speculative researchers. Another solution is to keep speculative endeavors unofficial and low-key until they are mature enough to survive submission to institutional scrutiny. An added suggestion to this method is to involve peer collaborators in that initial process, hence another need for a network of speculative researchers.

There was more discussion on the details for implementing these suggestions, and these ideas are summarized below:

### RECOMMENDATIONS:

1. Continue to have Vision-21 type symposiums to enhance communication and increase awareness of the value of emerging technological possibilities.

2. Create an environment for the occasional group indulgence in far range speculation, perhaps by hosting a retreat style symposium.

3. Create a communication network.
    a. Papers available as software over computer networks.
    b. Subjects sorted by various criteria:
        i. Subject
        ii. Idea maturity.
        iii. Scope of information (summary versus details).
    c. Use Artificial Intelligence to screen/sort submissions.
    d. List potential collaborators by subject interest/ expertise.
    e. List unsolved problems that are in search of new ideas.
    f. Evolve into a dedicated technical support center to maintain and upgrade network, provide peer reviews, and resolve issues. (Government or Industry?)

4. Work first on small scale unofficial projects with peer collaborators until idea matures enough to submit to institutional scrutiny.

5. Focus on testing or demonstrating ideas with hardware to enhance the believability of new methods.

### Vision 21 Workshop
### April 4, 1990

## SPACE STRUCTURES AND LUNAR ARCHITECTURE

**Moderator: David Spera**
**Notetaker: Gregory Fedor**

The purpose of this workshop was to elicit conversation on the subject of structures and their relationship and use to space and lunar applications. The following questions were posed to the group as a starting point and ideas to give direction to the discussion.

1) What is possible in the next century?
2) Why do we need this?
3) What's next?
4) What's possible in the next 100 to 1000 years?

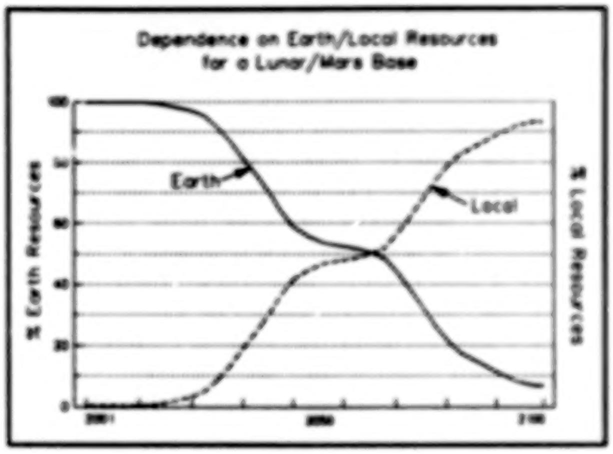
Structures of the type discussed fall into one of these three categories:

1) Biological support. (habitats)
2) Machine maintenance. (garages)
3) Mining and resource conversion (factories and power plants)

The need to house humans and grow food were seen as major forces driving structural design. Since machines will be used extensively for Extra-Vehicular Activity, they will need special buildings constructed to house their maintenance and replenishing facilities. As communities grow, they will be able to take local materials and produce usable goods.

The traditional thinking of how these structures should be designed will not apply. Providing shelter will be more than putting a roof over ones head. It will be a sealed enclosure capable of providing all support for life. The structures used for this purpose will have to be very massive, a feat that on earth is difficult to accomplish due to gravity. Space should prove somewhat easier. On earth there is both the luxury and hinderance of providing temporary support while structures are built. In space no such support is needed but the simple task of keeping a long tube straight and unmoving is difficult. Space Station Freedom should provide valuable data on different construction techniques and obstacles.

Any structure, regardless of it being earth or space based, requires material. At first it was seen that the material will come exclusively from earth and many of the structures will be manufactured or at least designed there. As bases become established, the use of local resources will grow to a point where the majority of the complexes will be self-sufficient. Space structures (ie. space stations) will not have the luxury of self-support for building materials, but it was felt that once bases appear on the moon, mars, and asteroids, the need to rely on the earth will diminish. Below is a graphical representation of how, over the next 100 years, this decrease may occur.


Dependence on Earth/Local Resources for a Lunar/Mars Base

It is felt that initially "low-tech" items such as concrete, will be constructed from local resources while "high-tech" items such as cable, will be brought from earth. Perhaps consumable goods can eventually be made as byproducts of the construction process. Basalt fibers can be woven from the debris left from mining operations on the moon. As the local independence increases, the need to conform to earth ideas for how best to build structures will diminish.

Structures are "alive". Every object we put in space is going to be subjected to a variety of environmental conditions. Questions were raised whether current designs for Space Station Freedom will be able to cope with these stresses. It was found that while it is easy to manufacture close tolerances here on earth, construction for on-orbit operations will have to take into account such things as contraction and expansion due to sun warming. This will lead to a structure that has many small built-in tolerances to allow for movement. These tolerance will accumulate and the resulting movement will always be a problem that is very difficult to counteract.

Very large structures will be subject to many more hazards found in space. There was no practical way seen to provide complete protection from solar flare radiation for these massive structures. The "bomb-shelter" concept was introduced as a way of protecting human space travelers from these dagers.

The question was raised, "who is going to build all this?" Telerobotics should play a major role in getting the initial structures in place. Once humans can reside at the location, then the design of structures will evolve.

## Interstellar Travel

Wednesday Session, combined sessions.

Moderators: G. Scott Williamson, Ron Cull
Note Takers: Douglas Bewley, Mike Binder
Attendance: 26 total

The two sessions were combined into one. The discussion touched on a variety of interstellar concepts, ranging from manned or unmanned flight, sub-light to faster than light travel, interstellar communication and interstellar hazards. A brief list of recommendations on how to keep the interstellar travel dream alive was generated. The ideas and recommendations are summarized below.

### IDEAS:

### MANNED or UNMANNED Flight:

The session started with the question of manned or unmanned flight and why? A consensus was reached to start with unmanned flight which would lead to manned flight. Unmanned missions would have to be autonomous in nature due to the time lag involved for communication with the spacecraft. Advanced unmanned craft could include the Von Neumann type machines which would self replicate and slowly spread their way from star system to star system, preparing the way for manned flight. Manned flight ideas included one-way missions where ships the size of cities would be able to colonize the stellar systems which had previously been determined to contain habitable planets. The idea of slowed consciousness was discussed, supported by medical research showing that this concept might be feasible (Rats lives extended 20%). The concept of "Hybrid" type travel where a probe is launched to a nearby star system, finds a suitable planet, sets up a receiving antenna and then a hybrid "Machine-Human Consciousness" is down loaded from Earth's transmitting antenna to the probe's, thus transporting the Human Species.

### WHY?

The question of why to even go to interstellar space came up early. Several reasons were put forward, the idea that it is man's destiny to explore, that the Earth is getting awfully crowded, that we might meet someone/something out there were discussed.

### FASTER-THAN-LIGHT-TRAVEL:

Different types of faster than light travel schemes were discussed along with the question of whether it is even feasible. It was pointed out that Tachyons are believed to exist and travel at velocities faster than that of light, but would faster than light travel violate causality? Worm Holes have been theoretically predicted to exist and might be ways of by-passing large volumes of space (simulating faster than light travel). Einstein-Rosen-Podolsky experiments in super luminal quantum effects were discussed. The question was raised "Is any one pursuing these possibilities?" A general consensus was reached to keep our minds "open".

### SUBLIGHT PROPULSION:

The Orion project was discussed as a way of reaching low light-type speeds (approx 3% speed of light). The idea of the interstellar Ram-jet, where interstellar hydrogen would be used as propellant for a fusion engine, thus allowing continuous acceleration to relativistic velocities. Since a large inexhaustible fuel supply will be needed, anti-matter power sources could be a source of this power for ion type propulsion schemes. Details of the production of even the small amount of anti-matter needed, and containment of the material for long periods of time have yet to be worked out. Solar sails were discussed as long as the sails were used relatively close to a stellar body. Using the most powerful propulsion systems discussed here, speeds on the order of 90% the velocity of light should be attainable in the near future for unmanned space probes (low mass) while manned flights should be able to obtain speeds close to 50% that of light's.

**NON-PROPULSION:**

The question; "Have we ever been visited by interstellar material?" was discussed. This question led to the possibility of there existing comets that travel between stellar systems, since we know that the Ort cloud extends deep into interstellar space. If there are comets that travel the interstellar voids of space and they can be identified and analyzed, then information might be ascertained concerrning their system of origin. It may also be possible to "hitch" a ride on such a comet and thus "travel" to the stars.

**COMMUNICATION:**

The key to interstellar travel is communication link to Earth. This point was an underlining concept throughout the entire workshop. A recommendation to develop the existing communication technology so that interstellar communication is possible was put forth, this would than enhance the possibility of interstellar travel. High powered transmitters, on the order of gigawatts, and large antennas, approximately ten Earth diameters, are needed to link the small unmanned probes (small transmitting antennas) with the earth. One way to help in the communication area would be to create transmission lines out of asteroid materials via relativistic extrusion techniques. A medium size asteroid would be enough material to reach from earth to the nearest star. Through these lines video, radio and other forms of communication would be possible.

**INTERSTELLAR INTERMEDIATES:**

The question was raised about objects between earth and the nearest stars. Objects ranging in size from small particles to "Brown Dwarfs" about the size of Jupiter have been predicted. These Brown Dwarfs could be used as "way stations" for the proposed interstellar missions. Speculation as to how many of these objects might exist (there density) was discussed. The point was made that it cannot be too dense since we don't see the stars "winking out" when these objects come between the earth and the stars.

**INFORMATION:**

Listed below are several papers on the subject of interstellar travel. The papers can be obtained by contacting V-21, the author, or through certain libraries. Also listed here are the people who attended the interstellar workshops and their respective organizations.

**RECOMMENDATIONS:**

Some recommendations: In order to stimulate the public interest hold a contest (with PRIZES!) to come up with the "best way of traveling between the stars". This would be held yearly for high school students. Continue working on the interstellar ramjet scoop which needs to be researched. Continue in research on anti-matter power sources. Investigate possible faster than light travel. Target velocities in the next 100 years should be a minimum of 50% c for manned missions and 90% c for unmanned missions. Research ways to use potential wealth as a justification for a new age of exploration. This is analogous to Columbus' journeys, where wealth from "spices" was the ultimate reason he was funded; come up with the "spices" for interstellar travel.

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## ROBOTICS SESSION 1B

The session started with a discussion of what a "robot" actually is. The working definition became: "A robot is a mechanism which is used in place of otherwise needed human presence".

In general robotics will be needed becuase:

1. They are cost-effective
2. They will enhance future missions
3. They increase safety and reliability
4. Robots are disposable; there is no need to return them to earth.
5. Robots can do work which would otherwise be too tedious for human beings.
6. By designing and simplefying tasks for robots, the essential elelments of needed processes are developed.

There was some discussion as to how specialized robotic systemns should actually become. The general feeling was that general purpose robots might be best for space missions; each being equipped to use specialized "end-effectors". In other words, use general robots to position and manipulate specialized mission-specific robotic end-effectors. This approach captures Paul MacCready's notion that, "The task designs the robot". Also, it captures the notion of using a common inventory of robots each with consistant operating characteristics and procedures.

Generalized robots will be desirable for long-duration missions where self-assembly and and/or self-replication would be desirable. It was felt that this type of robot needed the most discussion since it had the broadest range of applicability. The vision was that that many such "dumb" robots could be used with distributed inteligence- or adaped as needed through the use of special "smart" appendages.

Several issues were discussed as follows:

- Rethink trade-offs between reliability and redundancy
- Devlop robotic "pioneers" as a substitute for human presence.
- Rethink trade-offs between autonomy versus telepresence

## ^WRobotics 1B^W

What is a robot?

A robot is a mechanism which is used to avoid the necessity of a human presence. For example, a thermostat is a simple sort of robot.

Why do we need robots?

1. They are cost effective.
2. Robots are capable of mission enhancement.
3. They increase safety and reliability.
4. Robots are "disposable", there is no need to return them to Earth.
5. Robots can do jobs that humans would find too tedious.
6. By simplifying tasks for robots, one is more aware of the essential elements of a process.

Should robots be specialized or generalized?

"The task designs the robot." - Paul MacCready
Specialized robots are designed to be end effectors.

Generalized robots can be designed for self-assembly and/or self replication.

Distributed intelligence - a "dumb" robot could have special interchangable "smart" appendages.

Issues:

1. Rethinking reliability versus redundancy.
2. Robotic "pioneers" could prepare for human presence.
3. Autonomy versus telepresence.
   Time delay to the moon and back - approx. 0.5 sec.
   Time delay to Mars and back - approx. 20 min!
4. Robots would need scalable, reliable general purpose joints. These joints would need "fail-safe" mechanisms to lock the joint in the event of a failure.
   No workable direct-drive ball joint has been engineered yet.
5. What is the minimum size possible (or desirable)?
   How small would a system have to be in order to effectively utilize extraterrestrial resources?
6. Should robotic exploration be done with venture capital? It might be profitable enough that the government wouldn't need to be involved.
7. "Bar-coding" on all small hardware might overcome the limitations of robotic vision.
8. Other robotic architectures would have to be designed to meet accuracy, precision, and power density demands. Instead of limbs, an octopus, snake, elephant trunk, or flying bee configuration might be

more appropriate.

9. Man-machine interfaces for telepresence:
    - virtual reality (near term)
    - force and torque feedback (sometimes more effective than high-resolution graphics)
    - connecting directly to the operator's brainwaves with "squids"
10. Using the prize process to advance technology.
11. Standardized robot programming languages - Many companies have their own "in-house" programming languages.

## Nuclear Propulsion

3109 DEB, 4-4-90

This panel discussion was attended by approximately 20 people. Not surprisingly, all were proponents. A major area of concern was the actual, and publicly perceived, environmental effects of nuclear propulsion. There was a consensus on two points relative to this issue:

- Radioactive matter is a natural component of the space environment. The earth's magnetosphere for example, is composed almost entirely of a form of radioactive matter. However, it was pointed out that one needed to acknowledge the relatively small density of any kind of matter in space. One person indicated that all the material in the magnetosphere was only sufficient to fill an olympic size swimming pool.

- Concern for environmental effects of nuclear propulsion ought to be a comparative thing. To the extent that nuclear propulsion reduces the required mass in LEO it may greatly diminish the adverse environmental effects of chemical launch activities.

# Interstellar Travel
# Panel Discussion

Wednesday Session, combined sessions.

Moderators: G. Scott Williamson, Ron Cull
Note Takers: Douglas Bewley, Mike Binder
Attendance: 26 total

The two sessions were combined into one. The discussion touched on a variety of interstellar concepts, ranging from manned or unmanned flight, sub-light to faster than light travel, interstellar communication and interstellar hazards. A brief list of recommendations on how to keep the interstellar travel dream alive was generated. The ideas and recommendations are summarized below.

## IDEAS:

### MANNED or UNMANNED Flight:

The session started with the question of manned or unmanned flight and why? A consensus was reached to start with unmanned flight which would lead to manned flight. Unmanned missions would have to be autonomous in nature due to the time lag involved for communication with the spacecraft. Advanced unmanned craft could include the Von Neumann type machines which would self replicate and slowly spread their way from star system to star system, preparing the way for manned flight. Manned flight ideas included one-way missions where ships the size of cities would be able to colonize the stellar systems which had previously been determined to contain habitable planets. The idea of slowed consciousness was discussed, supported by medical research showing that this concept might be feasible (Rats lives extended 20%). The concept of "Hybrid" type travel where a probe is launched to a nearby star system, finds a suitable planet, sets up a receiving antenna and then a hybrid "Machine-Human Consciousness" is down loaded from Earth's transmitting antenna to the probe's, thus transporting the Human Species.

### WHY?

The question of why to even go to interstellar space came up early. Several reasons were put forward, the idea that it is man's destiny to explore, that the Earth is getting awfully crowded, that we might meet someone/something out there were discussed.

### FASTER-THAN-LIGHT-TRAVEL:

Different types of faster than light travel schemes were discussed along with the question of whether it is even feasible. It was pointed out that Tachyons are believed to exist and travel at velocities faster than that of light, but would faster than light travel violate causality? Worm Holes have been theoretically predicted to exist and might be ways of by-passing large volumes of space (simulating faster than light travel). Einstein-Rosen-Podolsky experiments in super luminal quantum effects were discussed. The question was raised "Is any one pursuing these possibilities?" A general consensus was reached to keep our minds "open".

### SUBLIGHT PROPULSION:

The Orion project was discussed as a way of reaching low light-type speeds (approx 3% speed of light). The idea of the interstellar Ram-jet, where interstellar hydrogen would be used as propellant for a fusion engine, thus allowing continuous acceleration to relativistic velocities. Since a large inexhaustible fuel supply will be containment of the material for long periods of time have yet to be worked out. Solar sails were discussed as long as the sails were used relatively close to a stellar body. Using the most powerful propulsion systems discussed here, speeds on the order of 90% the velocity of light should be attainable in the near future for unmanned space probes (low mass) while manned flights should be able to obtain speeds close to 50% that of light's.

**NON-PROPULSION:**

The question; "Have we ever been visited by interstellar material?" was discussed. This question led to the possibility of there existing comets that travel between stellar systems, since we know that the Ort cloud extends deep into interstellar space. If there are comets that travel the interstellar voids of space and they can be identified and analyzed, then information might be ascertained concerrning their system of origin. It may also be possible to "hitch" a ride on such a comet and thus "travel" to the stars.

**COMMUNICATION:**

The key to interstellar travel is communication link to Earth. This point was an underlining concept throughout the entire workshop. A recommendation to develop the existing communication technology so that interstellar communication is possible was put forth, this would than enhance the possibility of interstellar travel. High powered transmitters, on the order of gigawatts, and large antennas, approximately ten Earth diameters, are needed to link the small unmanned probes (small transmitting antennas) with the earth. One way to help in the communication area would be to create transmission lines out of asteroid materials via relativistic extrusion techniques. A medium size asteroid would be enough material to reach from earth to the nearest star. Through these lines video, radio and other forms of communication would be possible.

**INTERSTELLAR INTERMEDIATES:**

The question was raised about objects between earth and the nearest stars. Objects ranging in size from small particles to "Brown Dwarfs" about the size of Jupiter have been predicted. These Brown Dwarfs could be used as "way stations" for the proposed interstellar missions. Speculation as to how many of these objects might exist (there density) was discussed. The point was made that it cannot be too dense since we don't see the stars "winking out" when these objects come between the earth and the stars.

**INFORMATION:**

Listed below are several papers on the subject of interstellar travel. The papers can be obtained by contacting V-21, the author, or through certain libraries. Also listed here are the people who attended the interstellar workshops and their respective organizations.

**RECOMMENDATIONS:**

Some recommendations: In order to stimulate the public interest hold a contest (with PRIZES!) to come up with the "best way of traveling between the stars". This would be held yearly for high school students. Continue working on the interstellar ram-jet scoop which needs to be researched. Continue in research on anti-matter power sources. Investigate possible faster than light travel. Target velocities in the next 100 years should be a minimum of 50% c for manned missions and 90% c for unmanned missions. Research ways to use potential wealth as a justification for a new age of exploration. This is analogous to Columbus' journeys, where wealth from "spices" was the ultimate reason he was funded; come up with the "spices" for interstellar travel.

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# INTEROFFICE MEMORANDUM

**Date:** 11-Jun-1990 11:02am EST
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**TO:** Jack Harper ( PAPER MAIL )

**Subject:** Vision 21 - Space Advocacy Panel Report

Approximately 15 people assembled for this panel. They included NASA program managers, aerospace engineers, press and public relations people and representatives of space advocacy groups. The divergence of backgrounds produced lively multi-faceted discussions.

The group first considered the question of optimal strategies for advocates of an expansion or redirection of our space effort to follow. There was agreement that not only was a strong grass roots effort vital but that presidential leadership would be necessary i.e. both top down and bottom up efforts are needed. Profit, benefits, and the romance of space travel were identified as broad categories of selling points. There were a number of difficulties to over come that were identified. They included public malaise, the lack of credibility of technologists and the public's limited scientific literacy. It was suggested by some of the panel that space program advocates would be most effective if they were realistic about human political institutions and avoid trying to sell utopias.

The issue of whether the civilian space effort should be a public or a private enterprise activity was also raised. Both sides of this question were defended and it seemed that, not surprisingly, the consensus reached encompassed both. It was felt that a key to wider use of space is cheap transportation.

There was also general agreement that NASA needs a better interface with the interested public and that our vision should be more international.

Submitted:

Paul R. Aron
Moderator and Rapporteur

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# Vision-21 Symposium 4/5/90

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NASA
National Aeronautics and
Space Administration

# Report Documentation Page

**1. Report No.**
NASA CP-10059

**2. Government Accession No.**

**3. Recipient's Catalog No.**

**4. Title and Subtitle**
Vision-21: Space Travel for the Next Millennium

**5. Report Date**
April 1990

**6. Performing Organization Code**

**7. Author(s)**

**8. Performing Organization Report No.**
E-5838

**10. Work Unit No.**
505-62-21

**9. Performing Organization Name and Address**
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135-3191

**11. Contract or Grant No.**

**13. Type of Report and Period Covered**
Conference Publication

**12. Sponsoring Agency Name and Address**
National Aeronautics and Space Administration
Washington, D.C. 20546-0001

**14. Sponsoring Agency Code**

**15. Supplementary Notes**
Editor, Geoffrey Landis, Sverdrup Technology, Inc., Lewis Research Center Group, Brook Park, Ohio.

**16. Abstract**

The symposium Vision-21: Space Travel for the Next Millennium was held at the NASA Lewis Research Center on April third and fourth, 1990. The purpose of the symposium was to stimulate innovative thinking in the sciences and technologies which will be required for exploration and development of space over the next thousand years. The keynote speaker was Dr. Robert Forward. Other invited speakers were Marvin Minski, Joel Sercel, Theodore Taylor, John Anderson, and Paul MacCready. The proceedings consists of transcripts of the invited talks and the panel discussion by the invited speakers, summaries of workshop sessions, and contributed papers by the attendees.

**17. Key Words (Suggested by Author(s))**
Vision-21
Space travel
21st century

**18. Distribution Statement**
Unclassified – Unlimited
Subject Category 12

| 19. Security Classif. (of this report) | 20. Security Classif. (of this page) | 21. No. of pages | 22. Price* |
|---|---|---|---|
| Unclassified | Unclassified | 599 | A25 |

NASA FORM 1626 OCT 86

*For sale by the National Technical Information Service, Springfield, Virginia 22161

★U.S. GOVERNMENT PRINTING OFFICE 1991 -[illegible]

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DATE

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JUN 20 1991